A Comparative Investigation of a Ground and Air Launch of LEO-Bound Microsatellites

By

Michael Labib
B.Eng., Carleton University
M.Sc., International Space University

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To Diane and Reda

ABSTRACT

Microsatellites have shown that it is possible to execute missions effectively and rapidly. Despite the current oversupply of launch capacity, there are no operational dedicated microsatellite launch vehicles. This study investigated a ground and air launched vehicle based on the Orbital Express conceptual launcher. Analytical considerations were largely based on a commercial trajectory optimization software package and were validated with numerical models. Favourable ground and air launch circumstances were identified while accounting for technical and operational aspects.

Ground and air launch payload capability results to a 600 km and 60° circular orbit were 55 kg and 35 kg with 0.34% and 0.74% payload mass fractions respectively. Although the ground launch was often preferred, the air launch was better suited in some instances, particularly for responsive services. In developing a Canadian micro class access to space, this study proposed to operate a ground launched vehicle followed by an air launched system.

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NOMENCLATURE

ACRONYMS

ASTOS	AeroSpace Trajectory Optimization Software
ASTP	Advanced Space Transportation Program
ATK	Alliant Techsystems Incorporated
Bristol	Bristol Aerospace Limited
CFB	Canadian Forces Base
COTS	Commercial Off-The-Shelf
DARPA	Defence Advanced Research Projects Agency
DOF	Degree of Freedom
EELV	Evolved Expendable Launch Vehicle
ERV	Expendable Rocket Vehicle
ESA	European Space Agency
FALCON	Force Application and Launch from the Continental United States
GEO	Geosynchronous Orbit
HLV	Heavy Launch Vehicle
ICBM	Intercontinental Ballistic Missile
ISS	International Space Station
LEO	Low-Earth Orbit
LOX	Liquid Oxygen
MicroSat	MicroSat Launch Systems Incorporated

MicroSpace	International MicroSpace Incorporated
MIPCC	Mass Injection Pre-Compressor Cooling
MPV	Main Propulsion Vehicle
MSFC	Marshall Space Flight Centre
NASA	National Aeronautics and Space Administration
RASCAL	Responsive Access Small Cargo Affordable Launch
RLV	Reusable Launch Vehicle
SPHADS	Small-Payload High-Altitude Delivery System
SpaceX	Space Exploration Technologies
SRM	Solid Rocket Motor
SSLV	Standard Small Launch Vehicle
SSO	Sun-Synchronous Orbit
SUA	Special Use Airspace
TVC	Thrust Vector Control
t/LAD	Trapeze-Lanyard Air Drop
US	United States

SYMBOLS

A_e	Exit nozzle area (m^2)
a	Semimajor axis (m)
a_n	Acceleration component normal to the flight path (m/s^2)
a_t	Acceleration component tangent to the flight path (m/s^2)

C	Centre of curvature
C_D	Drag coefficient
C_L	Lift coefficient
D	Drag force (N)
g	Gravitational acceleration of the Earth (m/s^2)
g_0	Gravitational acceleration of the Earth at sea level (m/s^2)
h	Altitude (m)
I_{sp}	Specific impulse (s)
L	Lift force (N)
M	Centre of mass
m	Mass (kg)
m_0	Initial mass (kg)
m_f	Final mass (kg)
P	Centre of pressure
p	Ambient air pressure (Pa)
Q	Thrust application point
q	Dynamic pressure (Pa)
R_0	Radius of the Earth at the equator (m)
R_E	Radius of the Earth (m)
r	Position (m)
S	Reference area (m^2)

T	Thrust (N)
t_0	Time at ignition (s)
t_b	Time at burnout (s)
$\hat{u}_n$	Unit vector normal to the flight path
$\hat{u}_t$	Unit vector tangent to the flight path
V	Velocity (m/s)
x	Downrange distance (m)

Greek Letters

α	Angle of attack (°)
β	Azimuth (°)
ΔV	Change in velocity (m/s)
ε	Thrust angle (°)
φ	Latitude (°)
γ	Flight path angle (°)
κ	Radius of curvature (m)
μ_E	Geocentric gravitational constant of the Earth (m^3/s^2)
ω	Angular velocity (rad/s)
ρ	Atmospheric density (kg/m^3)
θ	Pitch angle (°)

CHAPTER 1

INTRODUCTION

Microsatellites have shown that it is possible to execute missions very effectively and rapidly at low cost. Applications such as forest fire detection, urban planning, land use, crop monitoring, precision farming, environmental studies and geological mapping are among the numerous and important science applications which can be performed by microsatellites. This study comparatively investigated a ground and air launch to orbit. The major focus of this report was the capability of launching micro (10 kg to 100 kg) class satellites. The extension of the launch capability to the small satellite class (100 kg

to 1000 kg) was also addressed in the literature review.

In this study, both performance and operational comparisons of a ground and air launch were examined. The performance segment was based on the Orbital Express conceptual launch vehicle and investigated criteria such as payload capability and trajectory which were compared to similar studies in literature. These considerations were largely substantiated by the AeroSpace Trajectory Optimization Software (ASTOS) in addition to in-house numerical models. ASTOS was developed at the Institute of Flight Mechanics and Control of the University of Stuttgart and financed by the European Space Agency (ESA). This commercial software is a design tool for trajectory simulation and optimization of launch vehicles [Well, 1997]. The operational component details key aspects from integration to launch. Ultimately, this study identified favourable ground and air launch circumstances with consideration for a multitude of technical and logistical factors.

An assessment of the current state of the world launch business is followed by a review of existing space launch technologies relevant to small and microsatellites. The major objectives of this chapter include setting the stage and making the case for the viability of a microsatellite launch system.

1.1 Background

There is currently an oversupply of launch capacity in the world and a large number of launch vehicles are suffering from the lack of launch orders. The anticipated growth in the number of spacecraft to be launched during the period of 1998 to 2000 led to several new launch vehicle development programs. Large mobile telecommunication constellations such as Iridium with 66 satellites and broadband multimedia systems such as Teledesic with the initially proposed 840 satellites encouraged the industry to develop new launch vehicles. Market surveys conducted during this period predicted a shortage of launch capacity and radical changes in the aerospace industry. The unique combination of the end of the cold war, strong capital markets and technological advancements in the telecommunications industry fuelled these predictions [Caceres, 1999]. With the unexpected commercial shortcoming of these large constellations, most of the launch vehicle development efforts such as Roton, Kistler, and Eclipse did not go further than the concept design phase [Caceres, 2001]. Between 1965 and 1990, there were over 100 annual launches, peaking beyond 120 from 1975 to 1985. According to a 1999 study, an average of 80 launches per year was estimated during the next decade [Caceres, 1999]. This was actually an optimistic guess. The total number of launches in 2002 through 2006 was 65, 63, 54, 55 and 66 respectively [Futron Corporation, 2007]. These figures can easily be satisfied by the world's current fleet of launch vehicles, leaving a very small market share for the new launch systems. The same study assumed that the cost of launching a payload to orbit would remain high during the next decade: between United States (US) $4000 to $16,000 per kg to Low-Earth Orbit (LEO) and US $16,000 to

$30,000 per kg to Geosynchronous Orbit (GEO). A significant decrease in launch cost would definitely revolutionize the access to space and change the market predictions [Caceres, 1999]. However, there is no current program promising a substantial reduction of the cost, especially after the premature ending of the Lockheed Martin's X-33 or VentureStar program with a goal of ten–fold decrease in the launch cost in ten years.

Reusability has been proposed as a solution for the high cost of access to space. Launch vehicle companies hope to decrease the operating cost by using their vehicles as frequently as possible. This was the main drive behind Reusable Launch Vehicles (RLV). There are currently a number of conceptual studies, but no operational RLVs available except the US Space Shuttle. The low cost RLV technologies proposed in recent years lack funds to develop technology, complete testing and build flight models [Caceres, 2001]. In order to sufficiently reduce launch prices such that the demand is stimulated, RLVs will have to be launching 25 to 50 times per year. It is difficult to justify a need for an RLV with the current annual rate of launches; in addition to higher development cost, RLVs are more expensive to operate and maintain. It will also take time to build customer confidence in RLV programs which further delays the entry service of the reusability concept as a viable solution for access to space [Caceres, 1999].

With regards to new expandable launchers, high development costs of new launchers such as the European Ariane 5 and the US Evolved Expendable Launch Vehicle (EELV) programs (Lockheed-Martin's Atlas V and Boeing's Delta IV) indicate that a decrease in

launch cost is unlikely for the near future. Development costs can be decreased with reduced testing but this can lead to unforeseen failures and scheduling delays. The US Defence Department faced the challenge of covering the unexpected cost growth of the EELV program. It has been reported that the Pentagon had to absorb a 50% cost increase on the program due to the poor commercial market and contractual problems [Wall (a), 2003]. It was clear that the US government would be the main customer for the EELV program and a commercial exploitation would be very difficult. However, as stated in a separate article, assured access to space "is too important to be a budgetary issue" [Wall (b), 2003]. It is widely accepted that space development programs have to be different from other projects as a result of the unique demands of cutting-edge technology and strategic importance.

Despite the current oversupply of launch capacity in the world, dedicated launch vehicles for small and microsatellites are very limited. A dedicated launcher eliminates multi-payload integration risks, the need to find another customer to share the ride and can diminish scheduling delays. Many countries including ESA members, US, Israel, Russia, and Brazil are all developing low cost small launch vehicles, indicating a clear interest and shift towards smaller satellites and rapid launch.

1.2 Launching Small and Microsatellites

While the heavy-lifting launch vehicles are dominating the market, small and low cost launchers are becoming increasingly interesting with the current trend of building and

launching smaller satellites. Small and microsatellites have shown that it is possible to execute both civil and military missions very effectively and rapidly at low cost. However, the launch fee is still too high in comparison to the low cost of small and microsatellites. Currently, launch costs can double the total mission price leading to significant restriction for the use of such satellites. One of the key design drivers towards realizing such small launch systems is therefore cost effectiveness. The main question remains the same, whether these launch systems will be economically feasible and whether they will be able to compete in the world market. Small and microsatellites have often been launched using the former Soviet Union surplus Intercontinental Ballistic Missiles (ICBM). These vehicles are highly reliable and are typically offered at a lower cost in comparison to Western launchers. Relatively inexpensive launchers such as Pegasus XL, Minotaur, Taurus XL, or piggy-back launch as secondary payloads on large launchers have also been used. However, secondary payload launches either do not allow sufficient mission flexibility or the launch cost is more than that of building the satellite. In summary, the launch price is consistently a problem for the small and microsatellite business and there is a strong demand for low cost launchers.

The National Aeronautics and Space Administration (NASA), US government customers and the US military cannot use the Soviet Union ICBMs due to export-control rules and buy-American policies. As a result of these restrictions and growing interest in small satellites by the US military, the US Defence Advanced Research Projects Agency (DARPA) introduced the Force Application and Launch from the Continental US

(FALCON) program to create a breakthrough small launch vehicle. This vehicle would deliver a 450 kg payload into LEO for under US $5 million by 2010 [Morring, 2003]. A responsive access to space is critical for protecting the US military's communication and reconnaissance capabilities; these assets could be vulnerable to attack where replacements are not available given the long lead time (approximately one month) to prepare a launch vehicle [Hellman, 2006]. The FALCON program along with other proposed systems which seek to develop a rapidly deployable satellite launcher are further discussed in Section 1.3.1.

Small launchers, such as NASA's Scout, have been around for many years. The proposed vehicles however have more ambitious goals than being a small launcher as the Pentagon wants a responsive and affordable vehicle. This will of course be possible using innovative technologies, which are currently unavailable. In terms of cost, it will be challenging to meet the Russians. Access to the internal US market will be much easier if the cost is affordable for universities and small aerospace companies which do not have sufficient funding. It has been reported that the main criteria for the success of the small launchers will be reliability. As an example, reliability concerns raised the launch cost of Pegasus XL from the initial attractive price of US $6 to US $20 million owing to additional quality control, testing and qualification requirements [Morring, 2003].

Based on the historical data, it seems clear that it will be difficult for the new small and micro launch vehicles to break into the world market in the near future. However, an

important observation is the US military's clear shift towards small and microsatellites along with rapid launch. This will be the main driving factor for a new class of such launch vehicle development. The need for such launchers is also stimulated by an unmet demand for low cost launchers by universities and small aerospace companies. The growing capability of small and microsatellites has become a major force in encouraging this trend.

1.3 Global Launch Vehicle Programs

An overview of the current launch vehicle market is described. Both available and proposed launch systems are considered. Some of the proposed systems detailed are at the conceptual level, whereas others are at the development stage. The first section pertains to small and micro launchers and is discussed in order of relative importance and relevance to the study herein. Then, as some other research interests are directed towards RLVs, exotic propulsion means and heavy launchers, these systems are also addressed. It should be noted that this is not an exhaustive list but gives the reader an outlook of the global launch vehicle programs and research interests.

1.3.1 Small and Micro Launcher Initiatives

In this section, global small launcher initiatives are discussed. Emphasis is directed towards the most relevant and interesting systems for the case study herein.

Responsive Access Small Cargo Affordable Launch

The Responsive Access Small Cargo Affordable Launch (RASCAL) program was a relatively quick, low cost access to space system initiated by DARPA. This launch vehicle targeted small and microsatellites, was partly reusable and had a short turnaround time [Wall (c), 2003]. It should be noted that although this project was ultimately cancelled, RASCAL details are provided to demonstrate the US' interest in such a venture.

Critical RASCAL requirements included a turnaround time of 24 hours along with independence of launch ranges given the high cost of range infrastructure. RASCAL's performance goals were to deliver a 50 kg payload into a 1250 km altitude Sun-Synchronous Orbit (SSO) and a 130 kg payload into an easier-to-reach orbit for US $750,000 per mission [Wall (c), 2003].

The launch system's main elements included a reusable Main Propulsion Vehicle (MPV) and an Expendable Rocket Vehicle (ERV). The MPV, a delta wing habited aircraft, was contracted to Scaled Composites and the propulsion system to Allied Aerospace. According to Scaled Composites, using a habited as opposed to an uninhabited aircraft would reduce associated costs. Due to the manoeuvres dictated by the operational concept discussed below, the four Pratt & Whitney F100 class turbofans selected for the MPV needed to be thrust augmented. In order to keep costs at a minimum, instead of resorting to an additional propulsion system such as a rocket, the thrust augmentation was

realized by using Mass Injection Pre-Compressor Cooling (MIPCC) technology. Here, water and liquid oxygen (LOX) were to be injected into the inlet to reduce the temperature which thereby imposed less strain on the engines (higher durability) and increased the mass flow (higher density) [Wall (c), 2003]. According to the program manager, "MIPCC lowers temperatures by hundreds of degrees at higher Mach numbers, and convinces an engine flying at Mach 3 and 100,000 ft (30 km) it is flying at Mach 1.2 and 20,000 ft (6 km)" [Warwick, 2003].

The ERV was a two-staged rocket vehicle. Since the ERV was to be deployed at an altitude beyond which atmospheric effects are mitigated, payload fairing and aerodynamic shaping of the ERV were not required. No final decision had been made as to whether the ERV would be an all-solid, all-hybrid or other combination of rocket systems. However, a hybrid first stage and solid second stage was the most probable. The mix of hybrid and solid rockets offered flexibility: if the hybrid's promising potential for a low cost system did not mature, the vehicle could resort to a solid first stage. This rocket architecture had significant safety considerations: since the first stage was to be fired in relatively close proximity to the aircraft, if an anomaly were detected with a solid rocket, the motor would be required to be destroyed and the ensuing debris could introduce challenges for the aircraft [Wall (c), 2003].

RASCAL's operational concept began with the MPV take-off. Once the MPV was at the desired launch position over the ocean, it accelerated to Mach 3.1 and climbed to an

altitude of 19 km. After loitering for up to 30 minutes before launch commit, the MPV initiated a zoom manoeuvre which consisted of a steep climb and subsequent deployment of the ERV at Mach 1.2 and 61 km in altitude. Next, the MPV descended to cruise altitude and landed [Wall (c), 2003].

Force Application and Launch from the Continental United States

A similar development program to that of this case study, the FALCON program, provides clear indication of the US' interest in a low cost small satellite access to space.

The joint DARPA, US Air Force Space Command and Air Force Research Laboratory FALCON program features a responsive and affordable small satellite launcher [Morring, 2003]. Space Exploration Technologies (SpaceX), a California start-up company that won one of the FALCON contracts [Morring, 2004], is slated to launch the Falcon 1 rocket in the first quarter of 2007 following the first attempt in March 2006 [Space Exploration Technologies (a), 2007]. The heaviest Falcon 1 capability is 668 kg to a 200 km orbital altitude and 28° inclination. Falcon 1 is a two-staged LOX and kerosene configuration with parachute water landing first stage [Isakowitz, 2004]. SpaceX is also developing two Falcon 9 versions to deliver multi metric tons to orbit [Space Exploration Technologies (a), 2007].

The cost of Falcon 1 was set to US $5.9 million in addition to range and payload specific expenses [Isakowitz, 2004]. The Falcon project was designed with reliability as the governing focus. According to a launch failure analysis between 1980 and 1999

conducted by Aerospace Corporation, 91% of known failures are attributed to engine, stage separation and avionics failures. As a result, the Falcon 1 has a minimum number of engines (i.e. one per stage) and a single stage separation event [Space Exploration Technologies (b), 2007].

Orbital Express

The Orbital Express was a joint effort by Bristol Aerospace Limited (Bristol), a Canadian company and MicroSat Launch Systems Incorporated (MicroSat), headquartered in the US, which began in 1989 to introduce an inexpensive access to space. The launch vehicle was designed to provide rapid, low cost, and dedicated small and microsatellite services to LEO. The initial payload target was 90 kg to LEO. MicroSat proposed to stack existing Bristol motors with few additions and modifications to build the orbital vehicle. However, subsequent design iterations yielded a four staged launch vehicle in which only the third stage was a Bristol product, the remaining being from the US. In 1992, the projected launch cost was estimated at US $6 million. Ultimately, given performance challenges and other factors beyond the scope of this study, this vehicle was not further developed [Hughes, 1996]. Additional details regarding this vehicle are provided in Chapter 2.

Shavit

With the Shavit launch in September 1988, Israel became the eighth nation with an access to space. Although the original three staged solid propellant Shavit was retired in 1990, Shavit 1, which is similar to Shavit with an upgraded first stage, made its maiden

flight in 1995. This vehicle is capable of delivering 350 kg to LEO. Due to Israel's geographical location, the Israeli Air Force's operated Palmachim launch facility requires retrograde orbit deliveries. The launches must be performed westward such that jettisoned stages land in the Mediterranean Sea instead of on Israel's neighbours. According to [Isakowitz, 2004], "in addition to traditional safety constraints, Israel is concerned that neighbouring, unfriendly states could learn about missile technology from any stages that might land in their territories".

Israel has attempted to market Shavit variants such as the LK-A, which has the same payload capability as that of Shavit 1, beyond its borders. In turn, Israel could increase its market and access launch sites which are better suited for efficient orbital access given the range restrictions described above. Recently, Israel has been pursuing the option of operating from Alcantara in Brazil [Isakowitz, 2004].

C-130 Aircraft Based Launcher

The Israeli academic sector, in conjunction with industry members, investigated a small satellite air launch scenario from a C-130 aircraft. This is an interesting approach to avoid the geographical constraints on Israel's launch activities discussed above along with the high maintenance costs of a launch site [Opall-Rome, 2003]. This initiative clearly indicates Israel's interest in developing a low cost small satellite access to space.

The proposed concept was to deliver a payload between 250 kg and 300 kg to an orbital altitude of 350 km. The mission consisted of releasing a two stage launcher from a C-130

aircraft at 6 km altitude. Once deployed, the launcher would open its three parachutes and ignite the first stage. This design would entail minimal development work given that the launcher utilizes existing motors and the aircraft is readily available [Opall-Rome, 2003].

F-15 Combat Aircraft Based Launcher

The US investigated the feasibility of launching a small satellite from an F-15. Interesting features of this system include its rapid deployment, access to all inclinations, and the initial kinetic and potential energy imparted to the launcher.

The US Air Force Research Laboratory examined the feasibility of launching a 100 kg payload into a 225 km orbit altitude from an F-15 combat aircraft. The thrust-augmented aircraft was to take-off and climb to 12 km altitude with Mach 1.7 and a 60° flight path angle, at which point the launcher would be released. In contrast to the RASCAL concept described above, this mission design deployed the launcher at a lower altitude and thus higher atmospheric density which introduced fairing implications. According to Air Force Lt. Julia Rothman, space systems research physicist at the Air Force Laboratory's Space Vehicles directorate, the proposed demonstration mission would cost US $5 million. Hence, this relatively low cost launcher's market could be extended to the academic space research industry [Singer, 2003].

Pegasus XL

The Pegasus XL air launched vehicle from a Lockheed L-1011 aircraft has nearly 92% successful launch rating to its credit [Orbital Sciences Corporation, 2007]. Orbital

Sciences Corporation's Pegasus XL winged booster launch costs range from US $15 to $25 million per mission [Isakowitz, 2004].

This launch system involves coordination between the carrier aircraft's pilots, onboard launch panel operators and ground control centre. Typically, spacecraft and booster system checks, including approximately 125 pre-take-off countdown steps, are conducted from the control console inside the aircraft 4.5 hours before the launch. Waypoint milestones and a launch box with precise time gates are also setup for the flight [Covault, 2003].

Similarly to an expendable booster and payload mounted vertically on a launch pad, the Pegasus XL has ground infrastructure requirements once it is coupled to the L-1011. This equipment, however, is portable which provides launch location flexibility. An important requirement for a Pegasus XL launch is a restricted or unoccupied airspace for deployment. Since the carrier aircraft may be required to cross busy airways, this may result in unexpected delays and possibly a launch cancellation which also plays a critical role in abiding to the flight plan [Covault, 2003].

Minotaur

In addition to the Pegasus XL launcher, the US also has the ground launched Minotaur vehicle which targets heavier payloads; it can deliver 607 kg to LEO for US $17 to $20 million [Isakowitz, 2004]. Following the decision to recover the multistage Minuteman II ICBMs from underground silos, a use for the 650 missile stages needed to be determined.

The US Air Force contracted Orbital Sciences Corporation to devise the Minotaur which consisted of a combination of motors from both the Minuteman II ICBMs and the Pegasus XL. The program manager at Kirkland Air Force Base, New Mexico, Major George Stoller, explained that "the whole idea behind Minotaur was to try to put together a low cost vehicle" [Iannotta, 2000].

Taurus

Orbital Sciences Corporation's ground launched Pegasus XL derivative, Taurus, was originally designed for DARPA's Standard Small Launch Vehicle (SSLV) program. DARPA was seeking a responsive and transportable access to space which could be operated from remote launch sites [Isakowitz, 2004].

Following the successful four staged SSLV Taurus maiden flight in late 1994, Orbital Sciences Corporation developed a commercial Taurus version with an upgraded first stage and first launch in 1998. This was followed by the development and launch of an even larger version, the Taurus XL, which was launched in 2004. In terms of payload capability, the SSLV, commercial Taurus and Taurus XL launchers can deliver 1310 kg, 1370 kg and 1590 kg to 200 km and 28.5° inclination respectively for US $25 to US $47 million [Isakowitz, 2004].

Rockot

In addition to the US, Russia is also interested in LEO-bound small satellites as attested by the Rockot launcher. The three stage liquid fuelled vehicle can deliver a 1000 kg

payload to an 800 km SSO. The Rockot launch vehicle system offers two dedicated launch facilities at Baikonur and Plesetsk and the flexibility of deploying single and multiple satellite configurations [Schumacher, 2003].

Vega

ESA's small launcher, Vega, is currently in development. This vehicle will be utilized for scientific and Earth observation missions with the first launch in 2007 [European Space Agency, 2007].

The Vega design philosophy is cost driven; it is based on a design-to-cost process where performance is traded with cost requirements. Three stacked main solid propellant stages are utilized for the vehicle [Crosicchio, 1996]. Vega is expected to further reduce its cost per flight by capitalizing on the existing Ariane 1 launch facilities from which Vega will be launched in French Guiana [European Space Agency, 2007].

VLS-1 and VLM

The Brazilian launch vehicle program began in the early 1980s and was slated to provide an independent access to space for small environmental payloads and develop Brazil's aerospace technology. Brazil's strong interest in a small launch vehicle is clearly demonstrated by the estimated US $250 to $300 million launcher program investment program [Isakowitz, 2004].

Brazil's VLS-1 is a four stage, all-solid propellant launch vehicle capable of delivering 380 kg into a 200 km orbit height with a 5° inclination for an estimated US $8 million. A smaller version of the VLS-1, VLM, is in development and is intended to launch microsatellites at a lower cost. VLM is a four stage all-solid propellant which is expected to be capable of lifting 100 kg into the same orbit as that of the VLS-1 for an estimated US $4 million [Isakowitz, 2004].

TOR and ARS Boosters

The literature review identified that small satellite launchers have sparked interest in booster development as attested by the Russian TOR and ARS boosters. Such boosters could be utilized for plane changes or injection into interplanetary trajectories following LEO delivery; this may be a preferred alternative rather than resorting to a larger launcher which could incur additional costs and/or scheduling challenges. The small-sized TOR booster was based on solid propellant and had a mass of 20 kg with a US $800,000 estimated development cost. In contrast, the 1000 kg liquid propelled ARS booster's estimated development cost was projected at US $5 million [Kislitsky, 2003].

SR-71 Reconnaissance Aircraft Based Launcher

The SR-71 reconnaissance aircraft, modified as the first stage of an orbital launch system using Bristol and US solid rocket motors (SRMs) has been proposed indicating interest in air launching payloads to orbit [Anderson, 1996].

The winged launch vehicle, attached to a dorsal pylon, would be launched at 21 km altitude and Mach 3.2 from the carrier aircraft. In turn, the mounting and separation sequence introduce challenges which need to be addressed. These include the implications of the launch vehicle being required to traverse the shock wave field generated by the carrier aircraft along with the potential for unwanted aerodynamic forces and moments. According to [Anderson, 1996], a 2120 kg three staged launch vehicle, configured with Black Brant, Nihka and Star 20A SRMs mounted on the modified SR-71, could deliver 75 kg into a 200 km circular orbit altitude. In the case of a larger 3000 kg launch vehicle configured with two Black Brants and Star 20A SRMs, this system could deliver 110 kg into a 250 km circular orbit altitude. Finally, a 6200 kg Orion 50XL, Orion 38H and Star 30E SRMs could deliver 350 kg into a 230 km circular orbit [Anderson, 1996].

Small-Payload High-Altitude Delivery System

The Canadian Small-Payload High Altitude Delivery System (SPHADS) development program is a low cost sounding rocket effort which could be a stepping stone towards an orbital vehicle.

The SPHADS solid rocket technology was designed for 7 kg scientific payload deliveries to the upper atmosphere. Two SPHADS variants, which primarily differed in the propellant grain configuration, were considered. Variants 1A and 1B were characterized by a specific impulse (I_{sp}) of 240 s and 215 s respectively. In the case of suborbital ground launches, for similar total propulsive energy, Variant 1B was preferred given its

higher acceleration through the transonic region translating into a heating level which was one order of magnitude below that of Variant 1A. However, in an air launched suborbital scenario, since the effects of drag are mitigated, it is best to accelerate quickly and promptly to delay the vehicle's declination rate and hence, Variant 1A was deemed to be preferable [Greatrix, 1998].

1.3.2 Exotic and Heavy Launch Programs

In this section, launch technology research programs at the conceptual phase are summarized. Two heavy launch programs are also discussed to demonstrate that the industry continues to be interested in building larger launch vehicles.

NASA's Advanced Space Transportation Program

The literature review identified RLVs using air-breathing propulsion to be an area which is benefiting from research efforts. Among others, these new space transportation systems seek to reduce cost and increase safety.

The Advanced Space Transportation Program (ASTP) at NASA's Marshall Space Flight Centre (MSFC) is focusing its efforts on a third generation launch system which is to be fully reusable and operational by 2025. ASTP's main objectives are to reduce cost and improve safety over current systems by factors of 100 and 10,000 respectively. Their approach is to raise "design and operability margins to improve hardware robustness by operating well below the hardware design limits". It has been reported that "this approach

will increase life, reduce maintenance and refurbishment requirements, and improve reliability" [Cook, 2003].

A potential solution to ASTP's objectives is hypersonic air-breathing propulsion with research being conducted in three key propulsion areas including scramjets, rocket-based combined cycle and turbine-based combination cycle. Compared to rocket propulsion, air-breathing systems deliver a higher I_{sp}, improve the design/operating margin given their higher structural mass ratio and provide expanded launch windows which increase mission flexibility. Moreover, an air-breathing vehicle can operate from existing runways as opposed to a prescribed launch pad. In turn, this can reduce infrastructure costs which are addressed in Chapter 4. Also, an air-breathing vehicle benefits from additional safety characteristics since it can abort during or soon after take-off as discussed in Chapter 4 [Cook, 2003].

Andrews Space's Reusable Launch Vehicle

In addition to the RLV efforts by government agencies, industry is also developing RLV concepts. The will to develop a reusable launcher is widespread.

Andrews Space is pursuing an RLV concept for the forecasted near-term markets likely to grow on the International Space Station (ISS). New markets are expected to emerge provided the RLV offers a service below US $1100 per kg to LEO and a launch failure probability less than one in 10,000. Andrews Space's design is a Boeing-747 sized horizontal take-off and landing two-stage-to-orbit scheme producing LOX in flight; air

would be liquefied by liquid hydrogen and separated by a centrifuge. Once the LOX tanks are filled, the vehicle would transition from turbofan to rocket propulsion and climb to 61 km altitude at which time an upper stage would deliver up to approximately 27,000 kg to the ISS [Morring, 2002].

Reusable Launch Vehicle Hopper

Under ESA's Future European Space Transportation Initiation Program, the RLV Hopper concept was adapted by the national German ASTRA program to reduce the risk and cost of space transportation beyond that of current and future expendable launch vehicles. As a result, the RLV Hopper design adopted existing technologies, predominantly from the Ariane 5 program [Spies, 2003].

The RLV Hopper is an uninhabited, fully automated, semireusable, rocket-powered system. The launch mission begins with a horizontal take-off supported by a rail-guided sled in order to promote the degree of safety and reduce thrust requirements. This enables the RLV Hopper to proceed with the launch even with the failure of one of the three main engines. The RLV Hopper climbs to 100 km altitude and ejects an expendable upper stage which delivers the payload to orbit. The RLV Hopper then autonomously re-enters the atmosphere for a horizontal landing [Spies, 2003].

Heavy Launch Vehicle

In addition to the industry's interest in developing small satellite launchers, there are also some efforts to build even heavier launch vehicles given the commercial importance of large satellite systems.

Most current large launchers such as the European Ariane 5, the Russian Proton and the US Delta IV Heavy launch systems can deliver approximately 20,000 kg to LEO. However, heavy launchers capable of delivering well beyond 20,000 kg to LEO have not yet been successfully developed. According to [Hempsell, 2003], "the reasons for such projects being abandoned are normally complex, but failure to prove a market for the large capacity despite the lower specific costs is a factor". In an attempt to validate a heavy launcher, it can be argued that it is not necessary to introduce such a vehicle given that the ISS is being successfully constructed in orbit with existing vehicles. Also, in orbit assembly techniques have undoubtedly matured to a level which may not warrant the development of a Heavy Launch Vehicle (HLV). However, although the HLV may not be necessary, [Hempsell, 2003] indicates that it "is the most cost effective approach to establishing large in orbit systems". An HLV decreases the mass, complexity and acquisition time in comparison to existing large launchers and thus yields a reduction in the total acquisition costs [Hempsell, 2003].

Potential options for an HLV capable of delivering 50,000 kg to LEO include a modified Ariane 5, a Space Shuttle variant and a new reusable vehicle. In turn, these could cater to

space stations, orbital transfer vehicles along with lunar and planetary landers [Hempsell, 2003].

Semireusable Launch Vehicle

A semireusable system using the existing Ariane 5 has been proposed by MAN Technologies AG. The study investigated the impacts of replacing the Ariane 5's solid rocket booster with a liquid fly-back booster. The 520,000 kg total lift-off mass, fully cryogenic launcher was determined to be capable of delivering up to 10,000 kg to geostationary transfer orbit and 20,000 kg to LEO. According to [Staniszewski, 2001], "this launch vehicle concept proposal could culminate in a fully reusable launch vehicle, a booster and an orbiter stage".

1.4 Summary

Since small and microsatellites have shown that it is possible to execute missions very effectively and rapidly at low cost, space launch technologies relevant to such satellites have been addressed. Despite the current oversupply of launch capacity in the world, dedicated launch vehicles for small and microsatellites are very limited. However, many countries are currently developing low cost small launch vehicles indicating a clear interest and shift towards smaller satellites and rapid launch. This is partially motivated by the high launch price of small and microsatellites and the strong demand for low cost launchers.

The Orbital Express, introduced in Section 1.3.1, is further detailed in Chapter 2; this launch system was utilized as the basis for the performance investigations of a ground and air launch. The launch system's features along with the modeling methods are described.

CHAPTER 2

THE ORBITAL EXPRESS PROJECT

The analytical segment of the ground versus air launch study was based on the Orbital Express launch vehicle introduced in Section 1.3.1. The primary criteria which led to this vehicle selection were the fact that it included a Canadian initiative along with its small and micro class spacecraft capability.

The major objectives of this chapter include further describing the Orbital Express along with the ground and air launch vehicle parameters. It should be noted that since the

vehicle was not ultimately put into service for reasons which were beyond the scope of this project, vehicle data is limited. The information used is that which was deemed to be most reliable from literature. However, as this study's objective was to compare the ground and air launch, model details are secondary, provided consistency is exercised.

2.1 Overview

The Orbital Express was a joint effort by Bristol and MicroSat. This project originated in the mid to late 1980s with a mandate to build a small, inexpensive, commercial launch vehicle to cater to the then-projected unsatisfied small LEO spacecraft market [Hughes, 1996].

A Canadian ground launched orbital launch vehicle was proposed by Bristol in 1984. Although this vehicle did not materialize, a contributing factor may have been the world launcher situation at the time; the US Space Shuttle was in its infancy and Canada could obtain an inexpensive, occasionally free, delivery of its science payloads. In 1989, MicroSat entered the marketplace with the Orbital Express proposal which consisted of combining a selection of existing Bristol and US motors along with few additions and modifications [Hughes, 1996].

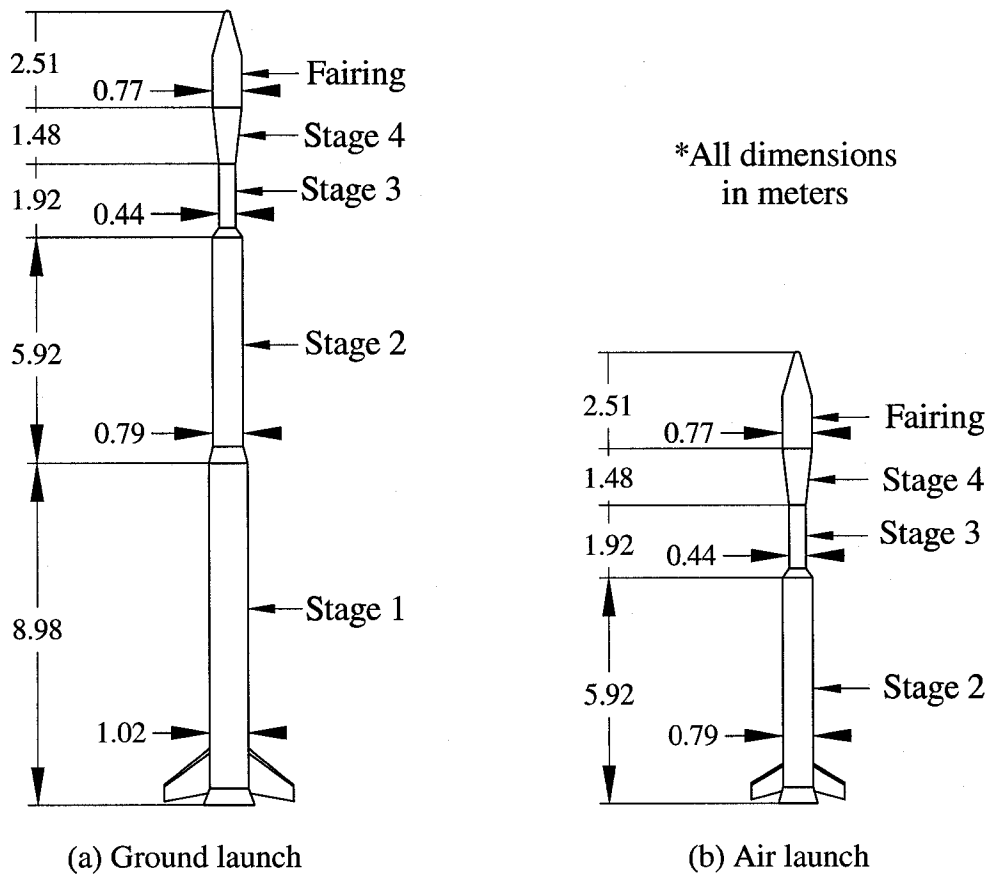
MicroSat's mission was to provide a low cost dedicated LEO launch vehicle particularly for the emerging private industries' small spacecraft launching needs. This market niche was identified by MicroSat as being unexploited and potentially lucrative. At the time,

this market was an undeclared service interest among the launch vehicle community and the short lead time was an added value. The launch cost would be partially reduced through the development methodology which relied upon proven, off-the-shelf technology. The proposed starting service cost was slated at US $1 million for an unspecified payload capability; the first mission's objective was to deliver a payload into orbit regardless of the mass [Hughes, 1996].

In 1991, International MicroSpace Incorporated (MicroSpace), formerly MicroSat, officially announced their partnership with Bristol to develop the Orbital Express with a mid 1993 maiden launch. The four staged solid propellant expendable launch vehicle would provide a dedicated launch service for payloads with a mass of up to 180 kg into LEO for US $4.5 million. Although the total launch cost, including expenses such as range fees and insurance, increased to an estimated US $6 million in 1992, this was still lower than MicroSpace's US $13 million estimate for the LTV Scout, a US four staged orbital launcher with similar capability [Hughes, 1996].

The Orbital Express was a dedicated launch vehicle for microsatellites designed for rapid, low cost LEO deliveries. As shown in Figure 2.1 (a), this four staged solid propellant rocket system was characterized by a length of 20.8 m, a fully configured mass of approximately 16,400 kg and a maximum payload mass of 180 kg. The maximum payload size was 0.77 m diameter and 1.78 m in length [Hughes, 1996] which is competitive with the Pegasus XL's accommodations of 1.12 m diameter and 2.13 m in

length [Isakowitz, 2004]. Finally, the vehicle was designed to be compatible with US operations at Poker Flats, Alaska (polar orbit), Wallops Island, Virginia (mid-inclination) and Spaceport, Florida (low inclination) [Hughes, 1996].



(a) Ground launch (b) Air launch

Figure 2.1: Orbital Express vehicle derivatives [Hughes, 1996]

Although the ground launched vehicle, in the context of the study herein, was unmodified from the original Orbital Express, the air launched version, shown in Figure 2.1 (b), consisted of the upper three stages as discussed in Section 2.3 and had an overall length of 11.8 m.

2.2 Ground Launch Scenario

Since the original Orbital Express vehicle was ground launched, there were no modifications made to the baseline vehicle. Details regarding the ASTOS and in-house models are provided in this section.

2.2.1 AeroSpace Trajectory Optimization Software

The foregoing discussion focuses on the importance of selecting an appropriate ascent profile along with the ASTOS modeling procedures and assumptions. The model definition, phase configurations, mission definition, constraints and variables are described in turn. It should be noted that the models and results were validated by a member of the ASTOS development team.

Identifying an appropriate trajectory to orbit is critical since the final payload mass and vehicle design are typically highly sensitive to the selected ascent profile [Griffin, 1991]. An ascent profile's underlying principle is to begin with a vertical trajectory to gain altitude early followed by an acceleration as the vehicle pitches down where the pitch angle (θ) is defined as the angle between the local horizon and the body axis. This pitch down manoeuvre is initiated once the atmospheric effects have sufficiently subsided such that the desired orbital speed can be achieved [Greatrix, 2005]. As a result, typical constraints for the initial and final flight path angle (γ), defined as the angle between the local horizon and the velocity vector, are 90° and 0° respectively [Griffin, 1991]. As detailed in Section 2.2.2, a compromise trajectory must be selected to minimize losses; a

vertical trajectory decreases drag losses but increases gravity losses while a trajectory characterized by an early pitch down manoeuvre yields the opposite effects [Sarigul-Klijn, 2001]. A gravity gradient trajectory was selected for the cases herein where possible. As further discussed in Section 2.2.2 and Chapter 3, some optimized trajectories did not follow such a profile throughout the flight duration in order to maximize the payload capability and satisfy path constraints. Gravity gradient trajectories to orbit are widely accepted using current multistage launch systems owing primarily to their inherent structural and aerodynamic features. Such trajectories have a near 0° angle of attack (α), defined as the angle between the velocity vector and body axis, resulting in minimum vehicle bending stress [Goodell, 1995]. Assuming that the thrust, zero lift line and body axis coincide, which is reasonable in many cases, a gravity turn trajectory yields parallel thrust and velocity vectors. In turn, thrust is entirely devoted to acceleration along the flight path while the vehicle gradually transitions from the vertical to horizontal at burnout [Ashley, 1974]. By minimizing vehicle loading, the launcher can be designed with a lower structural mass which maximizes the payload mass inserted into orbit. Although designers must ensure the launcher can withstand axial loads, a gravity gradient trajectory introduces the opportunity to put less emphasis on transverse loads which renders a vehicle sensitive to flight at high speed and angle of attack in dense atmospheric pressure [Wiesel, 1997]. Ultimately, in order to minimize mass, typical launchers are devised for strength in lengthwise compression and relatively weak in bending, shear and torsion which are generated by lifting surfaces [Curtis, 2005]. It

should be noted that the gravity turn approach was applied for both ground and air launched models due to the above rationale.

Model Definition

The model definition file is a compilation of the model components which references the environment and launch vehicle characteristics. The flight environment including the atmosphere, wind and planet are detailed. The launch vehicle parameters comprise the aerodynamic configuration, propulsion systems and vehicle components. These models are discussed in turn.

Among the available ASTOS atmosphere library, the Kapustin Yar, Russia launch site Fall atmosphere and wind were selected. Although this study's launch site was Churchill, Canada, the similar latitudes were expected to result in the best atmospheric model approximations (48.3° North Kapustin Yar and 57.7° North Churchill). The Fall models were chosen as opposed to that of the Spring which was also available in ASTOS due to the added logistical challenges of the higher probability of a snow covered launch range in the Spring. The atmosphere file specified the density, pressure and speed of sound for a given altitude and the wind file provided the eastern and northern velocity components as a function of altitude.

The Earth model was based on the J_2 term of the geopotential expansion representing the Earth's oblateness at the equator. This zonal coefficient is the dominant effect by three

orders of magnitude which rendered this approximation appropriate for this study [Wertz (a), 1999].

The aerodynamic model used herein was adapted from that of the Rockot launch vehicle. Based on the available ASTOS aerodynamics, the Rockot vehicle was deemed to be the most appropriate in terms of vehicle configuration, capability and size. The reference area of this reduced three Degree of Freedom (DOF) model was scaled to the dimensions of the Orbital Express. This approximation was considered valid and appropriate as both launchers are similarly shaped in addition to the results obtained which are further discussed in Chapter 3. A three DOF system was selected based on ASTOS test examples along with the fact that a similar study using Program to Optimize Simulated Trajectories utilized the three DOF equations of motion [Sarigul-Klijn (a), 2005]. For reference purposes, the Rockot is configured with a 2.22 m inner fairing diameter [Isakowitz, 2004] compared to 0.76 m for the Orbital Express [Hughes, 1996]. The corresponding 0.46 m^2 reference area of the Orbital Express was incorporated in the ASTOS model.

The Orbital Express was a four staged solid motor vehicle. Stages one through four were the Castor IVb, Castor I, Nihka and Star 20 respectively all of which were manufactured by Alliant Techsystems Incorporated (ATK) with the exception of the Nihka Bristol motor [Hughes, 1996]. Although the vehicle data was limited as introduced above, the motor parameters used in this study, shown in Table 2.1, are those which were deemed to be the most realistic.

Table 2.1: Orbital Express motor parameters [Hughes, 1996, *Space and Tech, 2005, $^{\Delta}$Astronautix (a), 2005 and $^{\square}$Astronautix (b), 2005]

Parameter	Unit	Stage			
		1	2	3	4
Motor		Castor IVb	Castor I	Nihka	Star 20
Manufacturer		ATK	ATK	Bristol	ATK
Control		Thrust vector control (TVC)	Cold gas reaction control system		
Thrust	kN	432	306	51	27
Stage mass	kg	11,523	3967	402	301
Propellant mass	kg	9956	3416	322	273
Inert mass	kg	1567	551	80	28
Inert mass fraction	%	14*	14$^{\Delta}$	20	9$^{\square}$
Burn time	s	61*	27$^{\Delta}$	18	28$^{\square}$
I_{sp} (vacuum)	s	267*	247$^{\Delta}$	285	287$^{\square}$
Mass flow rate	kg/s	165	127	18	10
Length	m	8.98	5.92	1.92	1.48
Diameter	m	1.02*	0.79$^{\Delta}$	0.44	0.50$^{\square}$

It should be noted that the pressure thrust was neglected largely due to the lack of data availability along with the fact that this effect would take place for a relatively short duration. In turn, this typically promotes an optimistic approach but was deemed appropriate given the above factors. In an air launch case, this effect is minimized as the motors operate at higher altitudes.

Since limited fairing data was identified, this value was approximated based on selected world launchers with LEO capability of up to 2000 kg and available data such as the Minotaur, Pegasus XL and Rockot launch vehicles. The payload to fairing mass ratios of

those vehicles, where reliable data was available, are tabulated in Table 2.2 in alphabetical order. The average payload to fairing ratio was subsequently applied to approximate the fairing mass of the Orbital Express.

Table 2.2: Payload to fairing ratios for existing launch vehicles [Isakowitz, 2004]

Launch Vehicle	Payload	Fairing Mass	Payload/Fairing
	kg	kg	
Angara 1.1	2000	710	2.82
Athena I	820	535	1.53
Kosmos	1500	348	4.31
M-V	1900	700	2.71
Minotaur	607	194	3.13
Pegasus XL	443	170	2.61
Rockot	1950	800	2.44
Taurus XL	1590	360	4.42
VLM	100	109	0.92
VLS-1	380	109	3.49
Payload/Fairing Average			**2.83**

Although the Orbital Express was expected to be capable of delivering up to 180 kg into LEO, the vehicle also had some challenges in reaching this goal as discussed in Section 1.3.1 [Hughes, 1996]. As a result, the fairing mass was determined iteratively subject to the payload capability. It should be noted that the majority of LEO capabilities tabulated in Table 2.2 are to LEO inclinations equal to the launch site latitude which translates into the highest LEO capability. The Orbital Express ground launched vehicle was optimized to the launch latitude's inclination to identify the maximum payload capability. Based on the simulations, the highest circular orbit payload capability was identified at 725 km

altitude and yielded a 55.69 kg payload with a 22 kg fairing mass. This translated into a 2.53 payload/fairing mass ratio which was 10.5% more conservative than that of the average obtained from the world launchers shown in Table 2.2. The selection of this value was further validated by the fact that a similar study to that herein utilized a 28 kg fairing mass [Greatrix, 2005]. In order to promote commonality and ultimately reduce costs, the 22 kg fairing mass was applied to all simulations and optimizations.

Phase Configurations

The ASTOS phase configurations records describe the scheduling information of each flight phase. Parameters such as the aerodynamic configuration, active propulsion, jettisoned components and attitude controls either optimized or evaluated by a control law are specified. Two phase configuration records exist, one for initial guess purposes and the other for optimization.

The initializing phase configuration record's initial state consisted of the position and velocity records. The position record included the Churchill launch latitude, longitude and altitude and the velocity record specified the zero initial speed and vertical initial inclination and heading. The initializing phase configuration record comprised eight phases: Lift Off, Pitch Over, Pitch Constant, Stage 1, Stage 2, Stage 3, Coast Arc and Stage 4 each of which are detailed below. It should be noted that these phases along with their respective parameters were largely based on the ASTOS test examples library, model feasibility of the optimizations and consultations with a member of the ASTOS development team. All phases utilized the same aerodynamic configuration described

above and were all reduced Euler angle attitude controls based on the factors previously enumerated. Although some exceptions to the mean final times shown in Table 2.3 were occasionally made to improve the optimization results, those quoted in Table 2.3 were typically utilized and were based on the respective motor burn times shown in Table 2.1.

Table 2.3: Ground launch initializing phase configuration record sample

Phase	Final Time (s)			Active Propulsion	Jettisoned Components
	Low	Mean	High		
Lift Off	0.1	5.0	14.9	Stage 1	N/A
Pitch Over	5.1	15.0	29.9	Stage 1	N/A
Pitch Constant	15.1	30.0	60.4	Stage 1	N/A
Stage 1	59.5	60.5	61.5	Stage 1	Stage 1
Stage 2	86.5	87.5	88.5	Stage 2	Stage 2, Fairing
Stage 3	104.5	105.5	106.5	Stage 3	Stage 3
Coast Arc	110.0	200.0	5000.0	N/A	N/A
Stage 4	27.0	28.0	29.0	Stage 4	Stage 4

(a) Final time, active propulsion and jettisoned components

Phase	Attitude Controls	
	Yaw	Pitch
Lift Off	Constant law	Vertical take-off
Pitch Over	Constant law	Linear law
Pitch Constant	Constant law	Constant law
Stage 1	Constant law	Gravity turn
Stage 2	Target orbit inclination	Gravity turn
Stage 3	Target orbit inclination	Gravity turn
Coast Arc	Linear law	Linear law
Stage 4	Target orbit inclination	Required velocity

(b) Attitude controls

The Lift Off phase ended at a nominal five seconds into the flight. The first stage motor was active with no jettisoned components. The constant law, for which the control is constant during the phase [Institute of Flight Mechanics and Control (a), 2005], was applied to the yaw control with a value of 90° minus the target orbit inclination and the vertical take-off was applied to the pitch control. The Pitch Over phase ended at 15 s with the first stage motor still active; this phase had no jettisoned components. Yaw control was under the constant law while the pitch control used the linear law, where the control is a linear function of time during the phase [Institute of Flight Mechanics and Control (a), 2005], with a final value subject to the targeted altitude and inclination but was typically 85° for optimum results. The Pitch Constant phase was a constant pitch until the gravity turn condition detailed below was met; this phase ended at a nominal 30 s with the first stage motor still active and no jettisoned components. Both the yaw and pitch were controlled using the constant law. The Stage 1 phase ended at a nominal 60.5 s after which the first stage tank was jettisoned. It should be noted that Stage 1 along with the other motor stages were characterized by bounds although their burn time was a pre-determined value. However, the bounds were introduced given the improved optimization behaviour; the final optimized results indicated that the mean value was typically obtained within 1E-6 s. The yaw was again controlled using the constant law while the gravity turn was applied to the pitch control. The second stage was ignited and jettisoned during the Stage 2 phase which ended at a nominal 87.5 s; the fairing was also discarded at the end of the Stage 2 phase. The yaw control utilized the target inclination while the gravity turn continued to be applied to the pitch control. The third stage motor burned

during the Stage 3 phase and was jettisoned at the end of this phase at a nominal 105.5 s. Both the yaw and pitch controls were identical to those detailed for the Stage 2 phase. The Coast Arc phase had no active propulsion system and no jettisoned components. Although the nominal phase time for the Coast Arc was subject to the targeted altitude and inclination, 200 s was typical. Both the yaw and pitch controls used the linear law. During the upper stage, the fourth stage motor was burnt for a nominal duration of 28 s in contrast to the cumulative time for all other phases in order to mitigate challenges with the Coast Arc duration. As jettisoned components are only recognized by ASTOS if there is another phase which follows, the fourth stage motor tank was not discarded but subtracted from the resulting payload. This was in fact the method reflected in the ASTOS test examples. The yaw control used the target inclination while the required velocity including the target orbit perigee and apogee were applied to the pitch control.

The phase configuration record was used for optimization purposes. The initial state of this record was identical to that described for the initialization phase configuration. All phases were the same as those detailed for the initialization phase configuration except for the fact that the yaw and pitch controls were optimized during the Stage 2, Stage 3 and Stage 4 phases. As previously indicated, the controls for Stages 2 through 4 were optimized based on ASTOS test examples, model feasibility and communication with a member of the ASTOS development team.

Mission Definition

The mission definition record describes the mission objectives. Optimization cost functions such as maximum payload, minimum loss or minimum fuel consumption are available. The cases herein maximized the payload to evaluate the peak performances.

Mission Constraints

Three primary mission constraint types are offered by ASTOS: initial boundary, final boundary and path constraints. Initial boundary constraints define the initial position and velocity of the vehicle. With regards to final boundary constraints, in the case of expendable launchers, these would likely be the desired orbit. Path constraints such as heat flux, load factor and dynamic pressure can be enforced in order to accommodate vehicle design limitations and practical mission aspects [Institute of Flight Mechanics and Control (b), 2005].

The constraints applied in the model herein were largely based on ASTOS test cases and model feasibility. The initial, path and final boundary constraints are discussed in turn. The initial boundary constraints applied to the model were divided in the position and velocity constraints. The enforced position constraints included the Churchill launch altitude, latitude and longitude. The velocity constraints consisted of the flight path velocity, North velocity and relative East velocity all of which were set to zero and enforced. The dynamic pressure path constraint was enforced during the Pitch Constant and Stage 1 phases while the heat flux was enforced during the phases following fairing

separation: Stage 3, Coast Arc and Stage 4 phases. The details of the dynamic pressure and heat flux path constraints are detailed below. The 0° gravity turn final boundary constraint was enforced in the Pitch Constant phase such that it could be applied in the subsequent phase as required by ASTOS. The desired orbit perigee, apogee and inclination were all enforced and categorized as final boundary constraints. It should be noted that although a circular orbit was ultimately desired for the simulations herein, it was optimal to begin optimizing with a slightly elliptical orbit to obtain convergence. Once optimized, the orbit apogee and perigee were disabled while the equatorial altitude, circular altitude and radial velocity constraints were enforced. The equatorial altitude and circular altitude were set to the desired values while the radial velocity component was set to zero.

Variables

The ASTOS variables record defines parameters which are referenced throughout the model such as the Churchill launch range coordinates. Although this launch range is currently inactive, it catered to numerous Black Brant suborbital vehicle launches. This launch site is a convenient location for Northern orbits due to its high latitude. In fact, this launch location was deemed to be the most appropriate for future Canadian orbital launches. According to many experts including members from the Canadian space industry and government, the Churchill launch range would be an adequate selection for a ground launch campaign. Although it currently does not have the capability to support a launch campaign, it could be re-opened for launches from its current heritage site status. Many experts agree that it is technically feasible to re-open the Churchill launch range

provided funding is available for refurbishing and maintenance to meet established operating standards. Many of the Churchill launch range facilities still exist including heated buildings, service tunnels and a power supply. However, since the range is under the responsibility of the government of Manitoba, the fate of Churchill launch range is within their hands [Labib, 2004]. It should be noted that this launch site was also selected for a similar ground and air launch study [Greatrix, 2005]. Based on the gathered Churchill launch range coordinates data, the latitude was set to 57.7° North [Webber, 2004]. As the Churchill launch range longitude and altitude were not identified in technical publications, the parameters were chosen to be those at the Churchill airport; the longitude was set to 94.6° West and the altitude to 29 m above sea level [Climate Zone, 2005].

As previously indicated, the target orbit perigee, apogee and inclination were all defined in this record and subject to the desired orbit. Since the maximum dynamic pressure and aeroheating rate limit at fairing separation for the Orbital Express were not identified in literature; these values were set to 92 kPa and 1135 W/m^2 respectively based on existing launchers and ASTOS test examples. The maximum dynamic pressure and aeroheating limit of selected world launchers with LEO capability of up to 2000 kg and available data along with their respective fairing size are shown in Table 2.4. Also, the equivalent maximum dynamic pressure force and heat generation on the Orbital Express based on the fairing's frontal area were evaluated.

Table 2.4: Dynamic pressure and aeroheating limits of selected launch vehicles [Isakowitz, 2004]

Launch Vehicle	Fairing Diameter	Maximum Dynamic Pressure		Aeroheating Limit	
		Pressure	Equivalent Force	Heat Flux	Equivalent Heat
	m	kPa	kN	W/m^2	W
Angara 1.1	2.5	30.9	151.7	1135	5571
Athena I	2.3	140.0	581.7	1135	4716
Falcon I	1.5	33.5	59.2	1135	2006
Kosmos	2.4	42.8	193.6	600	2714
M-V	2.5	142.0	697.0	N/A	N/A
Minotaur	1.3	60.0	76.0	1135	1438
Pegasus XL	1.3	48.0	60.8	4542	5754
Rockot	2.5	66.4	325.9	1135	5571
Strela	2.5	65.8	323.0	1135	5571
Taurus XL	1.6	N/A	N/A	1135	2282
VLS-1	1.2	70.0	79.2	N/A	N/A
VLM	1.2	90.0	101.8	N/A	N/A
Average	**1.9**	**71.8**	**240.9**	**1454**	**3958**
Corresponding Orbital Express Parameters					
Orbital Express	0.8	528.2	240.9	8679	3958

The corresponding Orbital Express' maximum dynamic pressure force and heat generation are shown in Table 2.4 based on the average values and the fairing's frontal area. Given the average maximum dynamic pressure, the corresponding figure for the Orbital Express along with the ASTOS optimizations using various maximum dynamic pressure values, the lowest feasible value was 92 kPa which was ultimately selected. This maximum dynamic pressure was further validated by the fact that a similar project to that

herein assumed a 91 kPa maximum dynamic pressure. The [Greatrix, 2005] study investigated a ground and air launch of a 6.8 kg payload to 320 km altitude utilizing three SRMs. The ground launched version was nearly 2.5 times heavier than the air launch with a mass of 4500 kg and 1900 kg respectively and 0.05 m^2 frontal area [Greatrix, 2005]. These were similar to the Orbital Express based investigation where the ground and air launched versions had a mass of approximately 16,300 kg and 4700 kg respectively with 0.46 m^2 frontal area. The [Greatrix, 2005] analysis is further discussed in Chapter 3 at which point comparisons to the results herein are described. Finally, an 1135 W/m^2 aeroheating limit was selected based on the values of world launchers and to promote a conservative model.

2.2.2 MATLAB Model

The foregoing discussion details the methods utilized in creating an in-house ground launched trajectory simulation model using MATLAB to validate the ASTOS results and to gain insight into the ASTOS package given its high degree of complexity. The velocity budget is investigated and is followed by a description of the equations of motion. Finally, the simulation approach using the equations of motion is defined.

Velocity Budget Investigation

The required change in velocity (ΔV), provided by the launch vehicle to deliver a payload into orbit, is computed as follows [Sarigul-Klijn (a), 2005]:

$$\Delta V_{Ideal} = V_{Orbit} + \Delta V_{Drag} + \Delta V_{Gravity} + \Delta V_{Steering} + \Delta V_{Atmospheric\ Pressure} - V_{Earth\ Rotation} \tag{2.1}$$

Where:

V	Velocity

The V_{Orbit} term is subject the payload's desired orbital altitude and is given by [Wertz (a), 1999]:

$$V_{Orbit} = \left[\mu_E \left(\frac{2}{R_E + h} - \frac{1}{a} \right) \right]^{1/2} \tag{2.2}$$

Where:

μ_E	Geocentric gravitational constant of the Earth
R_E	Radius of the Earth
h	Altitude
a	Semimajor axis

The drag, gravity, steering and atmospheric pressure change in velocity terms are driven by the launcher's ascent profile. Drag loss reflects the frictional effects between the launch vehicle and the atmosphere. In the case of Delta and Atlas sized ground launched systems, drag loss ranges from 40 m/s to 160 m/s [Sarigul-Klijn (a), 2005] and can be computed from ignition (t_0) to burnout (t_b) as follows [Sarigul-Klijn, 2001]:

$$\Delta V_{Drag} = \int_{t_0}^{t_b} \frac{D}{m} dt \tag{2.3}$$

Where:

D	Drag force
m	Mass

Drag losses can be minimized with a vertical trajectory to rapidly clear the region of highest atmospheric density along with a favourable vehicle shape. Also, according to Equation 2.3, increasing the launcher size reduces drag losses given that drag is a function of surface area as opposed to volume [Sarigul-Klijn, 2001].

Gravity loss results from the fact that energy is expended to hold the launcher against gravitational effects. This loss is a function of the thrust to weight ratio and is typically between 1150 m/s and 1600 m/s for ground launched systems [Sarigul-Klijn (a), 2005]. The gravity loss from ignition to burnout is given by [Sarigul-Klijn, 2001]:

$$\Delta V_{Gravity} = \int_{t_0}^{t_b} g \sin(\gamma)\, dt \tag{2.4}$$

Where:

g Gravitational acceleration of the Earth

Equation 2.4 indicates that flying a trajectory which promptly minimizes the flight path angle decreases gravity losses [Sarigul-Klijn, 2001].

Steering losses are introduced when the instantaneous thrust (T) and velocity vectors are not parallel. However, this difference is necessary to steer the launcher [Sarigul-Klijn, 2004]. TVC methods include engine gimballing, nozzle injection, jet vanes and differential throttling when multiple engines are present [Griffin, 1991]. In addition, aerodynamic control surfaces can be utilized which reduce steering losses at the expense

of drag losses and inert mass. Typical steering losses are approximately 33 m/s for a Delta sized ground launched system and are given by [Sarigul-Klijn, 2004]:

$$\Delta V_{Steering} = \int_{t_0}^{t_b} \frac{T}{m}\left(1 - \cos(\alpha)\right) dt \tag{2.5}$$

The atmospheric pressure term in Equation 2.1 is attributed to the performance characteristics of rockets operating in vacuum compared to those in the atmosphere. Rocket thrust is based on both the momentum change of propellant and the nozzle exit pressure change whereby ambient pressure reduces the thrust [Sarigul-Klijn (a), 2005]. The atmospheric pressure loss is given by [Sarigul-Klijn, 2004]:

$$\Delta V_{Atmospheric\ Pressure} = \int_{t_0}^{t_b} \frac{pA_e}{m} dt \tag{2.6}$$

Where:

p	Ambient air pressure
A_e	Exit nozzle area

As discussed in Section 2.2.1 and per Equation 2.6, an air launch minimizes atmospheric pressure losses given the lower ambient pressure compared to that of a ground launch [Sarigul-Klijn, 2004].

The Earth's rotation term in Equation 2.1 is a function of the launch latitude (φ) and azimuth (β) which is the angle measured clockwise from North to the trajectory plane and given by [Sarigul-Klijn, 2004]:

$$V_{Earth\ Rotation} = (463\ \text{m/s})\cos(\varphi)\sin(\beta) \tag{2.7}$$

It should be noted that Equation 2.1 neglected the change in velocity caused by Coriolis acceleration, which is the effect of the Earth's rotation [Saraf, 2004], since its magnitude is less than 10 m/s [Sarigul-Klijn (a), 2005]. Wind effects were also ignored in Equation 2.1, the impact of which is addressed below.

Since the trajectory losses are integrated in time, reducing the atmospheric flight duration can decrease losses. Two such methods include increasing the launcher's acceleration and air launching. Higher accelerations can be realized with an increased thrust to weight ratio which in turn, reduces gravity and steering losses but increases drag losses. It is worth noting that the optimum thrust to weight ratio increases with altitude due to the lower corresponding drag losses [Sarigul-Klijn, 2004]. The air launching implications are discussed in Section 2.3.4.

As introduced in Section 2.2.1 and exemplified by the above loss equations, a compromised trajectory must be selected to minimize losses. On the one hand, a steep climb can minimize drag losses and improve propulsion efficiency given the lower atmospheric density but increases gravity losses. On the other hand, a trajectory characterized by an early pitch down manoeuvre followed by a climb to orbit minimizes gravity and steering losses at the expense of higher drag losses and lower propulsion efficiency. Since gravity loss produces a component normal to the flight path, trajectories with an initial non-vertical ascent will undergo a gradual turn toward the horizontal; however, control means can be utilized to bring the flight path angle to near horizontal

following an initial vertical ascent. As alluded to in Section 2.2.1, a gravity turn approach is most efficient only from an airless planet since such a trajectory incurs a longer flight time in the lower atmosphere. In an Earth to orbit case, the low steering losses of a gravity turn throughout the ascent would be overcome by factors such as higher drag losses. Hence, gravity turns typically comprise phases of an ascent to orbit for both ground and air launches as further described in Chapter 3 [Griffin, 1991].

Equations of Motion

The forces applied on a launch vehicle during a typical powered ascent are shown in Figure 2.2 [Curtis, 2005].

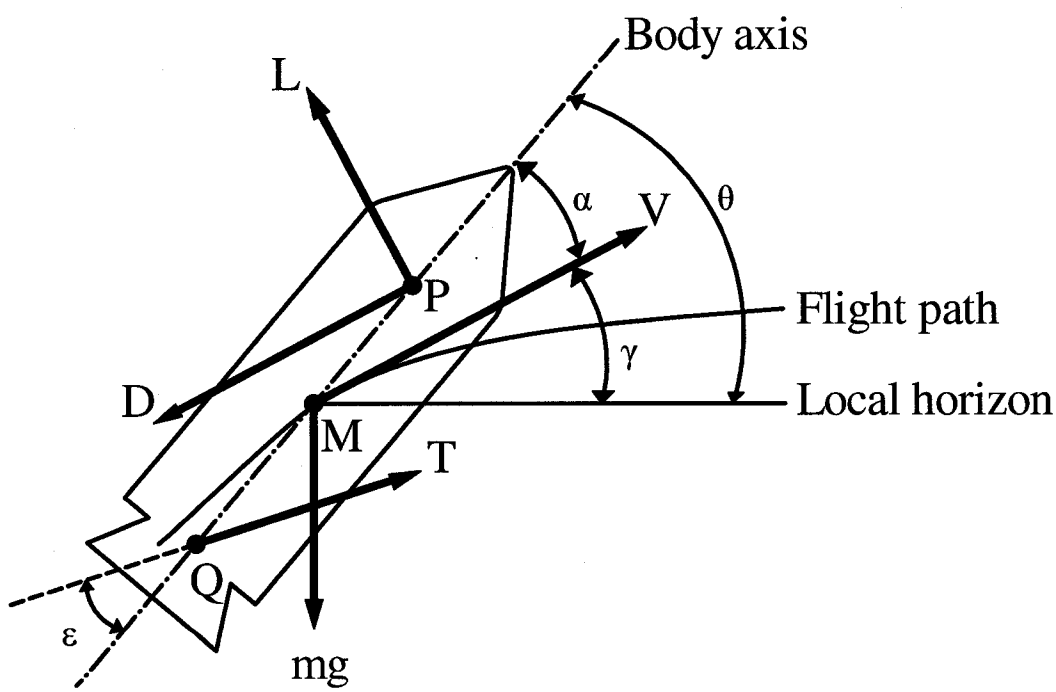


Figure 2.2: Forces acting on a launch vehicle during powered ascent

Where:

L	Lift force
ε	Thrust angle
P	Centre of pressure
M	Centre of mass
Q	Thrust application point

The lift vector is shown orthogonal to the velocity vector and the drag vector acts opposite to the velocity vector. Both the lift and drag forces are applied at P; although the vehicle configuration in Figure 2.2 is shown as being unstable, the purpose of the diagram was to depict the most general case and to clearly identify the forces. The lift and drag forces can be computed as follows [Vinh, 1980]:

$$L = qSC_L \tag{2.8}$$

$$D = qSC_D \tag{2.9}$$

Where:

$q = \frac{1}{2}\rho V^2$	Dynamic pressure
ρ	Atmospheric density
S	Reference area
C_L	Lift coefficient
C_D	Drag coefficient

It should be noted that the velocity term in the dynamic pressure computation is that which is relative to the atmosphere [Griffin, 1991]. However, this was neglected in this

study since its effect on the dynamic pressure is relatively small as described in Chapter 3 along with the fact that other sources in literature made the same assumption [Curtis, 2005, Wertz (a), 1999 and Vinh, 1980]. Both lift and drag coefficients are functions of the angle of attack, Mach number and Reynolds number [Vinh, 1980]. The thrust vector in Figure 2.2 is shown to be applied at Q with a thrust angle and the gravitational force is applied at M and is directed towards the Earth's centre. Finally, the angle of attack is measured between the velocity vector and body axis which assumes that the latter and zero lift line coincide. This was deemed reasonable since the angle between the zero lift line and body axis is relatively small, if not zero, for a wingless booster such as the case herein. In addition, this was sensible given the data availability as well as the results accuracy obtained and described in Chapter 3; other sources in literature also made the same assumption [Curtis, 2005, Wertz (a), 1999 and Vinh, 1980].

Assuming that the atmosphere is at rest with respect to the Earth, which is often justified due to the relatively small effects, the former has the same rotation as the latter. The equations of motion thus include an angular velocity (ω) to account for the Earth's rotation. Two terms involving the angular velocity appear in the governing equations: $\omega^2 r$ where r represents the position and $2\omega V$. Since the angular velocity is typically small, neglecting the first term ($\omega^2 r$) was justified. Although the second term ($2\omega V$), known as the Coriolis acceleration, has a notable influence in the case of a high-speed, long range flight, this term was ignored since the study herein was primarily concerned with the variations of speed and altitude. In turn, this entailed a non-rotating Earth assumption.

Moreover, since the powered phase is relatively short, assuming that the Earth was an inertial reference system where the atmosphere is at rest was reasonable for first-order approximations to satisfy the objectives indicated above. The equations of motion also presumed that the trajectory was in the plane containing both the launch and burnout positions which required that all the forces be within this plane. As a result, the vehicle was assumed to be characterized by a plane of symmetry within which the velocity vector, aerodynamic and thrust forces were acting. As in the case with the ASTOS models, points M, P and Q in Figure 2.2 were considered to be aligned to constitute a body axis fixed with respect to the vehicle [Vinh, 1980] which is typically appropriate for preliminary design purposes [Griffin, 1991]. All of the above assumptions were deemed reasonable due to the factors previously enumerated, the fact that other sources in literature made the same assumptions along with the results accuracy described in Chapter 3 [Curtis, 2005 and Wertz (a), 1999].

Recalling that the instantaneous velocity vector determines the current flight path, the unit vectors $\hat{u}_t$ and $\hat{u}_n$, tangent and normal to the flight path respectively were defined. With the unit vector normal to the flight path directed towards the centre of curvature C, the distance from the flight path to the point C, known as the radius of curvature (κ), was also defined. Resolving Newton's second law into components along the flight path's tangent and normal directions yields [Curtis, 2005]:

$$a_t = \frac{dV}{dt} \tag{2.10}$$

$$a_n = \frac{V^2}{\kappa} \tag{2.11}$$

Where:

a_t Acceleration component tangent to the flight path

a_n Acceleration component normal to the flight path

Noting that $V/\kappa = -d\gamma/dt$ enables one to express the normal acceleration as follows [Curtis, 2005]:

$$a_n = -V\frac{d\gamma}{dt} \tag{2.12}$$

In order to account for the Earth's curvature, Earth-centred Cartesian coordinates can be utilized to show that a term must be added to Equation 2.12 as developed in [Vinh, 1980] yielding [Curtis, 2005]:

$$a_n = -V\frac{d\gamma}{dt} + \frac{V^2}{R_E + h}\cos(\gamma) \tag{2.13}$$

The foregoing discussion assumes a zero thrust angle. The assumption that the thrust vector is aligned with the body axis was also reflected in the ASTOS optimizations. According to an ASTOS development team member, methods such as TVC are required to move the thrust out of the body frame, in which case moments need to be addressed. Although such computations could be useful for optimizations during higher project phases, these are typically only considered in guidance activities. Despite the fact that TVC can help regulate the angle of attack in an effort to minimize steering losses, such transient variations are typically small, between 2° to 5° in the case of gimballing for a relatively short duration which justifies the zero thrust angle assumption [Griffin, 1991].

Also, this was further justified given the data availability, the results accuracy obtained and described in Chapter 3 along with the fact that other sources in literature made the same assumption [Curtis, 2005 and Wertz (a), 1999]. Finally, expressing Newton's second law into components along the flight path's tangent and normal directions yields [Vinh, 1980]:

$$ma_t = T\cos(\alpha) - D - mg\sin(\gamma) \tag{2.14}$$

$$ma_n = -T\sin(\alpha) - L + mg\cos(\gamma) \tag{2.15}$$

Substituting Equations 2.10 and 2.13 into Equations 2.14 and 2.15 respectively yields [Curtis, 2005]:

$$\frac{dV}{dt} = \frac{T}{m}\cos(\alpha) - \frac{\rho V^2 S C_D}{2m} - g\sin(\gamma) \tag{2.16}$$

$$V\frac{d\gamma}{dt} = \frac{T}{m}\sin(\alpha) + \frac{\rho V^2 S C_L}{2m} - \left(g - \frac{V^2}{R_E + h}\right)\cos(\gamma) \tag{2.17}$$

Adding the downrange distance (x) and altitude yields [Curtis, 2005]:

$$\frac{dx}{dt} = \frac{R_E}{R_E + h} V\cos(\gamma) \tag{2.18}$$

$$\frac{dh}{dt} = V\sin(\gamma) \tag{2.19}$$

Simulation Approach

The equations of motion, Equations 2.16 to 2.19, must be solved numerically to account for the variation of thrust, mass, atmospheric density, lift and drag coefficients along with the gravitational acceleration [Curtis, 2005].

The simulation model was segmented in the same phases as those discussed Section 2.2.1. The same payload fairing mass was utilized and the final optimized payload mass as computed by ASTOS was inputted to the MATLAB model such that the initial vehicle mass could be evaluated. Also, the optimized flight path angle, pitch angle, lift and drag coefficients outputs of ASTOS were inputted to the MATLAB code in order to meet the objectives enumerated above. It should be noted that since the thrust vector depends directly on the pitch angle, the difference in pitch and flight path angle was utilized when resolving the thrust vector into components as opposed to the angle of attack. With regards to the motor data, the parameters discussed in Section 2.2.1 were utilized in the simulation. The density was that of the ASTOS model for the Kapustin Yar launch site which addressed the issue of the density variation with altitude. The gravitational acceleration, while neglecting the Earth's geographical features and oblate shape which was considered reasonable given the factors listed above, was computed as a function of altitude as follows [Sutton, 2001]:

$$g = g_0 \left[\frac{R_0}{R_0 + h} \right]^2 \tag{2.20}$$

Where:

$R_0 = 6378,388\,\text{m}$ Radius of the Earth at the equator

$g_0 = 9.80665\,\text{m/s}^2$ Gravitational acceleration of the Earth at sea level

The equations of motion were integrated at the same time steps as those for ASTOS and the altitude, vehicle mass, velocity and dynamic pressure were computed at each time step. Wind effects were not considered but were deemed to introduce minor discrepancies

as discussed in Chapter 3. Although a typical trajectory is characterized by a gradual increase in thrust and I_{sp} with altitude, the simulation herein assumed vacuum values since the effect was only for a relatively short duration of the trajectory [Ashley, 1974]. Discontinuities such as engine burn durations and stage jettisoning were addressed in the MATLAB simulations by invoking a new phase with updated initial conditions as depicted in Section 2.2.1 [Rainey, 2004].

2.3 Air Launch Scenario

The air launched Orbital Express vehicle derivative, for the purpose of this study, consisted of removing the first stage motor. This relatively simple modification in comparison to, for instance, resorting to a larger second stage such as a hybrid engine, was deemed to yield minimal cost and complexity in order to promote reliability. Details regarding the vehicle derivative and models are described in this section.

2.3.1 Methodologies

This report provided parameters and specifications of the launcher's initial conditions. Potential air launch methods and aircraft platforms were investigated from which recommendations are brought forward; however, a detailed analysis of the appropriate method and carrier aircraft was beyond the scope of this study. Selected air launch methods are assessed in Table 2.5.

Table 2.5: Assessment of air launch methods [Sarigul-Klijn, 2001 and *Sarigul-Klijn (b), 2005]

Air Launch Categories	Advantages	Disadvantages
Captive on Top	• Large launch vehicle carrying capability	• Extensive modifications to carrier aircraft • Launch vehicle must have large wings to support separation
Captive on Bottom	• Proven separation sequence (i.e. Pegasus XL)	• Launch vehicle size restrictions subject to carrier aircraft • Modifications to carrier aircraft
Towed	• Relatively simple separation sequence • Limited modifications to towing aircraft	• Broken towlines and take-off abort safety concerns • Limited access to launch vehicle prior to launch
Aerial Refuelled	• Reduced carrier aircraft's wings and landing gear size	• Size of jet engines is not reduced; must be sized for a fully fuelled carrier aircraft
Internally Carried	• Limited modifications to carrier aircraft • Access to launch vehicle prior to launch	• Flight crews exposed to risks of carrying fuelled rockets • Launch vehicle size restrictions subject to carrier aircraft
***Balloon Launch**	• Limited operational costs	• Requires operation of a very large balloon • Limited to favourable atmospheric conditions

Regarding the towing option, a Canadian Patent details a towed methodology where a launch vehicle, configured as a glider, is towed by an aircraft to the desired launch altitude [Canadian Intellectual Property Office, 2004]. Another issue worth noting is a supersonic separation approach. Although this would provide higher initial kinetic energy to the launch vehicle, the inherent supersonic flow fields introduce significant challenges to the separation sequence as concluded by the SR-71 D-21 drone accident investigation

board [Merlin, 2002]. In short, the benefit of a supersonic carrier aircraft is outweighed by the added expenses [Bonometti, 2006]. An important variation of the air launch sequence where the launch vehicle is mounted under the carrier aircraft, i.e. captive on bottom, is the trajectory of the launch vehicle following separation. Specifically, the launch vehicle can either cross the carrier aircraft's altitude ahead or aft of the carrier aircraft. A typical air launch is one where the launch vehicle is mounted under a carrier aircraft, dropped, and allowed to cross the carrier aircraft's altitude ahead of it. Examples of such a methodology include the X-15, Pegasus XL and SpaceShipOne. The Trapeze-Lanyard Air Drop (t/LAD) is similar to the examples above but crosses the carrier aircraft's altitude aft of the carrier aircraft. The t/LAD method is expected to improve simplicity, safety, cost and reliability of LEO launch systems. Other potential benefits include the elimination of launcher wings or fins, reduction in sideways accelerations and bending forces along with lower TVC. The t/LAD sequence, currently being tested, is initiated by dropping a launch vehicle mounted under a carrier aircraft. A lanyard, which rotates the launcher about its centre of gravity, is attached to the carrier aircraft via a trapeze used for stabilization purposes of the launch system. As the lanyard reels out, a pitch up force is applied on the launcher as shown in Figure 2.3 [Sarigul-Klijn, 2006].

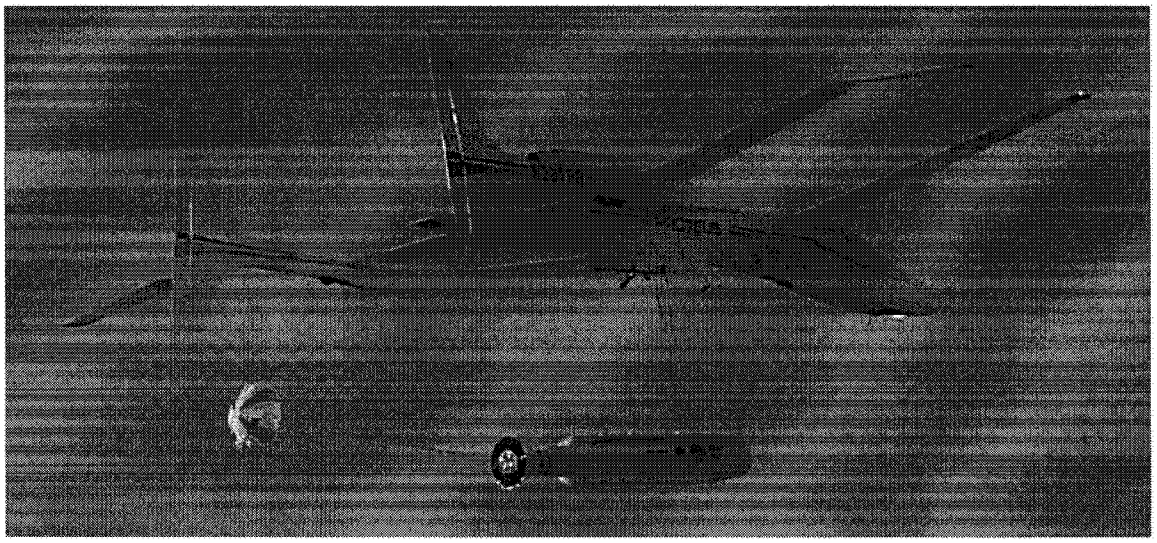


Figure 2.3: t/LAD method [Sarigul-Klijn, 2006]

Once the desired pitch rate is achieved, the lanyard pulls free from the carrier aircraft. While descending and rotating, a drogue parachute attached to the nozzle improves the pitch angle and rate and provides yaw stability. The drogue parachute then decreases the pitch rate to zero at which time the launcher has a 90° pitch angle; the desired orientation for the aft crossing trajectory. At motor ignition, the drogue parachute's risers are burnt and the launch vehicle continues its descent for six seconds as the rocket's thrust compensates for the negative velocity. Once the launcher crosses the aircraft's altitude, the launch vehicle follows a gravity turn type trajectory to orbit. The air launch sequence is shown in Figure 2.4 with images taken at 0.533 s intervals [Sarigul-Klijn, 2006].

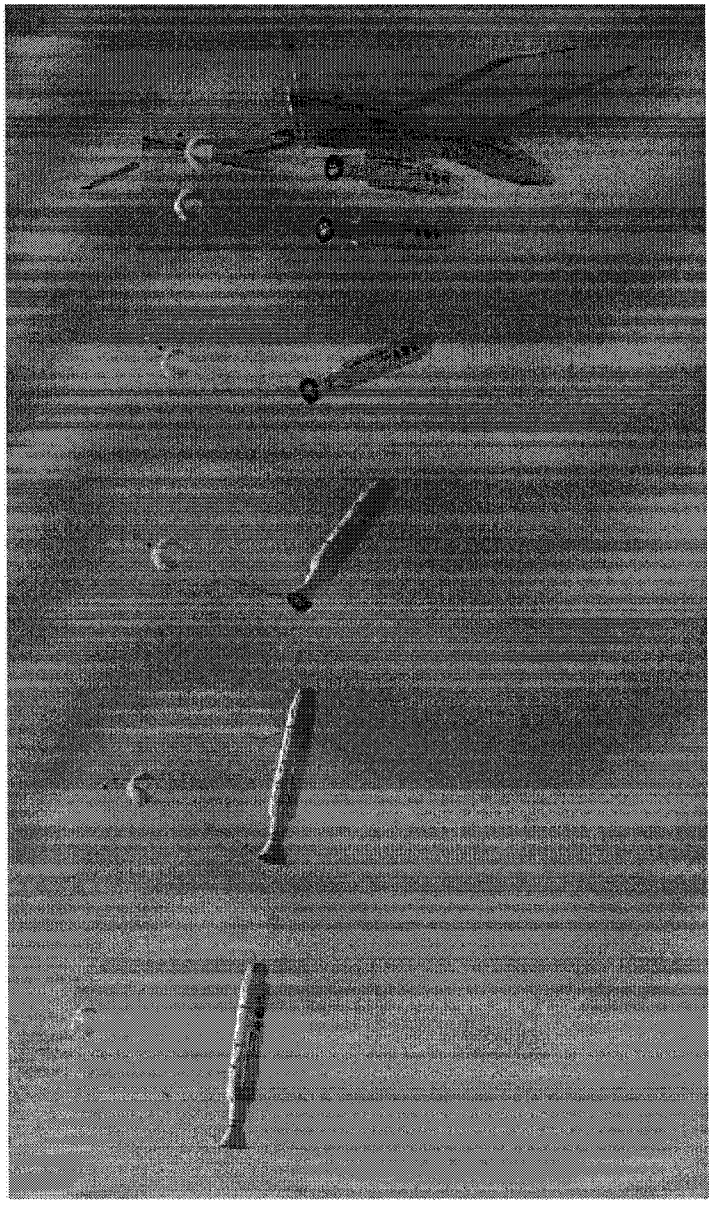


Figure 2.4: t/LAD sequence at 0.533 s intervals [Sarigul-Klijn, 2006]

As introduced above, the advantages of an aft versus a forward crossing trajectory include the elimination of wings or fins, such as those on the X-15, Pegasus XL and SpaceShipOne, and other systems, namely control surface actuators, auxiliary power units along with wing and fin thermal protection. Aft crossing also minimizes, or even eliminates, collision scenarios between the launch vehicle or its debris with the carrier

aircraft provided the latter executes a clearing turn following the separation. Another advantage is the lower sideways accelerations during the separation sequence: the X-15, Pegasus XL and SpaceShipOne underwent strong longitudinal bending (2g to 3g pull-up) while transitioning from horizontal to vertical flight. The [Sarigul-Klijn, 2006] study reported that t/LAD, compared to forward crossing, reduces the peak first stage engine TVC from 6° to less than 1.5°. Aft crossing also reduces the maximum dynamic pressure and eliminates high angle of attack flight while under high dynamic pressure in comparison to forward crossing wingless vehicles. In turn, a higher payload mass can be realized: the lower first stage TVC reduces steering losses and the lower angle of attack and dynamic pressure yields a smaller structural mass. For instance, a lower peak dynamic pressure contributed to reducing SpaceShipOne's mass [Sarigul-Klijn, 2006]. It should be noted that many of these ideas are at the conceptual and speculative phase as no full flight demonstration has been performed to date.

2.3.2 Recommendations

Based on the assessment in Section 2.3.1, this study recommends that the launch vehicle be mounted under the carrier aircraft given the flight heritage of such practice with an aft crossing trajectory to benefit from the numerous factors enumerated in Section 2.3.1. Details regarding the proposed carrier aircraft and alternative methods to achieve the desired initial conditions are addressed in this section.

Carrier Aircraft

The two main aircraft selection criteria were as follows: a Canadian product which is commercially available. The first criterion was to promote a Canadian system, albeit three of the four motors were US built. The second criterion was to broaden the availability of the launcher to numerous customers, potentially worldwide. It is worth noting that an air launch from a CF-18, a Canadian military supersonic aircraft, configured with a three staged solid motor launch vehicle was deemed to be capable of delivering 12.8 kg. The major advantages of such a carrier aircraft include the initial launch vehicle attitude and the initial velocity imparted. Difficulties encountered with this option consisted primarily of the limited payload and clearance capabilities of the CF-18 [Labib, 2003]. However, due to the above detailed specifications, this avenue was not further investigated in this study.

The Pegasus XL launch vehicle has a mass of 23,000 kg [Isakowitz, 2004] coupled to the Lockheed TriStar L-1011 carrier aircraft which is characterized by a 34,427 kg payload capability and total take-off thrust of 561 kN [Taylor, 1977]. These yield a payload and thrust to launch vehicle mass ratios of 1.5 and 24.4 respectively. The additional aircraft payload capability was surmised to account for the onboard computer stations for the launch in addition to the launch vehicle couplings to the aircraft and the thrust for conducting the separation sequence. These ratios were subsequently applied to the case herein. The initial investigation of Canadian aircraft yielded the Bombardier CRJ series to be the most appropriate aircraft based on take-off thrust, maximum payload, maximum

range, high cruise speed and ceiling as shown in Table 2.6. In particular, as the air launched version of the Orbital Express detailed below was 4727 kg, the resulting aircraft payload and total take-off thrust requirements based on that of the Pegasus XL were 7075 kg and 115.3 kN respectively.

Table 2.6: CRJ Series aircraft parameters [Bombardier, 2006]

CRJ Series		Total Take-Off Thrust	Maximum Payload	Maximum Range	High Cruise Speed	Ceiling
		kN	kg	km	Mach	km
CRJ700	701	112.8	8527	2655	0.825	12.5
	701ER	112.8	8527	3209	0.825	12.5
	701LR	112.8	9070	3708	0.825	12.5
CRJ705	705	116.8	10,387	3184	0.830	12.5
	705ER	116.8	10,387	3635	0.830	12.5
	705LR	116.8	10,591	3702	0.830	12.5
CRJ900	900	116.8	10,319	2956	0.830	12.5
	900ER	116.8	10,319	2950	0.830	12.5
	900LR	116.8	10,591	3385	0.830	12.5

All of the CRJ Series aircraft tabulated in Table 2.6, except for the CRJ700, exceed the total take-off thrust and payload requirements. These CRJ models are characterized by a ceiling of 12.5 km which is comparable to the 12.8 km ceiling of the L-1011 [Taylor, 1977]. The Pegasus XL separates at an altitude of 11.9 km from the L-1011 [Isakowitz, 2004]. Based on the fact that the CRJ Series aircraft have a ceiling of 12.5 km, setting the initial Orbital Express altitude at 11 km was deemed reasonable [Bombardier, 2006]. This was considered conservative given that a similar study to that herein assumed a 12.2 km launch altitude [Greatrix, 2005]. Another study indicated that a 10 km altitude release is

the minimum to generate substantial benefit to an air launch [Bonometti, 2006]. In addition, according to a ground and air launch performance study of the Minotaur launcher by [Sarigul-Klijn (a), 2005], only marginal change in velocity improvement was noted above 15,240 m in altitude. First order investigations suggested that all these CRJ705 and CRJ900 Series aircraft would be suitable for the Orbital Express air launched derivative. The CRJ705 Series, shown in Figure 2.5, is recommended but performance and clearance constraints, particularly in the length between the nose and main landing may warrant resorting to a larger aircraft, such as the CRJ900 Series or potentially the CRJ1000 and CSeries.

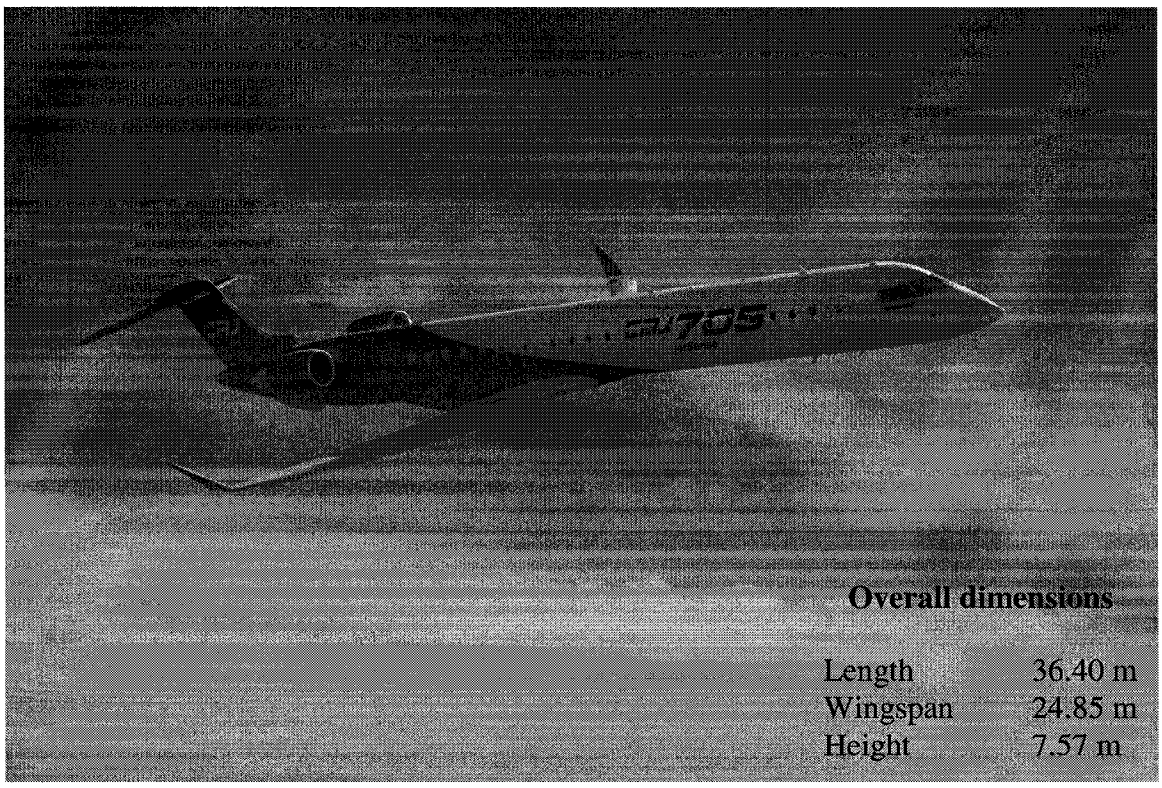


Figure 2.5: CRJ705 Series aircraft [Bombardier, 2005]

Initial Conditions

The initial attitude conditions of the launcher are important parameters as they significantly affect the vehicle's performance as depicted in Chapter 3. This study assumed an initial pitch angle of 90° in order to conduct an aft crossing trajectory as addressed in Section 2.3.1. The foregoing discussion introduces a few concepts beyond t/LAD which could be further investigated to generate the desired initial launch angle; however, as noted in Section 2.3.1, a detailed analysis of the appropriate method was beyond the scope of this project.

Immediately prior to the release of the launch vehicle, the carrier aircraft could perform a pull-up manoeuvre which would provide the launcher with a potentially high (close to 90°) initial pitch angle. This manoeuvre is subject to the carrier aircraft's capability. In such a case, the CF-18's high manoeuvrability capability is a major advantage. Another option investigated by Carleton University's Spacecraft Design Team is to deploy one or multiple parachutes to attain the vertical initial pitch angle [Carleton University Spacecraft Design Project, 2004].

2.3.3 AeroSpace Trajectory Optimization Software

The foregoing discussion focuses on the modeling procedures and assumptions in ASTOS. The model definition, phase configurations, mission definition, constraints and variables are described in turn. Similarly to the ground launched case, the models and results were validated by a member of the ASTOS development team.

Model Definition

The air launched flight environment model was identical to that of the ground launched case depicted in Section 2.2.1. Additionally, the aerodynamic configuration remained unchanged. The propulsion systems and vehicle component modifications reflected the first stage motor (Castor IVb) removal. The same fairing mass determined for the ground launch was maintained in the air launched derivative to maximize commonality and promote low cost and reliability.

Phase Configurations

The ASTOS phase configurations records introduced in Section 2.2.1 are described for the air launch case.

The initializing phase configuration record's initial state comprised the position and velocity records. The position record included the Churchill launch latitude, longitude and altitude and the velocity record specified the initial speed which was set to -50 m/s corresponding to approximately five seconds of free fall in vacuum for a conservative approach. This time delay is identical to that of the Pegasus XL launch system [Isakowitz, 2004]. The velocity record also specified the vertical initial inclination and heading. The initializing phase configuration record consisted of seven phases: Lift Off, Pitch Over, Pitch Constant, Stage 2, Stage 3, Coast Arc and Stage 4 each of which are detailed below. These phases, along with their respective parameters were largely based on ASTOS test examples, model feasibility of the optimizations and consultations with a

member of the ASTOS development team. All phases utilized the same aerodynamic configuration described in Section 2.2.1 and were all reduced Euler angle attitude controls given the above factors. Finally, although some exceptions to the mean final times shown in Table 2.7 were occasionally made to improve the optimization results, those quoted in Table 2.7 were typically utilized and were based on the respective motor burn times shown in Table 2.1.

Table 2.7: Air launch initializing phase configuration record sample

Phase	Time (s)			Active Propulsion	Jettisoned Components
	Low	Mean	High		
Lift Off	0.1	0.5	0.9	Stage 2	N/A
Pitch Over	0.6	1.5	3.9	Stage 2	N/A
Pitch Constant	1.6	5.0	26.9	Stage 2	N/A
Stage 2	26.0	27.0	28.0	Stage 2	Stage 2
Stage 3	44.0	45.0	46.0	Stage 3	Stage 3
Coast Arc	46.1	200.0	5000.0	N/A	Fairing
Stage 4	27.0	28.0	29.0	Stage 4	Stage 4

(a) Final time, active propulsion and jettisoned components

Phase	Attitude Controls	
	Yaw	Pitch
Lift Off	Constant law	Vertical take-off
Pitch Over	Constant law	Linear law
Pitch Constant	Constant law	Constant law
Stage 2	Constant law	Gravity turn
Stage 3	Target orbit inclination	Gravity turn
Coast Arc	Linear law	Linear law
Stage 4	Target orbit inclination	Required velocity

(b) Attitude controls

In comparison to the variables set for the ground launched case in Table 2.3, the air launch differs in the fact that the first stage motor has been removed in addition to the fairing jettisoning after the Coast Arc phase such that the heat flux path constraint, addressed below, could be satisfied. Also, selected phase times were modified to accommodate the air launched trajectory. Otherwise, the discussion in Section 2.2.1 applies to the case herein.

The phase configuration record initial state was identical to that described for the initialization phase configuration. All phases were the same as those for the initialization phase configuration except for the fact that the yaw and pitch controls were optimized during the Stage 3 and Stage 4 phases for reasons depicted in Section 2.2.1.

Mission Definition

Since the mission objective remained the same in the air launched case, i.e. maximized payload, the mission definition discussed for the ground launch was unchanged.

Mission Constraints

The air launch mission constraints are described and were largely based on ASTOS test cases and model feasibility.

The initial, path and final boundary constraints are detailed in turn. Similarly to the ground launched version, the initial boundary constraints included the Churchill launch altitude, latitude and longitude with the altitude set to the local altitude in addition to 11

km for reasons discussed in Section 2.3.2. Also, the radial velocity was set to the desired initial velocity, -50 m/s as noted above, along with North and relative East velocity which were set to zero and enforced. Although this inherently attributes the air launch with the initial velocity provided by the launch latitude, this was incorporated for comparative purposes and due to the variability of this parameter subject to the launch location. It should be noted that the lack of the Earth's velocity would be beneficial for air versus ground launched SSO missions. The dynamic pressure path constraint was applied during the Pitch Constant and Stage 2 phases while the heat flux was incorporated during Stage 4; the phase following fairing separation. The details of the dynamic pressure and heat flux path constraints are provided below. The 0° gravity turn final boundary constraint was enforced in the Pitch Constant phase. The desired orbit perigee, apogee and inclination were all applied and categorized as final boundary constraints. Using the method outlined in Section 2.2.1, the equatorial altitude, circular altitude and radial velocity constraints were also imposed. The equatorial altitude and circular altitude were set to the desired values while the radial velocity component was set to zero.

Variables

The only difference to the ground launch variables file was the launch altitude modification. Although the launch latitude and longitude could in theory, be any value, the Churchill launch range coordinates were selected for comparative purposes. In addition, as discussed in Chapter 4, the air launched scenario requires some ground equipment located at a base, which in practice, constrains the launch latitude and longitude. Moreover, Churchill was identified as a potential air launch base amongst

Canadian industry and government individuals; other candidate sites included Canadian Forces Base (CFB) Cold Lake, Alberta and CFB Bagotville, Québec [Labib, 2004].

2.3.4 MATLAB Model

Building on the discussion and objectives depicted in Section 2.2.2, this section focuses on the velocity budget, equations of motion and simulation approach with regards to the air launch case.

Velocity Budget Investigation

Similarly to the ground launched case, an air launched vehicle delivers a payload in orbit by providing the required change in velocity which is computed as follows [Sarigul-Klijn (a), 2005]:

$$\Delta V_{Ideal} = V_{Orbit} + \Delta V_{Drag} + \Delta V_{Gravity} + \Delta V_{Steering} + \Delta V_{Atmospheric\ Pressure} - V_{Earth\ Rotation} - V_{Carrier\ Aircraft} \tag{2.21}$$

The only exception to the ground launched case was the addition of the carrier aircraft velocity term; the factors described in Section 2.2.2 apply to the air launched case. Although the carrier aircraft term is expected to reduce the total required change in velocity, in some cases it may not, as depicted in Section 2.3.2: it is highly dependent on the carrier aircraft's capabilities and release conditions, particularly the thrust to weight ratio and flight path angle at launch [Sarigul-Klijn, 2004].

As indicated in Section 2.2.2, since the trajectory losses are integrated in time, an air launch incurs smaller losses owing to its shorter time to orbit compared to that of ground

launched systems; this is verified in Chapter 3. In fact, irrespective of the time to orbit, an air launch bypasses the region of highest atmospheric density which yields lower losses. Not only is the time of flight reduced due to the launch altitude but such results can occasionally be realized when the carrier aircraft imparts an initial velocity to the launch vehicle [Sarigul-Klijn, 2004].

Equations of Motion

The derivation of the equations of motion was identical to that described in Section 2.2.2.

Simulation Approach

The modifications to the approach detailed Section 2.2.2 are discussed; otherwise the MATLAB code was identical.

The motor data from Section 2.2.1 incorporated in the air launched MATLAB code reflected the removal of the first stage. It should be noted that the effects of utilizing vacuum motor values in the air launch were mitigated compared to those for a ground launch given the lower atmospheric density. Also, the initial conditions discussed in Section 2.3.3 were introduced.

2.4 Summary

The Orbital Express, presented in Section 1.3.1, has been further detailed; this launch system was utilized as the basis for the performance investigations of a ground and air launch. The primary factors which led to this vehicle selection included its Canadian

content along with its small and micro class spacecraft capability. The Orbital Express project, which originated in the mid to late 1980s, consisted of combining four existing SRMs to offer a dedicated launch service to the then-projected unsatisfied small and micro LEO spacecraft market [Hughes, 1996].

The ground and air launched parameters of the Orbital Express were addressed. No modifications were made to the baseline vehicle given that the original Orbital Express was ground launched. The air launched system derivative consisted of removing the first stage motor of the original vehicle to promote low cost and reliability with minimal increase in complexity. The ground and air launched vehicles had a mass of 16,270 kg and 4727 kg respectively and were assumed to be launched from Churchill. Air launch methodologies such as captive on top, captive on bottom and towed were assessed. Recommendations were then developed for the launch sequence and carrier aircraft. The study proposes that the launch vehicle be mounted under the carrier aircraft given the flight heritage of such practice with an aft crossing trajectory to benefit from factors such as the elimination of wings or fins along with the reduction of collision scenarios [Sarigul-Klijn, 2006]. The first order carrier aircraft investigation yielded the CRJ705 Series but performance and clearance constraints may warrant a larger aircraft such as the CRJ900, CRJ1000 or CSeries. Based on the recommended separation sequence, the air launched vehicle was assumed to be characterized by an initial pitch angle of 90°, an 11 km altitude and a -50 m/s velocity at launch.

Details regarding the ASTOS and MATLAB ground and air launched models were also provided. The results of Orbital Express ASTOS and MATLAB models described in this chapter are discussed and assessed in Chapter 3. The final results were largely based on the ASTOS optimizations and the MATLAB models were utilized to give insight into the ASTOS commercial software.

CHAPTER 3

ANALYTICAL INVESTIGATION

The analytical investigation of a ground versus air launch is described in this chapter. The results were largely generated from ASTOS. The models for both ground and air launches were validated by utilizing in-house numerical models. The objectives of this chapter include gaining insight into the ASTOS commercial software, identifying the underlying trajectory similarities and differences between a ground and air launch along with determining scenarios in which either a ground or air launch is preferred.

3.1 Ground Launch Scenario

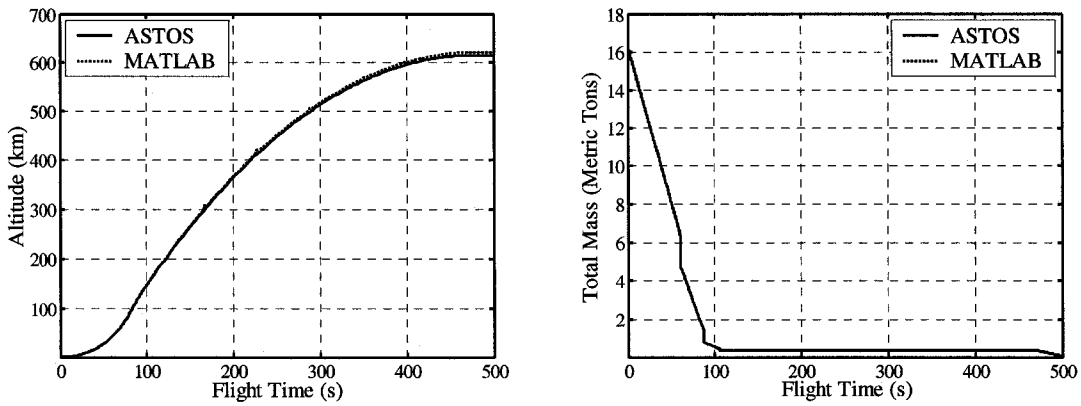
This section depicts the validation of the ASTOS ground launch model using two methods followed by a trajectory analysis. A more detailed analysis is provided in Section 3.3 at which point the ground and air launch parameters including the payload capabilities are investigated. The purpose of this section is to address the similarities and differences when varying the altitude and inclination of a ground launched system.

3.1.1 Validation

ASTOS has been validated, within a few percent, against several ESA launch vehicles and has been applied to confirm European industrial partners' in-house software where agreement was also achieved [Well, 1997]. The numerical MATLAB model created herein, introduced in Chapter 2, was utilized to conduct a comparison with the ASTOS results. Also, the change in velocity of the vehicle model in this study was evaluated and contrasted to the change in velocity of existing launchers.

MATLAB Simulations

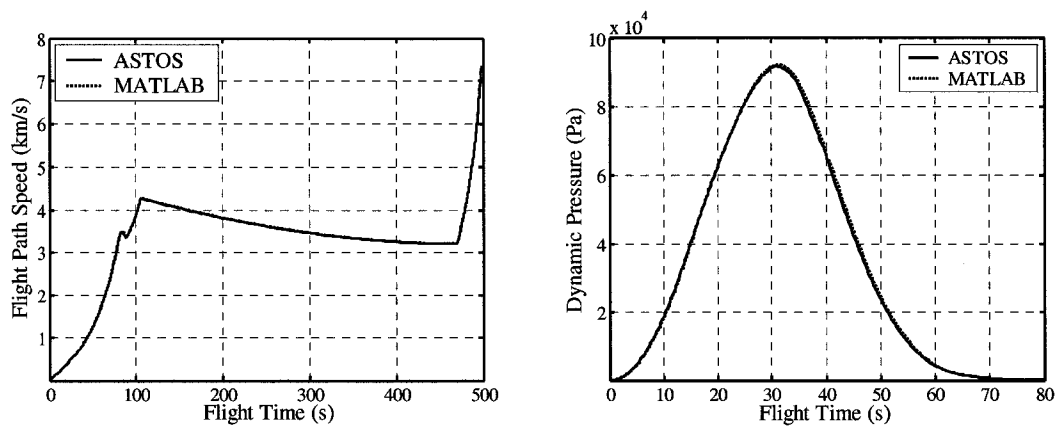
Selected MATLAB results are described in order to gain insight into the ASTOS package and validate the results. The altitude and total mass profiles are shown in Figure 3.1 and were characterized by 1.26% and 0.24% differences respectively with the ASTOS results. As these results had a relatively low percent difference and since [Sarigul-Klijn (a), 2005] and [Di Sotto, 2002] considered a 5.6% and below 10% relative difference to be good accuracy respectively, the results were deemed reasonable.



(a) Altitude (b) Total mass

Figure 3.1: Ground launched ASTOS and MATLAB altitude and total mass profiles to 600 km circular altitude and 60° inclination

The flight path speed and dynamic pressure profiles are shown in Figure 3.2; the results were within 1.25% and 4.59% respectively with that of ASTOS.



(a) Flight path speed (b) Dynamic pressure

Figure 3.2: Ground launched ASTOS and MATLAB flight path speed and dynamic pressure profiles to 600 km circular altitude and 60° inclination

Although both were considered reasonable based on the above discussion, the dynamic pressure profile was characterized by a slightly higher difference for reasons detailed in Section 2.2.2.

Velocity Budget Comparison

The change in velocity of selected world launchers with a LEO capability of up to 2000 kg and available data was computed and shown in Table 3.1. A 200 km circular altitude reference orbit was selected since this value is typically directly quoted to an inclination relatively close to the launch latitude which yields the maximum payload. The corresponding change in velocity of the ground launched Orbital Express was also computed.

Table 3.1: Total change in velocity of selected world launchers and of the ground launched Orbital Express [Isakowitz, 2004]

Launch Vehicle	Parameter				
	Payload	**Altitude**	**Launch Latitude**	**Orbit Inclination**	**ΔV**
	kg	**km**	**Degrees**	**Degrees**	**m/s**
Angara 1.1	2000	200	62.9 North	63.0	9620
Kosmos	1500	250	48.6 North	51.6	9612
M-V	1900	200	31.2 North	31.0	9978
Minotaur	607	185	28.5 North	28.5	9049
Rockot	1950	200	62.8 North	63.0	9024
Taurus XL	1590	200	28.5 North	28.5	9183
VLM	100	200	2.3 South	5.0	9755
Average ΔV					**9460**
Orbital Express	46	200	57.7 North	57.7	11,501

The data in Table 3.1 reveals that the ground launched Orbital Express' change in velocity is 21.6% above that of the average of world launchers and 15.3% above the highest computed change in velocity (M-V). Based on the fact that the Orbital Express' data was limited, confidence was yielded in the results since the values were within the

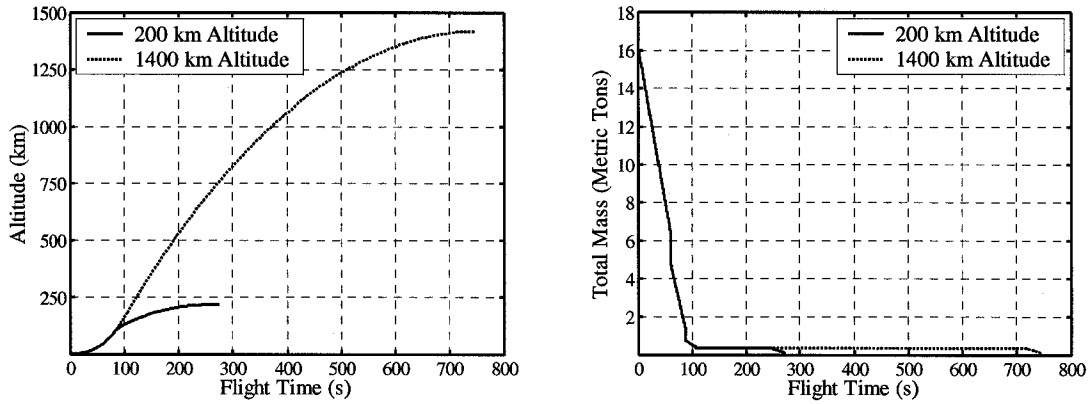
same order of magnitude. In fact, the Orbital Express data emphasized a conservative approach. It should be noted that the payload is highly sensitive to the dynamic pressure limit which is further detailed in Section 3.3. In cases where the dynamic pressure limit increased, the payload would also increase and reduce the change in velocity discrepancies.

3.1.2 Trajectory Analysis

A trajectory analysis of the ground launched system is provided. The discussion focuses on the trends for multiple optimizations at constant inclination and constant altitude respectively along with an assessment. It should be noted that the foregoing results were generated using ASTOS.

Constant Inclination

The 60° and 70° inclination ground launch performance to 200 km and 1400 km circular altitude, the feasible extremes, are described. Two inclinations were investigated to determine the trajectory trends with inclination variations. These altitudes were selected for comparison as these were the two feasible extremes for this model subject to the constraints and parameters detailed in Section 2.2.1. In addition, this approach enabled the investigation to focus on the underlying similarities and differences in the trajectories. The 200 km and 1400 km circular altitude launch systems were found to be capable of delivering 45.4 kg and 48.5 kg into orbit respectively to 60° inclination. The 60° inclination altitude and total mass profiles are shown in Figure 3.3.

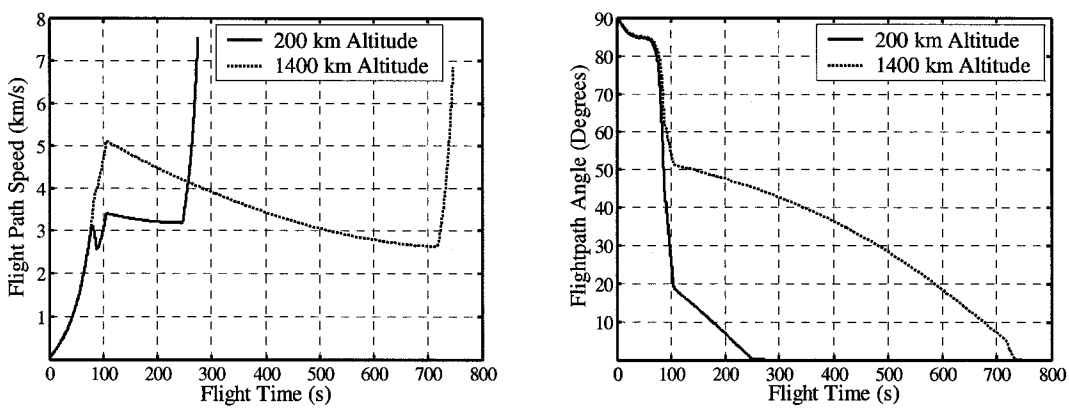


(a) Altitude (b) Total mass

Figure 3.3: Ground launched altitude and total mass profiles to 200 km and 1400 km circular altitude and 60° inclination

The altitude profiles in Figure 3.3 (a) revealed that the trajectory duration to 1400 km circular altitude was significantly longer (172%) than that of the 200 km circular altitude case. This was partly attributed to the higher altitude which in turn required a longer kinetic and potential energy exchange. It should be noted that the final altitude is not necessarily the circular altitude given that the latter is related to the equatorial radius. In the case of a spherical Earth, the altitude and circular altitude would be identical but with the ellipsoidal model such the case herein, the altitude varies with the corresponding latitude; this is carried throughout the optimizations. As expected, the total mass profiles shown in Figure 3.3 (b) were based on the staging parameters described in Section 2.2.1. Stages one through three were successively expended, followed by a Coast Arc phase at 106 s which was characterized by constant mass. Finally, the fourth stage inert mass was dropped at the end of the trajectory with the final payload mass remaining. The total mass values differed only to reflect the different payload capabilities.

Figure 3.4 addresses the flight path speed and angle profiles. As anticipated, the Figure 3.4 (a) profiles end at their respective flight path speed corresponding to the orbital altitude, a final orbit constraint. Both trajectories exhibited a segment in which the flight path speed was gradually decreased during the Coast Arc phase where the kinetic energy was exchanged for potential energy. The higher altitude case resulted in a longer Coast Arc phase given the longer energy exchange as introduced above. It should be noted that the reduction in flight path velocity at approximately 85 s in the 200 km circular altitude case was attributed to the strong negative pitch angle during this segment as further described below.

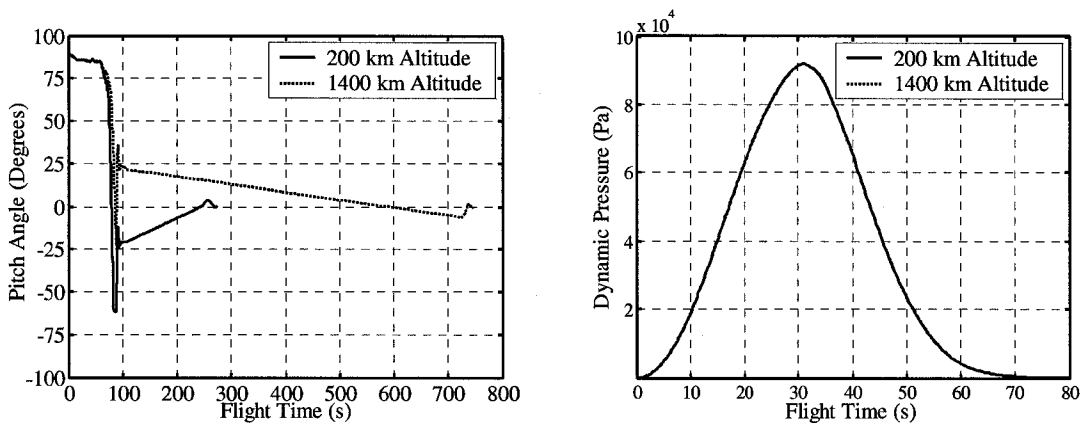


(a) Flight path speed (b) Flight path angle

Figure 3.4: Ground launched flight path speed and angle profiles to 200 km and 1400 km circular altitude and 60° inclination

With regards to the flight path angle profiles in Figure 3.4 (b), the trajectories began vertically, i.e. 90°, and decreased to 0° once the orbit had been reached. It is clear from the results that the higher altitude case was characterized by a steeper trajectory which is addressed below.

The pitch angle and dynamic pressure profiles are shown in Figure 3.5. The results indicated that both 200 km and 1400 km circular altitude cases achieved negative pitch angles within the 100 s flight time marker as shown in Figure 3.5 (a). However, the negative pitch angle was more pronounced in the 200 km circular altitude case compared to that of the 1400 km circular altitude. Although negative pitch angles are normal and optimal, as confirmed with an ASTOS developer, the strong negative pitch angles, particularly in the 200 km circular altitude case may be of concern. This was partly attributed to stacking existing motors as described in Chapter 2 and is further addressed in Section 3.3 where mitigation methods are also discussed.



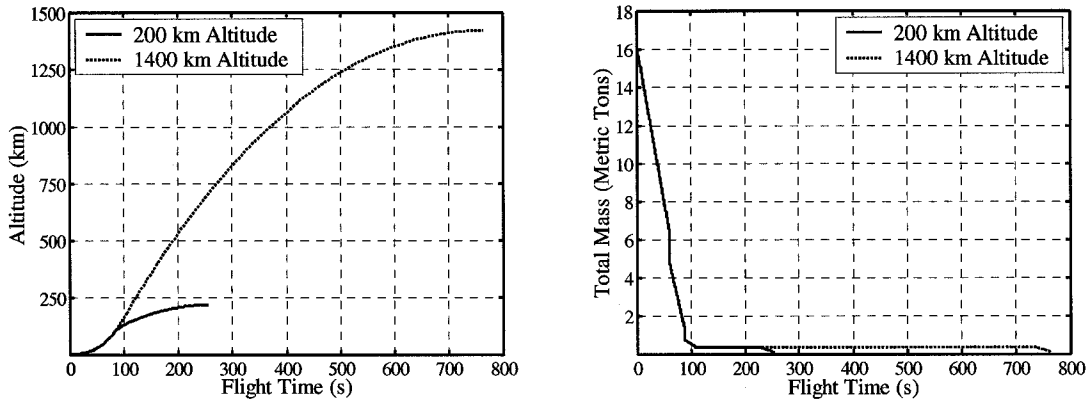
(a) Pitch angle (b) Dynamic pressure

Figure 3.5: Ground launched pitch angle and dynamic pressure profiles to 200 km and 1400 km circular altitude and 60° inclination

During the motor burns, although propulsive energy is transformed into potential and kinetic energy, a portion of the propellant's energy is dissipated as heat by aerodynamic drag [Vinh, 1980]. The dynamic pressure profiles illustrated in Figure 3.5 (b) were similar in both cases. The relatively high peak dynamic pressures were rationalized by the fact that smaller vehicles are more affected by drag as noted in Section 2.2.2. In turn,

smaller vehicles follow steeper ascent profiles to orbit which require a higher total change in velocity. However, steep trajectories are conducive to high accelerations given the shorter flight time in the lower atmosphere where the atmospheric density is relatively high. Since smaller SRMs are characterized by a higher thrust to weight ratio, such as those herein, these vehicles reach their drag peaks earlier and lower in comparison to their larger counterparts [Whitehead, 2005]. This argument further explains the underlying reasons why the case herein exhibits relatively high dynamic pressures given the small SRMs utilized throughout. Although not indicated in Figure 3.5 (b) due to the resolution, during the maximum dynamic pressure segment, the 1400 km circular altitude trajectory exhibited a consistently lower dynamic pressure albeit the difference is 0.04%.

In order to determine if the above trends held at higher inclinations, the 70° inclination trajectory parameters are assessed. The 200 km and 1400 km circular altitude systems were found to be capable of inserting 43.5 kg and 46.9 kg into orbit respectively. As expected, an increase in inclination resulted in lower payload capability due to the required plane change. The altitude and total mass profiles are shown in Figure 3.6.

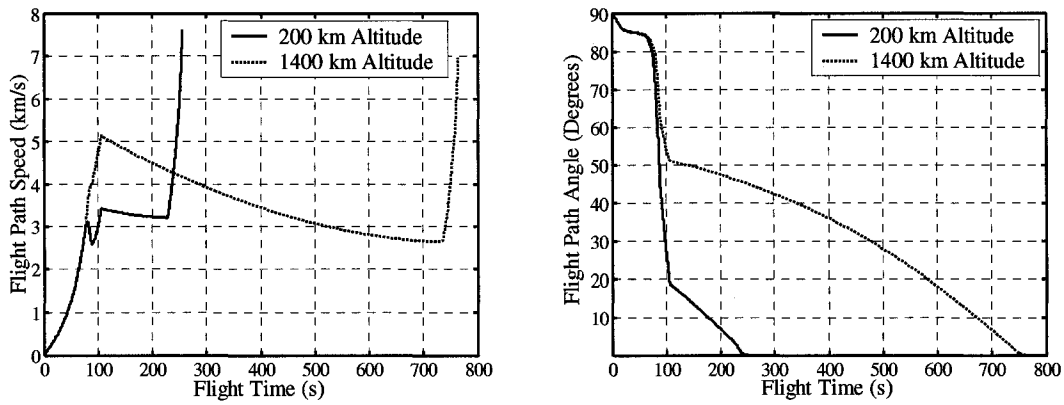


(a) Altitude (b) Total mass

Figure 3.6: Ground launched altitude and total mass profiles to 200 km and 1400 km circular altitude and 70° inclination

The 70° inclination altitude profiles, illustrated in Figure 3.6 (a), showed similar characteristics to those of the 60° inclination case. However, the flight time gap between the altitude profiles of the 60° and 70° inclination increased from 172% to 199% respectively yielding a modest 37 s increase in flight time. This increase could potentially have some guidance, control and power considerations. The total mass profiles in Figure 3.6 (b) were as expected based on the staging parameters described in Section 2.2.1.

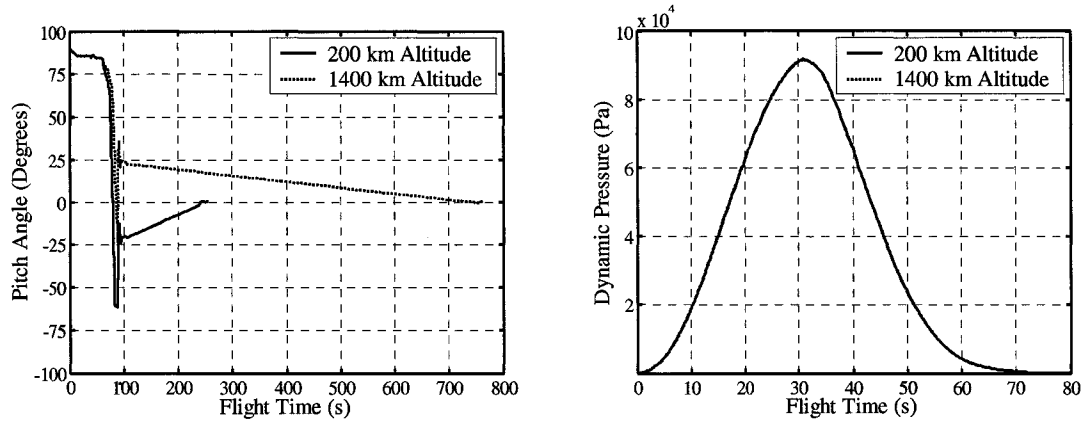
The flight path speed and angle profiles are shown in Figure 3.7. The trends identified in the 60° inclination case held for 70° inclination.



(a) Flight path speed (b) Flight path angle

Figure 3.7: Ground launched flight path speed and angle profiles to 200 km and 1400 km circular altitude and 70° inclination

The pitch angle and dynamic pressure profiles are shown in Figure 3.8. Although the pitch angle profiles (Figure 3.8 (a)) were similar to those in the 60° inclination case, the minimum pitch angle reached within the 100 s mark was 0.2% higher in the 200 km circular altitude case and 2.5% lower in the 1400 km circular altitude case. This indicated that for lower inclinations with low altitudes, the minimum pitch angle continues to increase which may yield the trajectory unfeasible while the opposite effect was identified with increasing inclination and altitude.



(a) Pitch angle (b) Dynamic Pressure

Figure 3.8: Ground launched pitch angle and dynamic pressure profiles to 200 km and 1400 km circular altitude and 70° inclination

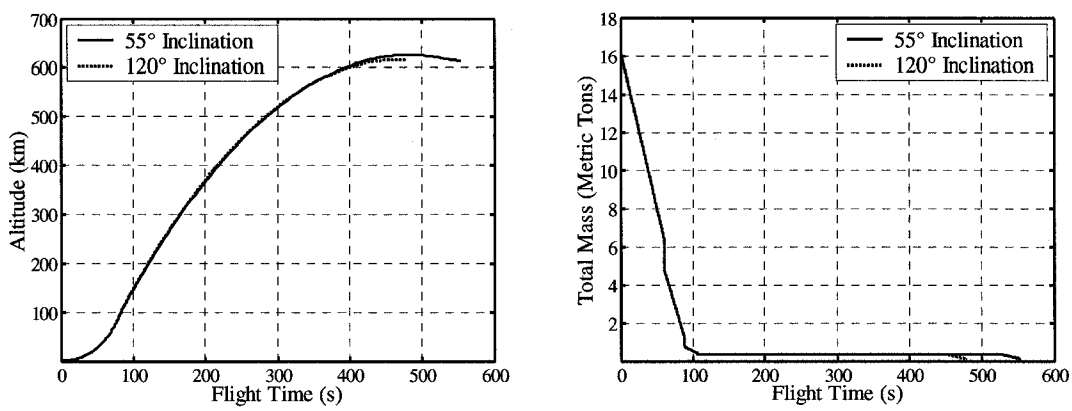
Similarly to the 60° inclination case, although the dynamic pressure profiles illustrated in Figure 3.8 (b) were nearly identical when varying altitude at the 70° inclination, during the peak dynamic pressure, the 1400 km circular altitude trajectory showed a consistently lower dynamic pressure of 0.04% but this was considered insignificant. It is also worth noting that the 70° inclination maximum dynamic pressure (i.e. 200 km circular altitude) was 0.1% lower than that of the 60° inclination case. Although this was a relatively small difference, it can potentially lead to less stringent heat protection considerations especially if this rule is extrapolated to larger inclination variations.

Constant Altitude

The 600 km and 800 km circular altitude ground launch performance to 55° and 120° inclination are described. Although the 600 km circular altitude case could reach inclinations lower than 55°, these were not considered for comparative purposes given that the minimum feasible inclination to 800 km circular altitude was 55°. However,

these are considered in Section 3.3. Similarly to the discussion above, these inclinations were selected for comparison as these were the two feasible extremes for this model with the constraints and parameters detailed in Section 2.2.1.

The results indicated that the 600 km circular altitude systems to 55° and 120° inclination could deliver 51.2 kg and 42.2 kg to orbit respectively. Figure 3.9 illustrates the altitude and total mass profiles. As shown in Figure 3.9 (a) both cases followed similar altitude profiles with the lower inclination case requiring 15.7% more time to reach the desired orbit. The altitude overshooting was attributed to the flight path angle described below.



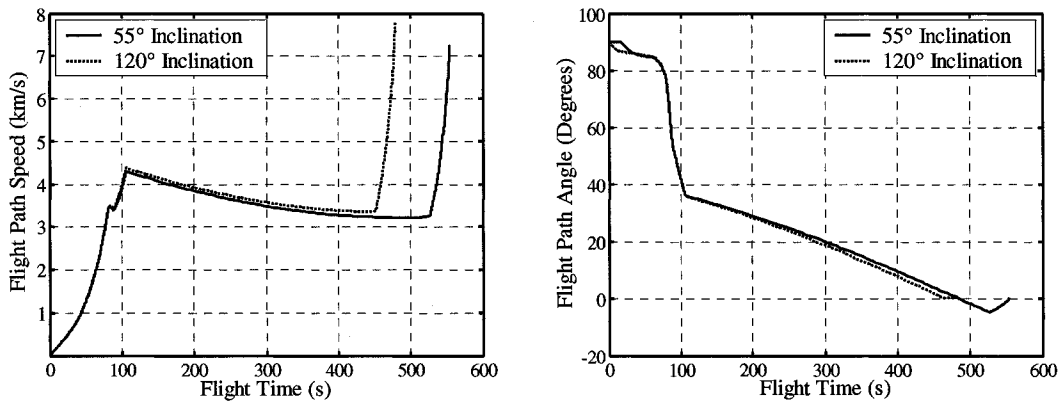
(a) Altitude (b) Total mass

Figure 3.9: Ground launched altitude and total mass profiles to 55° and 120° inclination and 600 km circular altitude

The total mass profiles shown in Figure 3.9 (b) were as expected based on the staging parameters described in Section 2.2.1.

The flight path speed and angle profiles are shown in Figure 3.10. Although both trajectories reached the same circular altitude, given that the altitude was different in both

cases for reasons noted above, the final flight path speeds differed to reflect the altitude disparity as shown in Figure 3.10 (a).

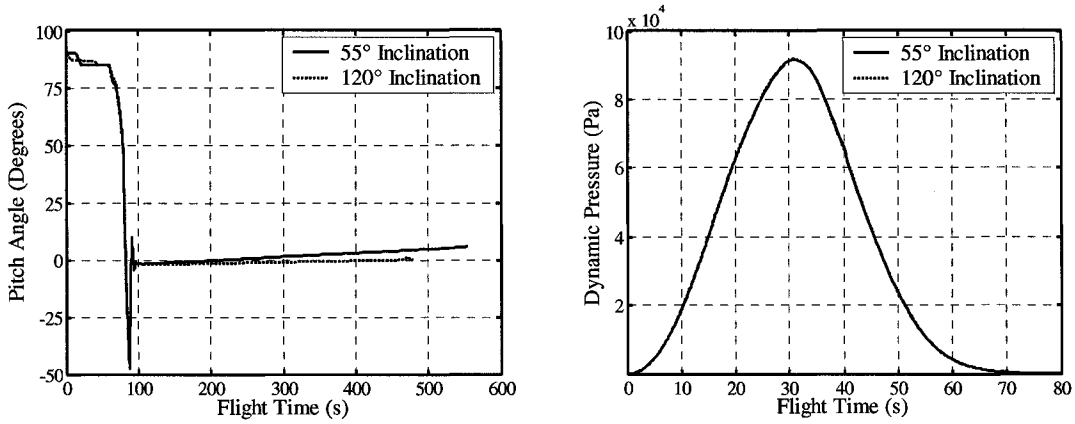


(a) Flight path speed (b) Flight path angle

Figure 3.10: Ground launched flight path speed and angle profiles to 55° and 120° inclinations and 600 km circular altitude

The flight path angle profiles illustrated in Figure 3.10 (b) ended at 0° but entered the negative region prior which was more evident in the 55° inclination case. This effect rationalized the overshoot of the desired circular altitude depicted in the altitude profiles in Figure 3.9 (a).

The pitch angle and dynamic pressure profiles in Figure 3.11 indicated that both trajectories exhibited similar parameters. The minimum pitch angle within 100 s of the trajectory in the 55° inclination case was 6.7% lower that of the 120° inclination.

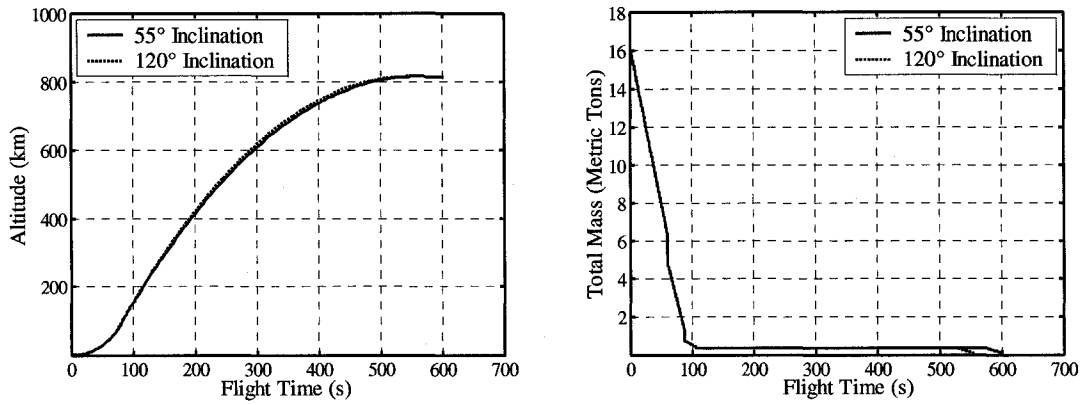


(a) Pitch angle (b) Dynamic pressure

Figure 3.11: Ground launched pitch angle and dynamic pressure profiles to 55° and 120° inclinations and 600 km circular altitude

With regards to the dynamic pressure, both were similar but during the maximum dynamic pressure segment, the 55° inclination case displayed a consistent lower dynamic pressure of 0.07% which was not considered significant but is worth noting given the same effects were observed in the above cases.

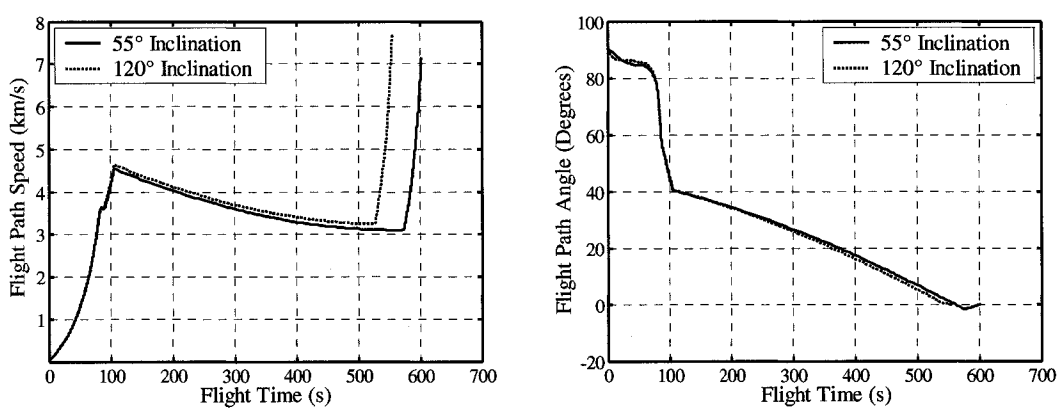
The 800 km circular altitude 55° and 120° inclination launch systems are investigated. These vehicles were found to be capable of delivering 52.3 kg and 42.3 kg to orbit respectively; the slight increase in payload capability from 600 km circular altitude is addressed in Section 3.3. The altitude and total mass profiles are illustrated in Figure 3.12. Both 55° and 120° inclination cases showed similar trends and the factors discussed for the 600 km circular altitude case apply.



(a) Altitude profile (b) Total mass profile

Figure 3.12: Ground launched altitude and total mass profiles to 55° and 120° inclinations and 800 km circular altitude

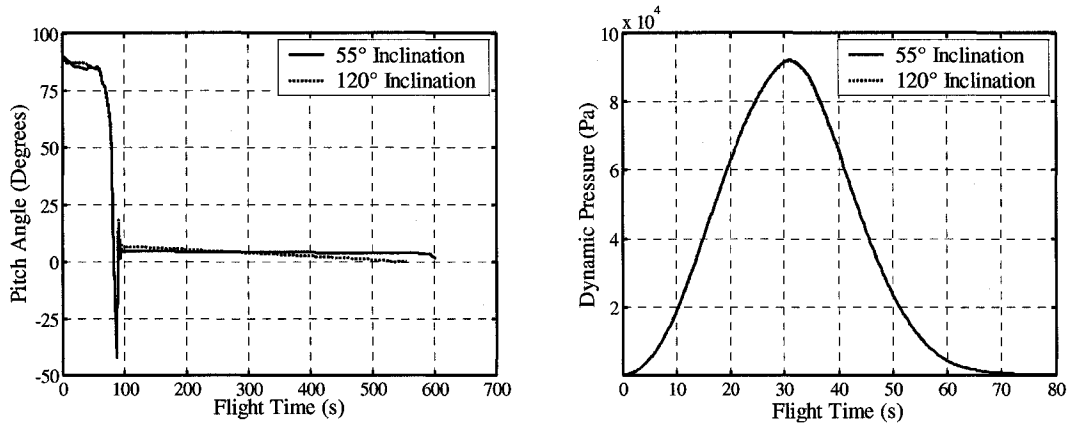
The flight path speed and angle profiles are shown in Figure 3.13 and indicated that the trends for 600 km circular altitude continued to hold. However, it is worth noting that the higher altitude case reached a lower negative flight path angle which reasoned the smaller overshoot to the desired altitude in the 800 km versus 600 km circular altitude cases.



(a) Flight path speed profile (b) Flight path angle profile

Figure 3.13: Ground launched flight path speed and angle profiles to 55° and 120° inclinations and 800 km circular altitude

The pitch angle and dynamic pressure profiles illustrated in Figure 3.14 were similar to those for the 600 km circular altitude case.



(a) Pitch angle (b) Dynamic Pressure

Figure 3.14: Ground launched pitch angle and dynamic pressure profiles to 55° and 120° inclinations and 800 km circular altitude

The minimum pitch angle, shown in Figure 3.14 (a), achieved within 100 s of the trajectory was lower than that of the 600 km circular altitude case; 11.2% for 55° inclination and 14.8% for 120° inclination. Also, the rate of this effect intensified with higher inclination. This indicated that the pitch angle profile approached that of the ideal case with increasing altitude.

3.2 Air Launch Scenario

This section addresses the air launch optimization results. The validity of the ASTOS results are first discussed using two methods. Subsequently, the trajectory analysis of various air launch optimizations is described followed by a parametric study to assess the effect of varying the initial flight path speed. A more detailed trajectory analysis is

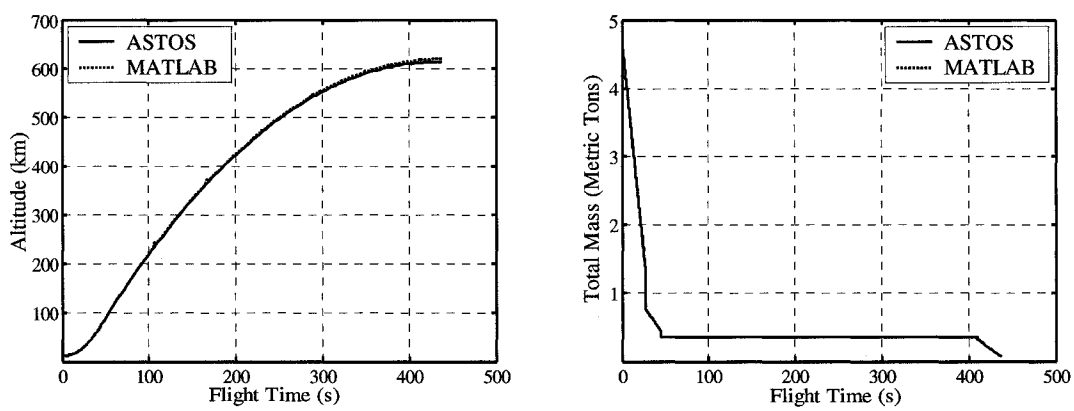
provided in Section 3.3 as indicated in Section 3.1. Also, the objectives enumerated in Section 3.1 are carried throughout this section.

3.2.1 Validation

Similarly to that described in Section 3.1.1, the ASTOS results were further validated using an in-house MATLAB model to compare the results and by evaluating the change in velocity of the vehicle model herein to that of an existing launcher. In this case, only one such vehicle is in operation i.e. the Pegasus XL.

MATLAB Simulations

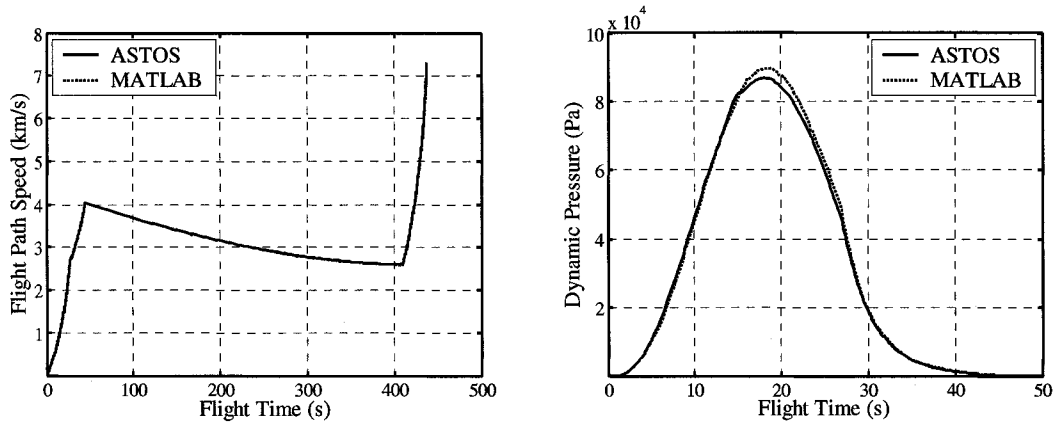
Selected MATLAB results are depicted in order to gain insight into the ASTOS package and validate the results. The altitude and total mass profiles are shown in Figure 3.15 and were both characterized by 0.48% differences with the ASTOS results. These were considered reasonable based on the discussion in Section 3.1.1.



(a) Altitude (b) Total mass

Figure 3.15: Air launched ASTOS and MATLAB altitude and total mass profiles to 600 km circular altitude and 60° inclination

The flight path speed and dynamic pressure profiles are shown in Figure 3.16. Here, the results were within 1.52% and 9.18% differences with that of ASTOS.



(a) Flight path speed (b) Dynamic pressure

Figure 3.16: Air launched ASTOS and MATLAB flight path speed and dynamic pressure profiles to 600 km circular altitude and 60° inclination

Although the Figure 3.16 results were considered reasonable based on the discussion in Section 3.1.1, the dynamic pressure profile was characterized by a slightly higher difference for reasons addressed in Section 2.2.2.

Velocity Budget Comparison

The change in velocity of the Pegasus XL, the only commercial air launched vehicle currently in operation, along with that of the air launched Orbital Express derivative were computed and shown in Table 3.2. The reference orbit was selected at 550 km circular altitude since this was the lowest feasible altitude for the air launched Orbital Express. As described in Section 3.1.1, the inclination was selected near the launch latitude to yield the maximum payload.

Table 3.2: Total change in velocity of the Pegasus XL and air launched Orbital Express derivative

Launch Vehicle	Parameter				
	Payload	**Altitude**	**Launch Latitude**	**Orbit Inclination**	**ΔV**
	kg	**km**	**Degrees**	**Degrees**	**m/s**
Pegasus XL	360	550	62.9 North	63.0	8047
Orbital Express	38	550	57.7 North	57.7	9239

The results revealed that the air launched Orbital Express' change in velocity was 14.8% above that of the Pegasus XL which provided confidence in the results based on the factors enumerated in Section 3.1.1.

3.2.2 Trajectory Analysis

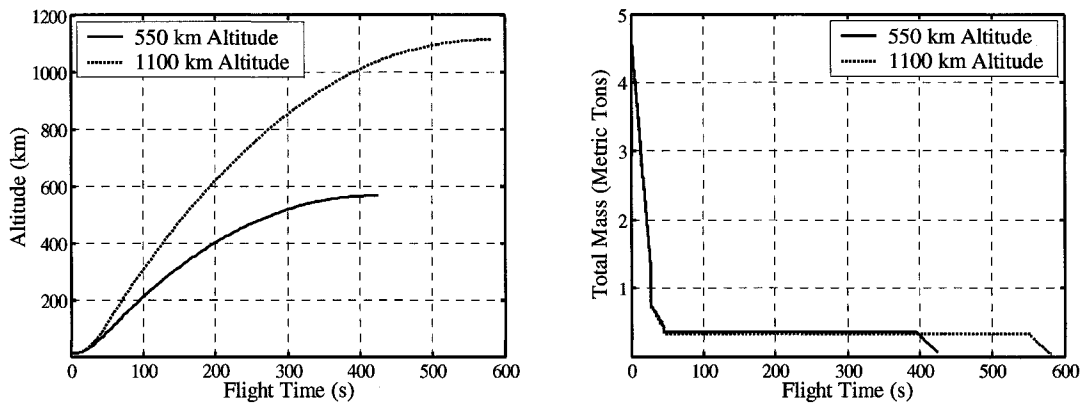
A trajectory analysis, based on ASTOS, of the air launched system is detailed. The discussion focuses on the trends for multiple optimizations at constant inclination and constant altitude respectively similarly to the ground launched case.

Constant Inclination

The 60° and 70° inclination air launch performances to 550 km and 1100 km circular altitude, the feasible extremes, are described. These inclinations and altitudes were selected for reasons addressed in Section 3.1.2.

The results indicated that the 550 km and 1100 km circular altitude and 60° inclination systems could insert 37.6 kg and 4.7 kg into orbit respectively. The 60° inclination altitude and total mass profiles are shown in Figure 3.17. As expected, the 550 km

circular altitude case achieved its orbital altitude with a reduced flight time of 26.9% given the shorter energy exchange duration.

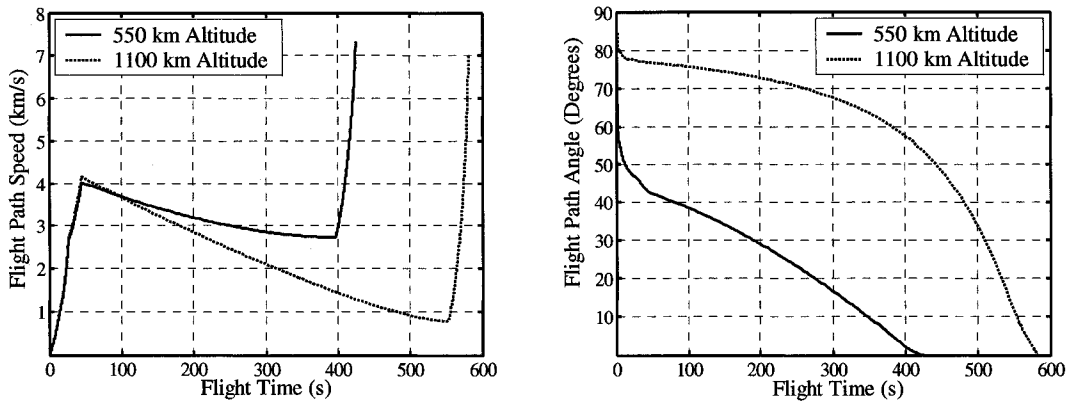


(a) Altitude (b) Total mass

Figure 3.17: Air launched altitude and total mass profiles to 550 km and 1100 km circular altitude and 60° inclination

The total mass profiles were as expected given the SRM parameters detailed in Section 2.2.1 and described in Section 3.1.2.

The flight path speed and angle profiles are shown in Figure 3.18. An interesting feature of the 1100 km circular altitude flight path speed case was the significant speed reduction during the Coast Arc phase which was attributed to the relatively high desired circular altitude.



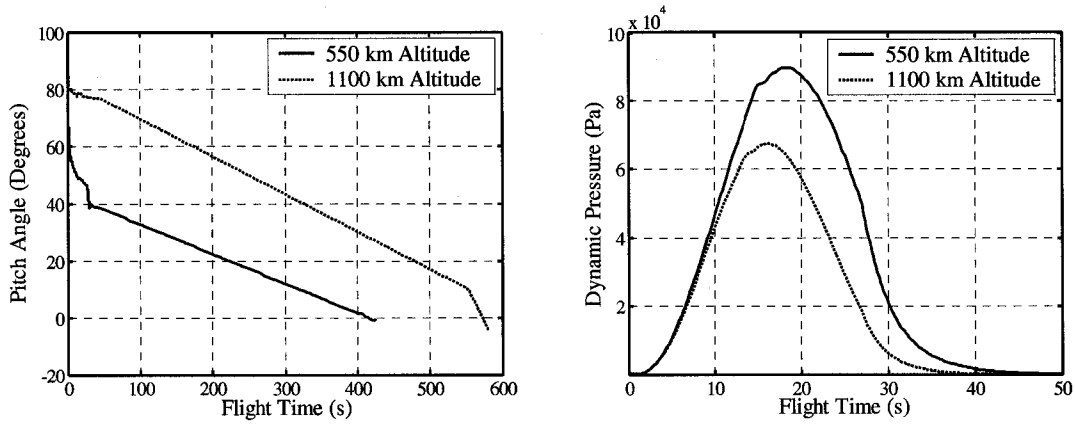
(a) Flight path speed

(b) Flight path angle

Figure 3.18: Air launched flight path speed and angle profiles to 550 km and 1100 km circular altitude and 60° inclination

The flight path angle profiles in Figure 3.18 (b) indicated that although both launch systems began at 90°, the 550 km and 1100 km circular altitude cases quickly decreased to approximately 60° and 80° respectively. This showed that higher orbital altitudes resulted in steeper trajectories such that the vehicle could quickly fly through the region of high atmospheric pressure despite the inherent gravity losses [Di Sotto, 2002].

The pitch angle and dynamic pressure profiles are illustrated in Figure 3.19. The pitch angle profiles (Figure 3.19 (a)) monotonically decreased in contrast to the ground launched cases owing to the lower dynamic pressures as shown in Figure 3.19 (b).

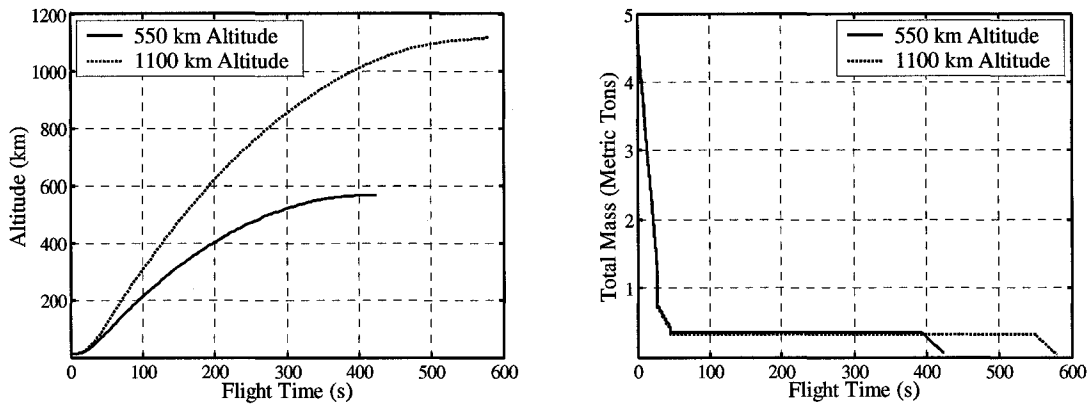


(a) Pitch angle (b) Dynamic pressure

Figure 3.19: Air launched pitch angle and dynamic pressure profiles to 550 km and 1100 km circular altitude and 60° inclination

It was clear from the Figure 3.19 (b) results that the higher altitude trajectory exhibited a lower dynamic pressure. This was largely attributed to the difference in the respective flight path speeds along with the atmospheric density to a lesser extent. The 1100 km circular altitude air launched vehicle's maximum dynamic pressure was 24.9% lower than that of the 550 km circular altitude case.

In order to determine if the above trends held at higher inclinations, the 70° inclination trajectory parameters are addressed. The 550 km and 1100 km circular altitude systems were found to be capable of delivering 35.6 kg and 3.8 kg into orbit respectively. As expected, the payload capability decreased given the higher inclination. The altitude and total mass profiles are illustrated in Figure 3.20. The trends identified for 60° inclination held for the 70° inclination case.

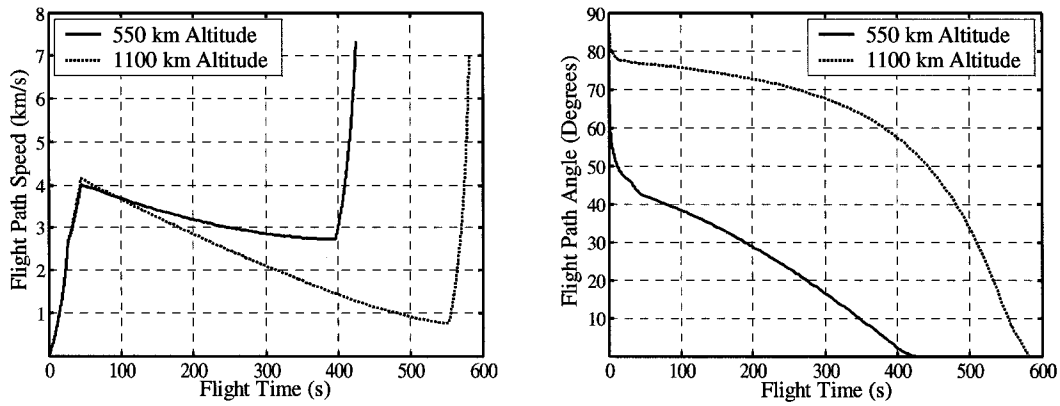


(a) Altitude (b) Total mass

Figure 3.20: Air launched altitude and total mass profiles to 550 km and 1100 km circular altitude and 70° inclination

It should be noted that similarly to the ground launched case, a higher inclination introduced an increase in the time gap from 26.9% to 27.1%, potential implications of which were described in Section 3.1.2.

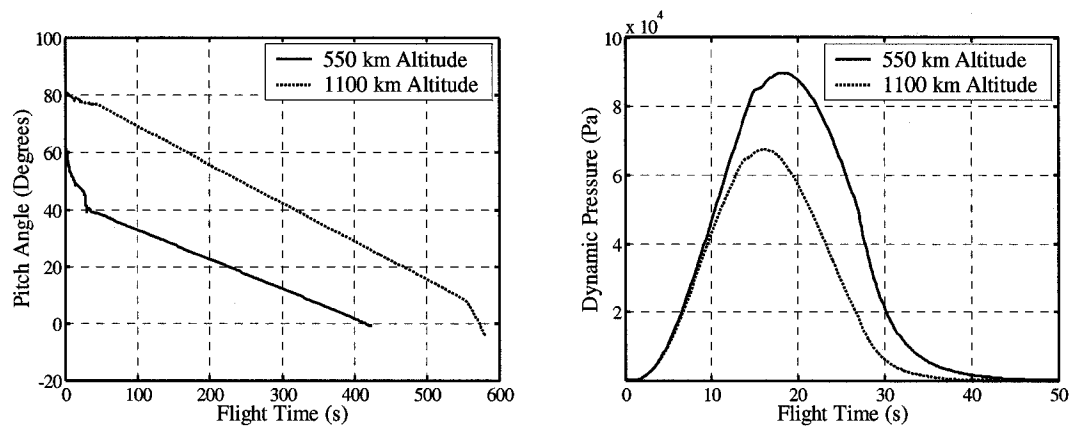
The flight path speed and angle profiles are shown in Figure 3.21. The 60° inclination trends continued to hold for the 70° inclination case.



(a) Flight path speed (b) Flight path angle

Figure 3.21: Air launched flight path speed and angle profiles to 550 km and 1100 km circular altitude and 70° inclination

The pitch angle and dynamic pressure profiles are shown in Figure 3.22.



(a) Pitch angle (b) Dynamic pressure

Figure 3.22: Air launched pitch angle and dynamic pressure profiles to 550 km and 1100 km circular altitude and 70° inclination

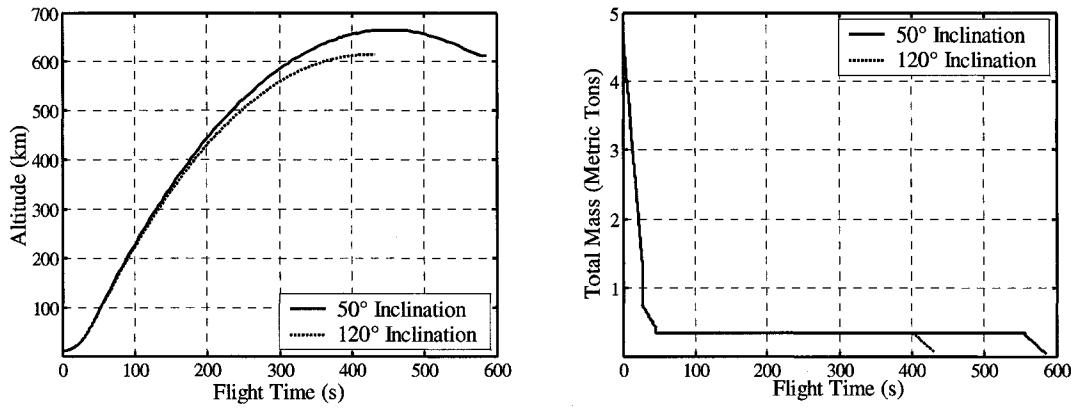
Although the 60° inclination trends were revealed by the 70° inclination case, the maximum dynamic pressure for the 550 km and 1100 km circular altitude cases were 0.22% and 0.02% lower respectively in the 60° inclination case which could potentially

lead to increased payload capability given the mass savings due to the lower heat shielding requirements.

Constant Altitude

The 600 km and 800 km circular altitude air launched performances to 50° and 120° inclination are described. Similarly to the discussion above, these inclinations were chosen for comparison as these were the two feasible extremes for this model with the constraints and parameters detailed in Section 2.3.3.

The results indicated that the 600 km circular altitude 50° and 120° inclination systems could insert 15.6 kg and 24.2 kg into orbit respectively. The altitude and total mass profiles are illustrated in Figure 3.23. The altitude profiles in Figure 3.23 (a) indicated that the trajectory duration to 50° inclination was 35.4% longer than that of the 120° inclination trajectory. Both 50° and 120° inclination temporarily exceeded their respective orbital altitude, most notably in the lower inclination case. This was attributed to the negative flight path angle during this segment similarly to the ground launched case in Section 3.1.2.

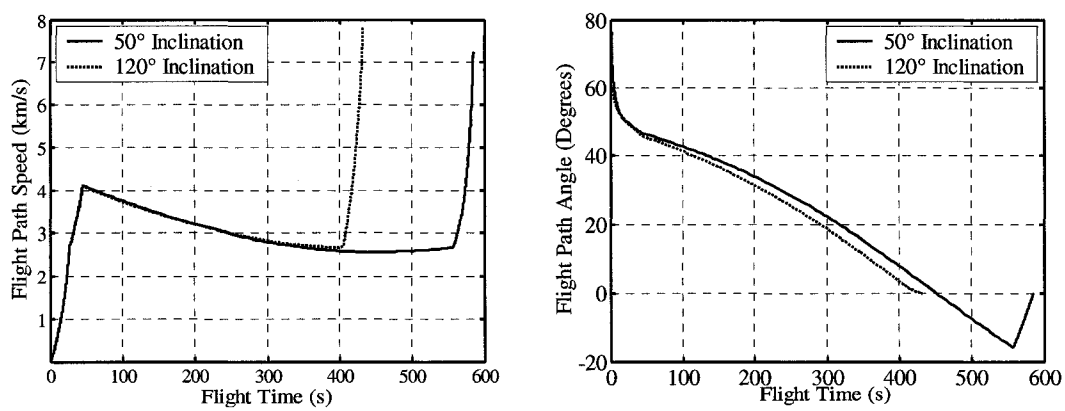


(a) Altitude (b) Total mass

Figure 3.23: Air launched altitude and total mass profiles to 50° and 120° inclination and 600 km circular altitude

The total mass profiles were as expected based on the SRM parameters described in Section 2.2.1.

The flight path speed and angle profiles are shown in Figure 3.24. The flight path speed profiles were also expected based on the discussion in Section 3.1.2.

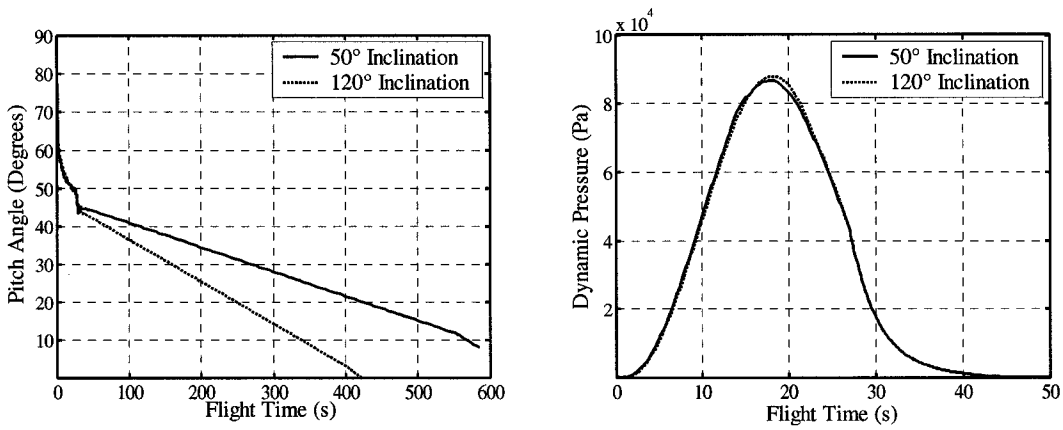


(a) Flight path speed (b) Flight path angle

Figure 3.24: Air launched flight path speed and angle profiles to 50° and 120° inclination and 600 km circular altitude

The flight path angle profiles in Figure 3.24 (b) indicated that the lower inclination case entered the negative flight path angle regime near the end of the trajectory which reflected the altitude overshoot observed in Figure 3.23 (a).

The pitch angle and dynamic pressure profiles are illustrated in Figure 3.25. Based on the pitch angle profiles in Figure 3.25 (a), both 50° and 120° inclination cases followed similar trends but diverged approximately 20 s into the trajectory. Subsequently, the lower inclination case followed a steeper trajectory.



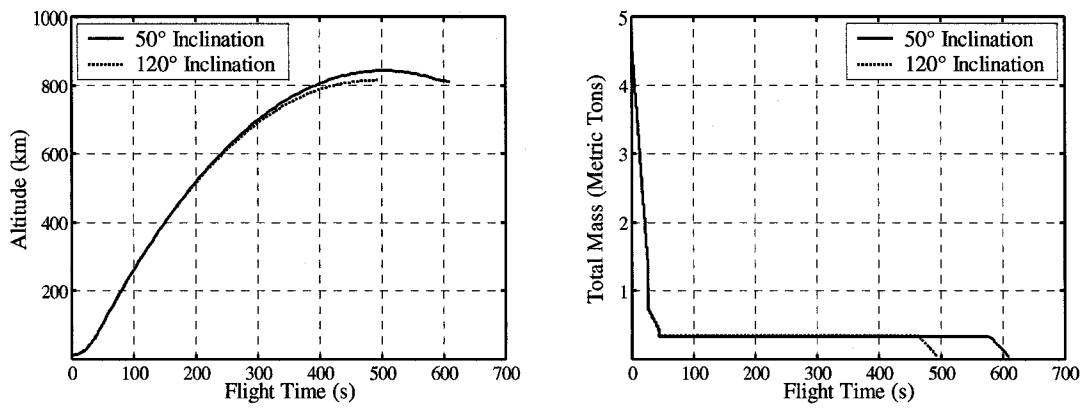
(a) Pitch angle (b) Dynamic pressure

Figure 3.25: Air launched pitch angle and dynamic pressure profiles to 50° and 120° inclination and 600 km circular altitude

The dynamic pressure profiles were similar in both cases as depicted in Figure 3.25 (b).

The 800 km circular altitude 50° and 120° inclination launch system results yielded 8.1 kg and 15.2 kg payload capability to orbit respectively. As expected, an increase in altitude led to a lower payload capability. The altitude and total mass profiles in Figure

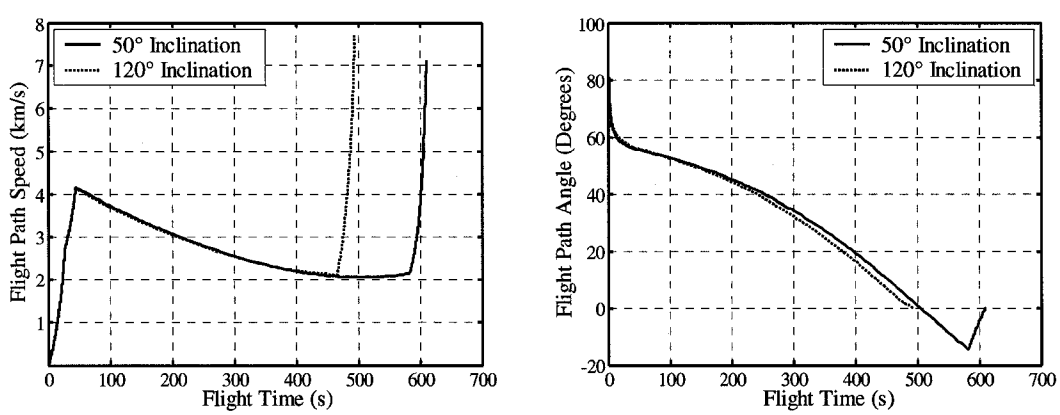
3.26 revealed similar trends for both 50° and 120° inclination cases and the factors discussed for the 600 km circular altitude case apply.



(a) Altitude (b) Total mass

Figure 3.26: Air launched altitude and total mass profiles to 50° and 120° inclination and 800 km circular altitude

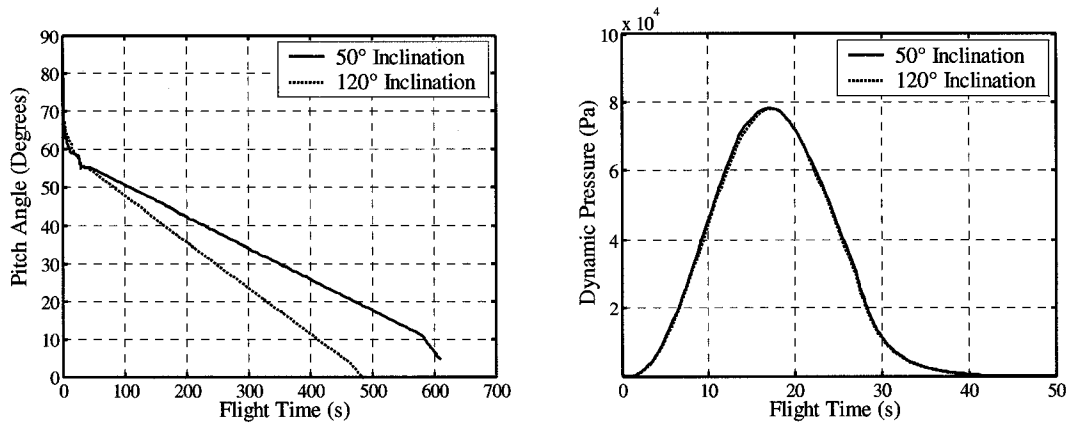
Based on the Figure 3.27 flight path speed and angle profiles, there were no significant changes observed in comparison with the 600 km circular altitude case.



(a) Flight path speed (b) Flight path angle

Figure 3.27: Air launched flight path speed and angle profiles to 50° and 120° inclination and 800 km circular altitude

The pitch angle and dynamic pressure profiles are shown in Figure 3.28. The factors for the 600 km circular altitude case apply since similar trends were observed in the higher altitude optimizations.



(a) Pitch angle (b) Dynamic pressure

Figure 3.28: Air launched pitch angle and dynamic pressure profiles to 50° and 120° inclination and 800 km circular altitude

The higher altitude cases pitched down at a slower rate early in the flight which in turn, resulted in lower maximum dynamic pressure values. Specifically, the 800 km circular altitude 50° and 120° inclination cases were 9.9% and 11.4% lower respectively than those of the 600 km circular altitude cases. These yield similar implications to those described above.

3.2.3 Parametric Study

The initial air launch velocity was investigated with ASTOS to assess its effect on the payload capability. The results of this analysis are depicted in Figure 3.29.

It should be noted that the assessment is highly sensitive to the pitch angle at launch, assumed to be 90° in the case herein. As discussed in Section 2.2.2, since a compromise trajectory is required to minimize losses, trajectory losses such as drag and gravity will influence the magnitude of the benefit of the initial velocity [Sarigul-Klijn (a), 2005].

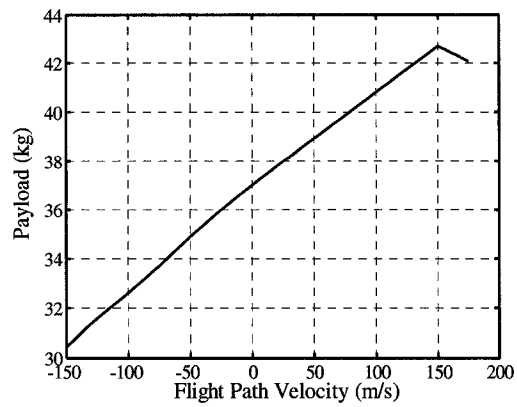


Figure 3.29: Payload profile with varying launch flight path velocity to 600 km altitude and 60° inclination

Between -150 m/s and 150 m/s, the slope of Figure 3.29 was 0.04 kg/(m/s). This trend was expected since the initial velocity imparted to the launcher reduced the overall change in velocity which the launch vehicle had to produce to deliver the payload into orbit. However, Figure 3.29 revealed that for initial velocities higher than 150 m/s, the payload capability decreased due to the constraints and was not feasible above 175 m/s. Given that the slope was relatively small, missions could afford a longer time lapse between separation from the carrier aircraft to first stage ignition in an effort to simplify operations and promote stronger safety features. It was clear however, that the added complexity of attempting to provide an initial velocity to the launcher outweighed the added payload mass in the case of a 90° launch pitch angle.

3.3 Ground and Air Launch Performance Investigation

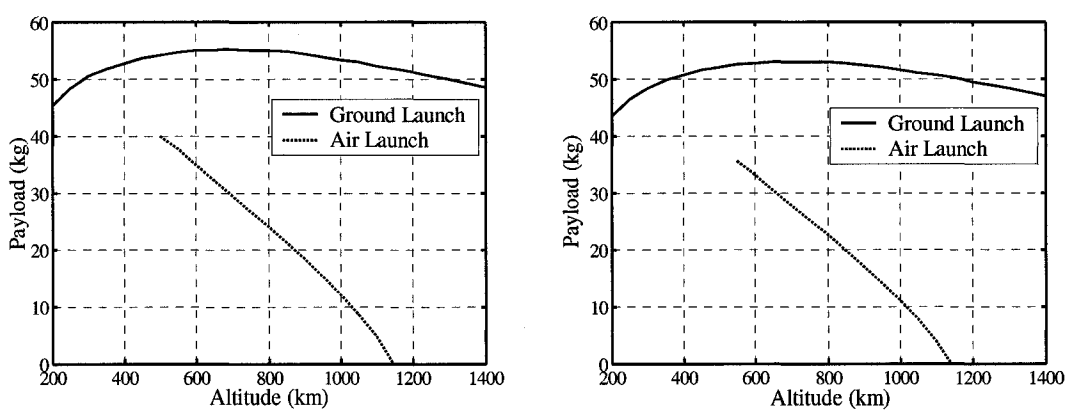
The foregoing discussion focuses on the performance segment of the ground and air launch cases. This investigation was largely based on ASTOS.

3.3.1 Performance Analysis

The performance analysis is investigated, in turn, at constant inclination, altitude and SSO.

Constant Inclination

The payload performance results to 60° and 70° inclination are shown in Figure 3.30. As expected, a higher inclination led to lower payload capability. The ground launch trends are described followed by those of the air launch.

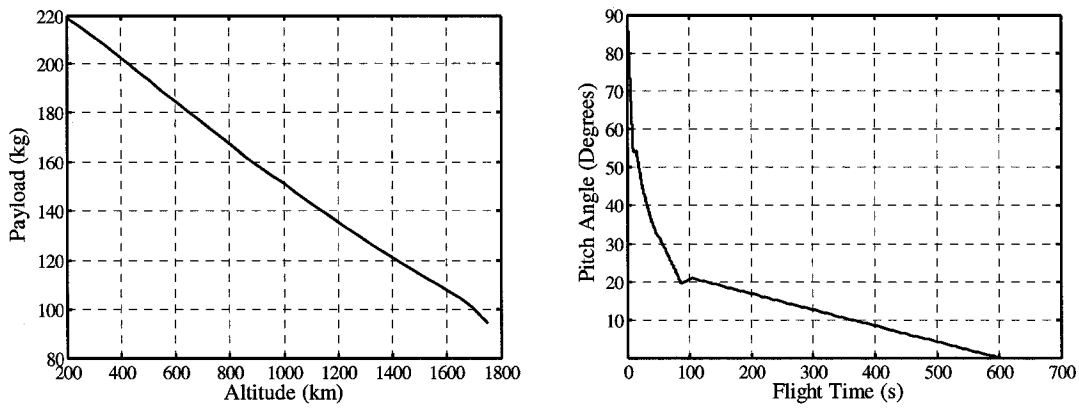


(a) 60° inclination (b) 70° inclination

Figure 3.30: Payload performances to 60° and 70° inclination

The ground launch results indicated that the maximum payload did not occur at the minimum altitude. However, launchers typically exhibit a monotonically decreasing

payload for increasing altitude. The main reason for this behaviour was the fact that the path constraints did not permit the payload to be higher for lower altitudes; this was verified by optimizing the models with no dynamic pressure and heat flux constraints as illustrated in Figure 3.31.



(a) Payload performance (b) Pitch profile

Figure 3.31: Ground launched payload performance to 60° inclination and pitch profile to 600 km circular altitude and 60° inclination without dynamic pressure and heat flux constraints

Without a throttling capability which is inherent to SRMs, it was surmised that the optimization software attempted to burn excess fuel by varying the pitch angle to satisfy the dynamic pressure and heat flux constraints as depicted in Section 3.1.2. However, this behaviour was eliminated when these constraints were removed as shown in Figure 3.31 (b). In the case of the air launch, since the dynamic pressure was lower than that of the ground launch as discussed below, this launch method did not exhibit such a trend.

Other factors contributing to the ground launch payload trend are depicted. It is important to recall that the Orbital Express proposal consisted of combining a selection of existing

motors with few additions and modifications along with a development methodology which relied upon proven, off-the-shelf technology [Hughes, 1996]. In order to emphasize commonality, reduce costs and increase reliability, the motor data incorporated in the models remained unchanged for all orbit altitudes and inclinations. In turn, although the trajectories were optimized for each orbit height and inclination, the models were not; if the amount of fuel per stage was modified according to the trajectory requirements, this would yield a typical performance curve. Stated differently, a consequence of stacking existing motors was that the thrust to weight ratio was not optimal for all stages; increasing the thrust to weight ratio of the second stage would be beneficial in generating a typical performance curve. Also, as previously indicated, reliability of the data used in the models may have contributed to this discrepancy. This behaviour was thus apparent in all ground launch optimizations.

It is worth noting that if the launcher was configured with one or more liquid and/or hybrid stage, this would enable the operator to regulate the tank filling. In such a case, additional propellant mass, beyond what is required to reach the desired orbit, would be subtracted from the propellant mass of selected stages, subject to the ascent profile, and added to the payload mass. Alternatively, if the motors were not yet developed, the tank fillings could be regulated as per the trajectory and any propellant not required would be added to the payload mass; in such a case, the structural mass ratio would vary as per the propellant mass and thus yield a typical performance curve. An additional feature of a liquid and/or hybrid stage is their throttling capability. Here, the vehicle could reduce its

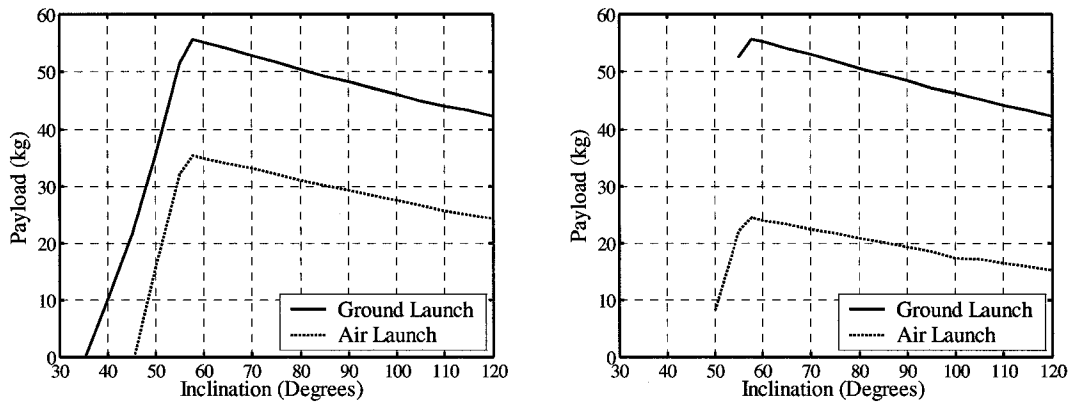
acceleration, particularly in the region of high atmospheric density, to decrease the maximum dynamic pressure and remain within the prescribed constraints [Donahue, 2004]. This is the case in many launchers such as the Ariane; it results in a pitch profile similar to that obtained in the air launch case. Within the context of this study, a hybrid motor was deemed to be the best alternative due to the growing Canadian hybrid technology as compared to liquid engines. Hybrid rockets are comprised of both liquid (or gas) and solid phase propellants initially in separate tanks. Although conventional hybrids have a solid fuel and a liquid oxidizer, reverse hybrids also exist in which the propellant phases are inverted. For conventional hybrid rockets, the solidified fuel grain is enclosed in the combustion chamber and the liquid oxidizer is fed from its tank by a variety of different feed system means and injected into the combustion chamber. This modulated flow enables a throttling, stop and restart capability which can mitigate the maximum dynamic pressure challenge described above and also introduce added safety characteristics due to the mission abort opportunity [Humble, 1995]. In turn, this could increase a launch vehicle's payload capability given the lower emphasis needed for constraints but also since multiple burns can be performed. Although hybrids can potentially deliver an I_{sp} similar to that of SRMs, they are characterized by combustion stability challenges [Greatrix, 2005] and lower density I_{sp} [Sutton, 2001]. In the case of an air launch, the smaller volume of SRMs is critical in order to fit within the carrier aircraft's geometrical restrictions. However, in order to benefit from the hybrid motor advantages described above, [Kwon, 2003] investigated the feasibility of using a hybrid motor to replace the Pegasus XL's first stage SRM. Their study demonstrated that a

hybrid motor could replace the first stage within the prescribed geometrical envelope and that such motors could be used for very small air launched vehicles [Kwon, 2003]. Also, [Karabeyoglu, 2005] investigated a 3603 kg two staged hybrid propulsion vehicle launched from a modified Canberra aircraft. According to their study, the launcher could deliver 45 kg into a 200 km polar orbit from 13,716 m altitude and a 60° pitch angle [Karabeyoglu, 2005].

As attested in Figure 3.30, the above-described cost minimization approach resulted in lower payload capability at altitudes lower than approximately 700 km. In regards to the air launch case, the maximum payload exists at a minimum altitude given that the dynamic pressure was lower. Also, the results indicated that a ground launch can access lower inclinations within the given constraints. Purely based on performance, without regard to operational matters, it was clear that at these inclinations, resorting to a ground launch was preferable for all altitudes but a case may be made to resort to an air launch for altitudes below approximately 700 km with a responsiveness requirement. It is important to note that for comparison purposes, the air launched launching site was set to that of the ground launched case throughout the optimizations as discussed in Section 2.3.3. However, an air launch could conceivably be conducted from any location. By benefiting from the air launch's strategic launch location, this would typically yield higher payload capabilities and access to lower altitudes and inclinations while continuing to satisfy the constraints.

Constant Altitude

Having assessed the effect of varying altitude, the inclination variation is investigated and the results are shown in Figure 3.32.



(a) 600 km (b) 800 km

Figure 3.32 Payload performances to 600 km and 800 km circular altitude

As expected, the results in Figure 3.32 indicated that the maximum payload capability was at the launch latitude, 57.7°. Both launch methods revealed sharp decreases in payload capability for inclinations below the launch latitude with the ground launch system capable of delivering payloads to nearly 35° inclination compared to 45° for the air launch case. In most cases, below these respective inclinations, the dynamic pressure constraint was no longer satisfied. Above 57.7° inclination, the 600 km circular altitude ground and air launch systems were characterized by -0.21 kg/° and -0.18 kg/° slopes respectively. When the circular altitude was increased to 800 km, the ground and air launch systems exhibited -0.21 kg/° and -0.15 kg/° slopes respectively. This indicated that as the altitude increased, the effects of higher inclinations were limited in the ground launch case but subsided for air launches. With increasing altitude, although the ground

launch was capable of maintaining a fairly constant payload capability, the air launched system's payload capability decreased significantly. As a result, a ground launch becomes increasingly favourable with higher altitude. However, low inclinations became unfeasible with increasing altitude. Ultimately, for the scenario herein, although a ground launch could deliver heavier payloads to orbit, an air launch became increasingly advantageous for low inclinations with altitude.

Sun-Synchronous Orbit

Figure 3.33 illustrates the payload performance to SSO. The respective curves were similarly shaped to that of the 60° and 70° inclination cases, however the curves are translated to lower payload capabilities due to the higher inclination.

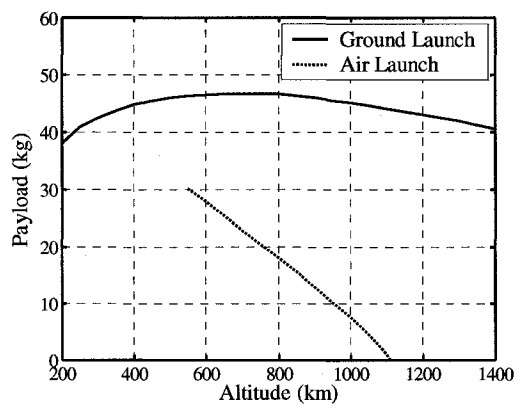


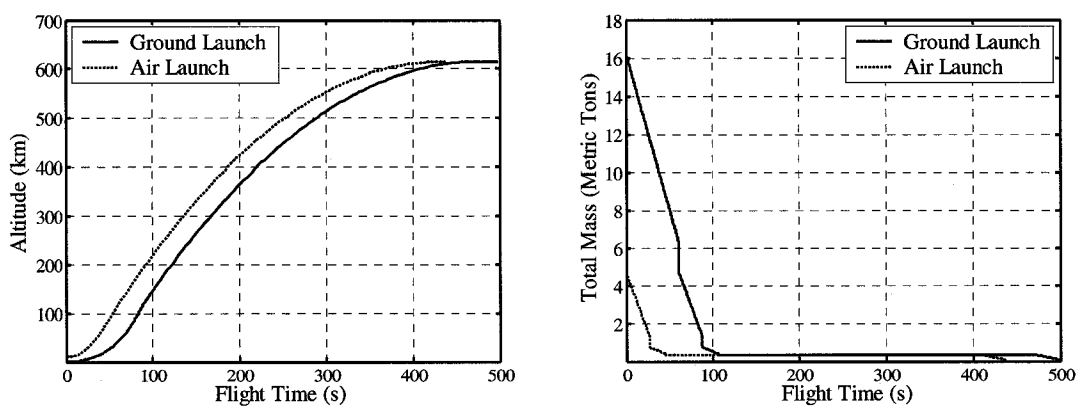
Figure 3.33: Payload performance to SSO

3.3.2 Trajectory Analysis

The altitude profiles to 600 km circular altitude and 60° inclination are shown in Figure 3.34 (a). These orbital parameters were selected given the adequate altitude for remote sensing, which is one of Canada's strong interests and an inclination close to the launch

latitude to yield high payload capability. The ground and air launched vehicles were found to be capable of delivering 54.9 kg and 34.9 kg respectively. In turn, this corresponds to a payload to take-off mass ratio of 0.34% and 0.74% for the ground and air launch respectively. The air launch clearly exhibits a favourable payload mass fraction, which is largely attributed to its higher launch altitude. However, the practical magnitude of this benefit is subject to the operational implications of such a launch architecture which are addressed in Chapter 4.

Based on the results shown in Figure 3.34, the ground launch flight time was 13.8% longer than that of the air launch which, as discussed in Section 2.2.2, led to higher trajectory losses given that these are integrated in time. This however, was strictly the launcher flight time; the carrier aircraft's flight time was not taken into account. When considering the total flight time, the air launch was significantly longer than that of the ground launch.



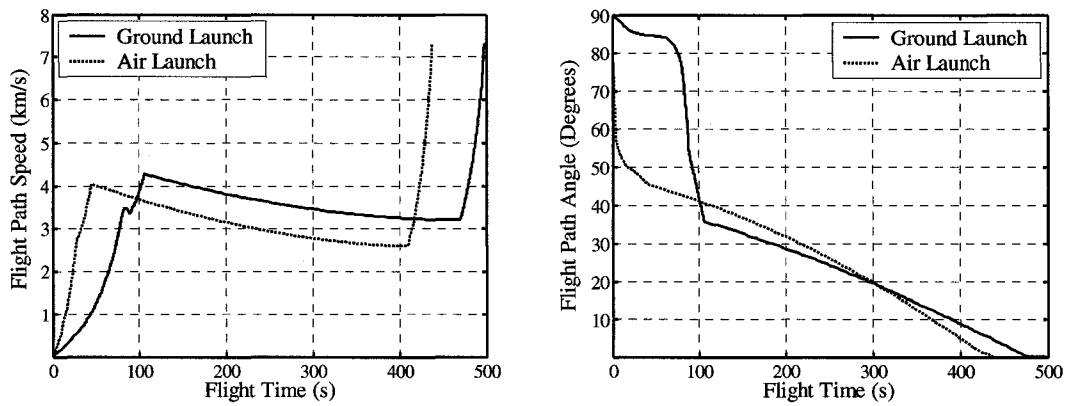
(a) Altitude (b) Total Mass

Figure 3.34: Altitude and total mass profiles of a ground and air launch to 600 km circular altitude and 60° inclination

The total mass profiles shown in Figure 3.34 (b) were characterized by discontinuities signifying ignition and burnout of each motor and the Coast Arc phase during which the mass was constant. Following second and third stage separation, which occurred at approximately 105.3 s and 44.8 s for the ground and air launched cases; the vehicles reached altitudes of 160.1 km and 74.4 km respectively. It should be noted that the trends were similar to those obtained by [Greatrix, 2005] who conducted a ground versus air launch study. In contrast to the investigation herein where the vehicles were four and three SRMs respectively, [Greatrix, 2005] compared three staged SRM-based ground and air launched systems. Also, although their upper stages were identical in both launch versions, the air launched first and second stage masses were 36% and 53% that of the ground launched system respectively [Greatrix, 2005]. In turn, these were surmised to promote the benefits enumerated in Section 3.3.1.

From the launch conditions to the Coast Arc phase, the flight path speed accelerated to 4.3 km/s and 4.0 km/s while the flight path angle decreased to 35.7° and 45.0° in the ground and air launch cases respectively as shown in Figure 3.35. Here, the vehicles began converting kinetic for potential energy. The lower acceleration early in the flight of the ground launched case compared to that of the air launch was attributed to the motor characteristics, instantaneous mass and higher atmospheric density. Although the slower acceleration reduced the dynamic pressure, it increased gravity losses. In the case of the air launch, since it operated at higher altitude, it quickly accelerated through the region of

high atmospheric density and minimized gravity losses since the maximum dynamic pressure constraint was not of concern as described below [Griffin, 1991].



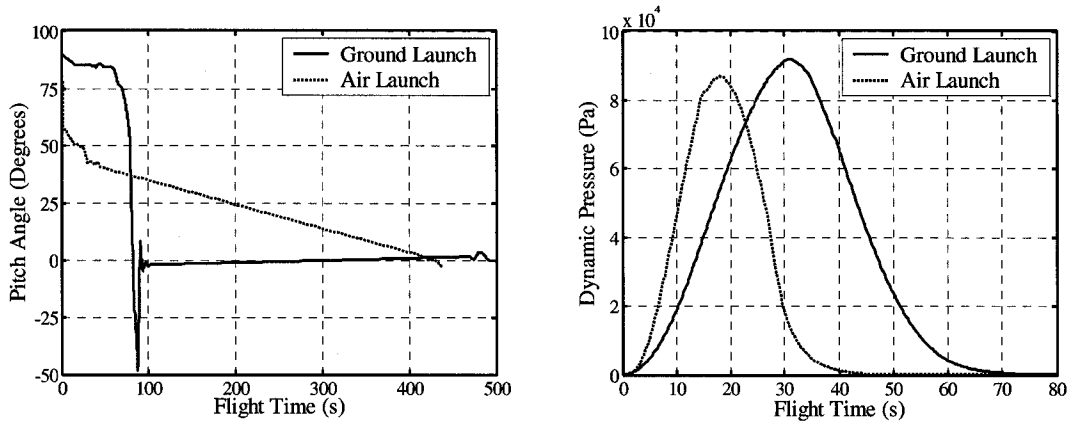
(a) Flight path speed (b) Flight path angle

Figure 3.35: Flight path speed and angle profiles of a ground and air launch to 600 km circular altitude and 60° inclination

Both ground and air launched systems began vertically at a 90° flight path angle as shown in Figure 3.35 (b) with the air launch quickly decreasing its flight path angle. In the case of the ground launch, the vertical launch was critical to clear the launch pad and a vertical air launch was dictated by the recommended separation sequence described in Section 2.3.2. The ground launch remained at a high flight path angle early in the flight in order to minimize drag losses by attempting to clear the region of highest atmospheric pressure as soon as possible. However, the air launch quickly reduced its flight path angle to minimize gravity losses given the fact that the worst region of atmospheric pressure had been bypassed. Following the initial boost, the flight path angle profiles decreased monotonically to near 0° at orbit injection [Greatrix, 2005].

The pitch angle and dynamic pressure profiles are shown in Figure 3.36. The small pitch oscillations in Figure 3.36 (a) were surmised to be a result of the pitch control attempting to reach the final orbital requirements [Greatrix, 2005]. The ground launch's negative pitch values early in the flight were due to the imposed constraints and associated factors previously discussed; these could introduce challenges which are addressed below.

In the ground launched case, the pitch angle approached 0° midway through the third stage burn while this occurred midway through the fourth stage in the case of the air launch. Despite the 0° pitch angle, which incurred a non-zero angle of attack given the flight path angle profile, the atmospheric density had sufficiently decreased such that relatively low aerodynamic loading was generated [Greatrix, 2005]. However, the high pitch rate of the ground launched system at approximately 60 s could introduce challenges for sensitive payloads but also due to the additional steering losses. As discussed in Section 3.1.2, this effect was mitigated with increasing altitude and inclination. With regards to the relatively high air launched pitch rate early in the flight, the flight path speed was sufficiently low which rendered the manoeuvre feasible. Finally, given the flight path angle profiles described above, the air launch yielded smaller steering losses compared to those of the ground launch.



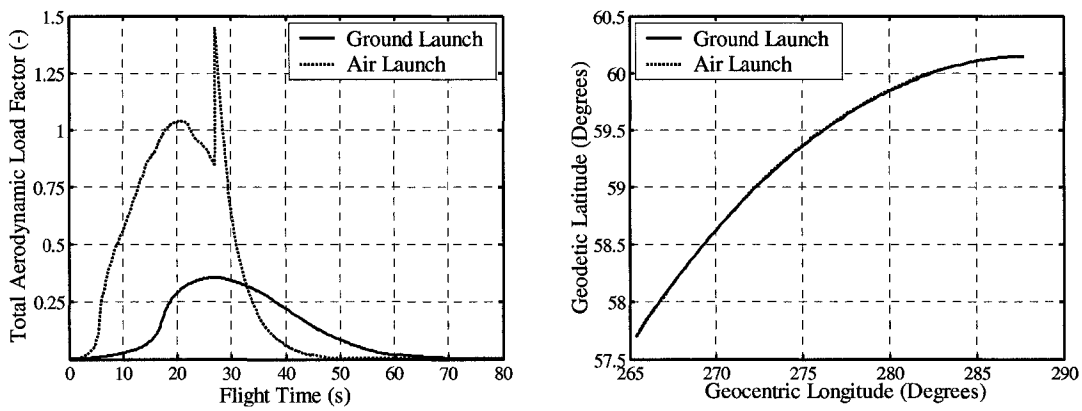
(a) Pitch angle (b) Dynamic Pressure

Figure 3.36: Pitch angle and dynamic pressure profiles of a ground and air launch to 600 km circular altitude and 60° inclination

The maximum dynamic pressure, which was constrained to 92 kPa, occurred at 30.9 s, and 18.0 s for the ground and air launch cases which was midway through the first and second stages respectively. Based on Figure 3.36 (b), the ground launch's maximum dynamic pressure was 5.8% larger than that of the air launched version. At the respective peak dynamic pressures, although the ground launched system was significantly slower than that of the air launch, i.e. 0.6 km/s versus 1.3 km/s, the air launched system was at an altitude of 19.0 km compared to 9.2 km for the ground launched system. In turn, the atmospheric density was 352.2% higher than that of the air launched vehicle which dominated the velocity difference and yielded a higher dynamic pressure. As a result, the air launch's lower peak dynamic pressure led to a more efficient trajectory such as lower gravity losses and/or design flexibility [Karabeyoglu, 2005].

The total aerodynamic load factor, defined as the instantaneous aerodynamic force and vehicle weight ratio, and groundtrack profiles are illustrated in Figure 3.37 [Institute of

Flight Mechanics and Control (a), 2005]. As per Figure 3.37 (a), it was apparent that the air launch case exhibited a larger load factor which was mainly attributed to the larger thrust to weight ratio. Since small SRMs typically accelerate faster than larger ones, which was the case herein, such vehicles are more influenced by drag and promote higher aerodynamic load factors [Whitehead, 2006].



(a) Total aerodynamic load factor (b) Groundtrack

Figure 3.37: Total aerodynamic load factor and groundtrack profiles of a ground and air launch to 600 km circular altitude and 60° inclination

As shown in Figure 3.37 (b), the groundtracks were similar in both cases; this was expected given the fact that both launch systems were permitted to access any launch azimuth, were launched from the same location and were reaching the same orbital parameters.

3.3.3 Assessment

At constant inclination and SSO, the results clearly indicated that the ground launched vehicle could deliver heavier payloads to orbit at all circular altitudes. Also, the ground system was feasible at low inclinations where the air launch was limited. However, this

applies strictly to cases where the strategic launch location inherent to an air launch is not incorporated. This methodology was carried throughout this study for comparative purposes along with the operational limitations of an air launch as discussed in Chapter 4. In cases where the strategic launch of an air launch is utilized, this will generally lead to higher payload capabilities and access to lower altitudes and inclinations. Based on the assumptions herein, the study indicated that for low altitudes, an air launch may be warranted if a responsive service is required.

At constant altitude, the results indicated sharp payload capability drops below the inclination corresponding to the launch latitude. Although the ground launch was found to be capable of achieving lower inclinations at low altitude, this was reversed with higher altitude where an air launch was favourable. In fact, when increasing the altitude, reaching low inclinations became increasingly challenging for both launch architectures. Above the launch latitude inclination, the payload capability of both launchers decreased at a slower rate than that below the launch latitude inclination and a ground launch was preferred from a capability perspective. A ground launch was increasingly advantageous with increasing altitude given the fact that the air launched system's payload capability for all inclinations decreased significantly.

The ground launched flight time was found to be slightly larger than that of the air launch which led to additional trajectory losses. However, the lower mass of the air launched system led to increased drag losses. The ground launch case underwent stronger gravity

losses early in the flight given its smaller acceleration compared to that of the air launch while the latter promptly accelerated through the region of high atmospheric density and minimized gravity losses. Drag losses were minimized in the ground launched case with a relatively high flight path angle while the air launched system focused on minimizing gravity losses with a relatively significant decrease in flight path angle early in the flight. The air launched vehicle could afford to conduct such a manoeuvre given that the region of highest atmospheric density was bypassed. A potential concern for the ground launched case was the high pitch rate at the end of the first stage which was attributed to satisfying the path constraints. Although this was mitigated with higher altitude and inclination and the aerodynamic loads were relatively low, sensitive payloads may require additional considerations. Alternatively, a motor with a higher I_{sp} or a throttleable stage would decrease this effect altogether. The ground launch's higher peak dynamic pressure compared to that of the air launched version introduced additional loading and heating implications. In cases where the fairing could be modified to sustain a lower dynamic pressure, this could potentially lead to higher payload capability in the air launch case. Alternatively, the lower dynamic pressure could be considered as an added safety characteristic. Given the flight path angle profiles, the air launch exhibited lower steering losses compared to those of the ground launch largely due to the altitude benefit. Finally, in cases where the aerodynamic loading is of concern, a ground launch is preferred.

As discussed above, the study also indicated that a throttling capability in one or more stages would be greatly beneficial particularly in the ground launch case. In turn, this could lead to increased payload capability, less stringent design requirements and additional safety features.

3.4 Summary

The analytical investigation of a ground versus air launch based on the Orbital Express has been described. The results were largely based on ASTOS and were validated with in-house models. The analysis identified and contrasted key performance and trajectory parameters between the ground and air launch in addition to those between multiple altitudes and inclinations of the respective launch architectures. Having addressed favourable ground and air launch circumstances from an analytical perspective, operational aspects are discussed in Chapter 4.

CHAPTER 4

OPERATIONAL ASPECTS

This chapter addresses key operational segments of the ground and air launched cases from launch responsiveness to safety factors along with cost considerations. Some air launch advantages include an initial potential energy imparted to the air launched vehicle and in some cases, an initial positive kinetic energy which can be supplied by the carrier aircraft. Also, the lower atmospheric density translates into a reduction in drag and atmospheric pressure losses. The main objectives of this chapter include delineating operational similarities and differences of the launch methods along with identifying

scenarios in which, from a logistical perspective, either a ground or air launch is preferred.

4.1 Responsiveness

An air launch benefits from the fact that the carrier aircraft can transport the launch system to a given launch latitude for efficient access to a desired orbit [Sarigul-Klijn (b), 2005] which renders the carrier aircraft to behave as a mobile launch platform [Karabeyoglu, 2005]. As discussed in Chapter 2, the study herein assumed that both ground and air launched systems were ignited from the same latitude for comparative purposes but the advantages addressed above for an air launch could be implemented. In the case of a ground launch constrained to a given launch site for associated ground infrastructure, plane change manoeuvres may be required to reach the desired orbit inclination which increases the required propellant mass and ultimately the launch cost. Alternatively, if other ground launch sites were available and appropriately equipped, this could mitigate the propellant mass challenge but would incur costs associated with transporting the launch vehicle to the launch site.

Current US range policies for a launch authorization, which were surmised to be largely similar to that of a Canadian orbital launch mission, include that the impact zone of a launcher's stages be clear of maritime traffic irrespective of the emission of official notices. Since such notices can require multiple days, [Sarigul-Klijn (b), 2005] note that a ground launch can never be responsive. Their study indicated that a ground launch cannot

deliver a payload to any desired orbit within 24 hours, which was one of the US Air Force Operationally Responsive Spacelift's objectives, due to the geographic constraints of current launch sites. For instance, only easterly and southerly launches can be conducted from Cape Canaveral and Vandenberg respectively. An air launch is clearly beneficial in such cases since the carrier aircraft can launch the system over an ocean, adequately positioned to eliminate maritime traffic concerns [Sarigul-Klijn (b), 2005]. Finally, since few ground launched systems have been conducted on their first scheduled launch date, an air launch offers the potential for launch on schedule [Sarigul-Klijn, 2001] by virtue of factors such as orbital inclination access and resiliency against unfavourable weather conditions which are discussed in Section 4.2.

4.2 Launch Site

An air launch benefits from a strategic launch location. This alternative permits access to all inclinations and is resilient against launch denial due to atmospheric conditions which could prevent a ground launch. This launch option can be conducted worldwide although practically, some ground infrastructure and base is still needed which could potentially limit the inclination access and launch location; this is addressed in Section 4.3. In cases where a ground launch shares the launch site with other vehicles, an air launch mitigates this challenge as it can be operated independently of scheduling predicaments of such scenarios [Sarigul-Klijn, 2000]. In addition, an air launch could conceivably be conducted from any airport with appropriate runway length and propellant loading

facilities, if needed, whereas this would not be possible for a ground launch even if the infrastructure was available given the associated noise and sonic boom [Donahue, 2004].

The air launch also minimizes any political objections to a launch; Israel for instance, must launch into retrograde orbits due to political reasons as discussed in Section 1.3.1 [Isakowitz, 2004]. In this case, an air launch would be advantageous as the launches could be conducted eastward provided the carrier aircraft launches the vehicle over a suitable area such as a water mass.

Ultimately the launch point would be selected based on suitable launch weather, over water ascent trajectory, locally clear of other aircraft, ships and launchers, directly under the desired orbital plane and appropriate orbit phasing to enable one orbit rendezvous, if required [Sarigul-Klijn, 2000].

4.3 Launch Infrastructure

Air launched vehicles are independent of a dedicated launch facility which yields significant savings in ground support costs. However, similarly to an expendable launcher, vertically mounted on a launch pad, although an air launch does not need a launch pad, it requires ground infrastructure once mated to the carrier aircraft as exemplified by the Pegasus XL. This infrastructure, nonetheless, can be portable which enables launch location flexibility [Covault, 2003] but in some cases, it may constrain the launch site. Also, since an air launch could potentially use existing runways for take-off

and landing, lower infrastructure costs are expected compared to those of a ground launch [Cook, 2003].

In the case of an air launch, access to the vehicle after take-off is very limited, if at all possible unless the launch system is inside the aircraft, compared to that of ground launches. In addition, testing and inspection of the vehicle is clearly simplified in the ground launch case. However, an air launch has the added flexibility to access spaceports with a favourable air, rail, road and sea transportation network.

4.4 Launch Vehicle Components

A ground launch is not typically limited in size and/or mass. Such is the case for an air launch which is subject to the carrier aircraft's capabilities [Karabeyoglu, 2005]. However, an air launched case exhibits a higher payload fraction in comparison to a ground launched vehicle as confirmed in Section 3.3.2 [Donahue, 2004]. Also, an air launch can deliver heavier payloads to orbit with less propellant, given the smaller change in velocity required which is largely attributed to the thinner atmosphere, hence a smaller vehicle can be realized. However, the lower stage mass may result in a smaller propellant mass fraction. A typical theoretical required change in velocity for an ascent to orbit is 7.5 km/s. When accounting for gravity and other trajectory losses, the total change in velocity increases to approximately 9.2 km/s and 8 km/s for the ground and air launched architectures respectively. It should be noted that the changes in velocity for both ground and air launches are within 2.9% and 0.6% of the averages computed in

Sections 3.1.1 and 3.2.1 respectively. Although the 1.2 km/s change in velocity difference between a ground and air launch may not seem considerable, given the exponential nature of Equation 4.1, even a small velocity change modification yields significant payload capability effects [Bonometti, 2006]. The initial to final mass ratio as a function of the change in velocity is given by [Sutton, 2001]:

$$\frac{m_0}{m_f} = e^{\frac{\Delta V}{g_0 I_{sp}}} \tag{4.1}$$

Where:

m_0 Initial mass

m_f Final mass

As depicted in Equation 4.1, even a small decrease in change in velocity mitigates the inert mass fraction effect and provides designers with more flexibility. For instance, in considering a single stage to orbit vehicle with a 450 s I_{sp}, the effect of a higher inert mass fraction is significantly decreased as shown in Table 4.1 [Bonometti, 2006].

Table 4.1: Ground and air launch inert mass fraction effects

Parameters	**Units**	**Ground Launch**		**Air Launch**	
Initial mass	Metric tons	500	500	500	500
ΔV	km/s	9.2	9.2	8.0	8.0
Inert mass fraction	%	10	12	10	12
Payload mass	Metric tons	13.52	2.46	35.11	24.54
Payload mass difference	%	448.6		43.1	

Although the case herein was based on existing rocket motors with a prescribed structural mass, the effect described above increases the design safety margin instead of introducing design flexibility [Bonometti, 2006]. Moreover, given the simplified abort modes inherent to an air launch as further discussed in Section 4.11, an air launch system mitigates the relatively high abort loads which increase a vehicle's structural mass [Sarigul-Klijn, 2000].

The length of a given rocket nozzle is shortened, subject to the altitude, such that it can operate at the local atmospheric conditions at the expense of higher I_{sp} with increasing altitude. Although extending bell housings have been developed for increased performance, these often incur additional mass and may negatively affect the launcher's reliability. However, by air launching a vehicle where the atmospheric pressure is lower, the nozzle can be designed for the given launch altitude which would enable improved thrust expansion (i.e. higher nozzle area ratio) and lead to better propulsion efficiency [Bonometti, 2006].

It is worth noting that in cases of hybrid and liquid propellant rockets, an air launch is particularly well suited, given the lower atmospheric pressure, to be configured with a vapour pressure propellant feed system as exemplified by SpaceShipOne. Here, the liquid is expelled from its tank via the pressure provided by the stored liquid's internal energy. Instead of turbopumps and gas generators, which are typically used in pump fed launch systems, a vapour pressure feed system can minimize costs and complexity [Sarigul-

Klijn, 2006]. However, in the case of cryogenic propellants, these systems incur the added complexity of propellant boil-off which can be particularly challenging in air launch cases where access to the launch system is limited.

4.5 Reusability

The air launch is characterized by a reusable stage (i.e. carrier aircraft) which uses the oxygen in the atmosphere and may be beneficial subject to the launch frequency. According to [Sarigul-Klijn, 2000], infant mortality is a major contributor to launch vehicle failures. Since the highest probability of failure is at first use, warranties typically accompany new products due to infant mortality. As a result, this promotes a reusable system to minimize such failure sources and in turn provides the opportunity for incremental testing to further mitigate failures, improve safety and ultimately reduce insurance costs [Sarigul-Klijn, 2000]. Although the air launched alternative is not fully reusable, the carrier aircraft's reusability has the potential to deliver the benefits enumerated above.

4.6 Integration and Testing Facilities

Integration and testing facilities are critical for any launch campaign. This is mainly attributed to the extensive flight qualification requirements and procedures especially when a new system is being developed. Since new subsystems, such as those for an air launch, will require expensive and lengthy testing and flight qualification processes, it is economically better to purchase foreign components. However, the technology could be

improved and/or developed as the project matures [Labib, 2004]. The above discussion applies to both launch methods but given the added ground launch know-how, the air launch will likely incur additional costs in these respects. Finally, for equivalent payload capabilities, since an air launch offers a smaller vehicle compared to that of a ground launch, this translates into less to inspect and refurbish [Donahue, 2004].

4.7 Launch Window

Air-breathing systems typically provide expanded launch windows compared to rocket-powered vehicles which increases mission flexibility. Subject to the desired inclination, launch windows can be up to several hours compared to minutes for air-breathing and rocket-based systems respectively [Cook, 2003]. Since the carrier aircraft can commute to the desired launch position, this introduces the opportunity to reach any orbital inclination such as that of the ISS without a plane change, eliminates waiting time for the desired orbital plane to pass above the ground launch site and enables one orbit rendezvous. As a result, for human missions, crews spend less time in the launch vehicle, approximately 1.5 hours, which reduces electrical power along with environment control and life support system requirements and in turn, maximizes payload mass [Sarigul-Klijn, 2000]. Ultimately, an air launch exhibits fewer delays since specific launch windows can be accessed directly, thereby minimizing waiting time [Sarigul-Klijn, 2006].

Another important aspect is the requirement for a restricted or unoccupied airspace for the air launch deployment which can potentially negatively influence a launch schedule

[Covault, 2003]. In fact, this is a major concern for Pegasus XL launches as discussed in Section 1.3.1.

These factors have important considerations to obtain the launch authorization. In the case of a Canadian military launch, the Minister of National Defence can authorize a launch. However, for civilian payload launches, in order to obtain a launch approval from the Minister of Transport, the civilian party must supply Transport Canada with documentation pertaining to matters including an environment assessment, mission manuals, flight plan data, safety organization, policy, procedures and equipment, accident contingency plan, insurance and a launch risk assessment. It should be noted that the same restrictions apply for human civilian launches but with additional details regarding the risk assessment. Once the civilian party has submitted the above-detailed documentation, Transport Canada conducts a risk assessment study and advises the candidate of the results and concerns. Provided Transport Canada's study is satisfactory, a launch authorization is issued to the party [Labib, 2004].

4.8 Launch Range

An air launch can potentially reduce range and tracking fees subject to the selected trajectory. In the US, the governmentally controlled range costs are US $1 to $1.3 million on a national range. Although RLVs can reduce launch costs as discussed in Section 1.1, range expenses remain unaffected. Some launch companies steer clear of federal ranges by launching from air or sea. However, range and tracking fees apply to an air launch if

the ascent profile intercepts a national range. Conducting a launch in international waters such as Sea Launch can potentially reduce range fees but regulatory concerns such as the International Traffic in Arms Regulation may need to be addressed, particularly if the trajectory utilizes a foreign range [Jurist, 2005]. When operating independently of national ranges, an air launch can circumvent scheduling limitations often associated with such ranges. Multiple days are required between missions at a given launch range which can typically accommodate a single launch at time. However, some air launch concepts may not have the ability to capitalize on the federal range independence. This may arise if components such as explosive bolt debris or cables are either designed or susceptible to be jettisoned, as is the case of the separation sequence detailed in Section 2.3.1. In turn, launch operators may be instructed by government and safety regulators to perform the launch over a prescribed zone such as a national range or ocean [Sarigul-Klijn, 2004].

4.9 Air Traffic and Licensing

An air launch system, capable of operating from typical runways, along with future RLVs conceivably utilizing these same runways will benefit from a strategic launch location. However, the reserved airspace allocation for such vehicles coupled with the higher launch frequency potential could create an undesirable predicament for the air transportation community. Also, in contrast to aircraft, since launch vehicles are characterized by lower relative reliability, additional considerations are required to address associated operational requirements such as the influence of launch systems on air traffic and populations [Cheng, 2003].

The operation of launch vehicles is fundamentally different from aircraft. For instance, launch systems achieve supersonic flight conditions within the same altitude range utilized by commercial aircraft. Also, launch vehicles are characterized by designed stage jettisoning along with lower relative reliability which incurs a higher failure probability and ensuing debris; all of which descend through the airspace. Although Special Use Airspace (SUA) currently segment launch operations from air traffic, commercial airliners may raise objections to the SUA particularly if the higher launch frequency potential materializes [Cheng, 2003].

The main safety considerations of launch vehicles include addressing jettisoning events and disintegration, either intentional as part of the flight termination system or unintentional. The Configurable Airspace Research and Analysis Tool, which enhances NASA Ames Research Centre's Future Air traffic management Concepts Evaluation Tool, evaluates the safety of launch operations and their air traffic interactions. These analysis tools define and propagate time-dependent hazardous airspace regions created by a given launcher's debris which is critical for the Federal Aviation Administration's licensing procedures. In turn, air, sea and ground traffic routes at a safe distance from potential landing sites of a launcher's debris, both intentional and accidental, can be created to ensure public safety [Cheng, 2003].

4.10 Complexity

From the vehicle perspective, the added stage of the ground launch adds complexity to the system when compared to that of the air launch. However, on a system level, the launcher's couplings and integration to the aircraft along with the fact that a significant ground launch heritage and know-how exists in comparison to that of air launches suggest that an air launch is in fact more complex.

Air launched vehicles must account for the cost/lease of the carrier aircraft as well as the additional cost associated with developing and operating a demonstrator vehicle. An air launch would likely require a larger investment and is more complex than a ground launch; areas of particular challenge include the separation segment along with the lower flight heritage which may result in a decreased reliability rate.

4.11 Safety and Reliability

An air launch promotes safety characteristics and simplified abort modes due to the relatively high launch altitude along with the carrier aircraft's range such that the launch can be conducted sufficiently far from populated areas [Karabeyoglu, 2005]. In fact, the launch location flexibility can promote public safety downrange. Also, since the air launched version is configured with three stages, as opposed to four for the ground launch, this is one less jettisoned stage to consider. Crew onboard the carrier aircraft may be at risk but this could be avoided with uninhabited aircraft launches or using a throttleable first stage such as a hybrid motor. In addition, an air launch improves

reliability and minimizes technical risk since staging occurs at higher altitude with lower atmospheric pressure [Sarigul-Klijn, 2000].

The initial altitude and velocity of an air launched system increase the emergency response time for human or automated intervention in comparison to a ground launch. Typically, a launch system's risk reduction is associated with high costs, largely attributed to the degree of complexity along with the low engineering tolerances such that mass can be minimized. However, since the air launch introduces additional safety measures described above, it increases the intact payload recovery probability and minimizes disruptions which accompany a payload loss [Bonometti, 2006]. For instance, a parachute recovery followed by a water landing may be feasible provided the abort execution time is conducive to a cross range [Sarigul-Klijn, 2000]. In contrast, a ground launch must accurately jettison the capsule such that it reaches an altitude sufficiently high for a parachute recovery [Sarigul-Klijn, 2006]. Finally, in the case of a catastrophic failure, this study surmised that an air launch would likely require less time to recover and return to normal operating conditions in cases where the significant infrastructure of a ground launch requires extensive refurbishing.

During a typical ground launch campaign, the close proximity to the Earth's surface plays a critical role in the vehicle's design. The direct reflection of the launcher's exhaust plume creates strong mechanical vibration loads for which the launch system, including the payload, must withstand. In turn, this can lead to additional mass and redundancy and the

required vibration testing can negatively affect cost and scheduling. Typical launch pads are equipped with exhaust flame deflectors for acoustic attenuation and vibration damping; these facilities incur up-front and maintenance costs as described in Section 4.13. In this case, an air launch is clearly at an advantage since direct vibrations are dampened through the air pressure and velocity while the significance of reflected vibrations is minimized above approximately 300 m. In turn, the degree of infrastructure can be reduced while introducing additional opportunities for abort, recovery, lower risk and total cost [Bonometti, 2006].

As discussed in Section 1.3.2, air-breathing vehicles can have the ability to abort a given mission during or soon after take-off. The safety level of such a system is particularly noted in its inherent cross range capability which introduces additional potential abort opportunities and recovery sites in comparison to rocket-powered systems. According to studies performed under contract with NASA MSFC, "horizontal take-off air-breathing vehicles offer an order of magnitude decrease in failure rates over vertical take-off rocket configurations" [Cook, 2003]. It is worth noting that horizontal launches are gaining attention such as Scaled Composite's SpaceShipTwo and White Knight Two along with Andrews Space's Gryphon Aerospaceplane.

4.12 Insurance

Insurance fees including launch, delivery and liability coverage are approximately 15% of the payload, launcher, facility and service costs. In contrast to an air launch, a typical

ground launch will likely incur reduced insurance premiums given the larger pool of know-how. However, an air launch could potentially reduce such fees by ferrying to an area sufficiently far from populated regions and performing the launch [Jurist, 2005]. The same effect would likely result from improved abort procedures of an air launch where intact payloads are returned [Bonometti, 2006]. Ultimately, the strongest influence on insurance premiums is flight heritage and increased system reliability.

4.13 Cost

As the above sections inherently described cost considerations, the main recurring and non-recurring costs, business aspects and low cost mission approaches are addressed.

4.13.1 Recurring and Non-Recurring Costs

Typical launcher resources include vehicle stages, launch pads and runways, launch control centres, processing facilities, clean rooms, access towers, ground support equipment, electrical power, lifting equipment, and transporters [Rainey, 2004].

One of the most significant recurring cost reductions for the air launch is the fact that it has less expendable hardware and infrastructure as shown in Table 4.2 [Anderson, 1996].

Table 4.2: Space launch system's recurring costs excerpt [Anderson, 1996]

Recurring Costs	Ground Launch	Air Launch	Rationale
Expendable rocket hardware	Strong	Variable	Air launch has less rocket stages
Launch platform	Strong	N/A	Only ground launch has platform
Carrier aircraft maintenance	N/A	Strong	Only air launch has carrier aircraft
Operations personnel	Comparable		Similar personnel requirements
Facilities maintenance	Strong	Variable	Ground launch: additional support facilities
Equipment maintenance	Comparable		Similar equipment
Support inventory maintenance	Strong	Variable	Ground launch: additional components create higher inventory demand
Insurance	Variable	Strong	Higher comparative ground launch heritage

Although an air launch is characterized by aircraft and runway maintenance along with fuel and operations costs, these expenses may occasionally be warranted in comparison to the cost of an additional motor and associated ground infrastructure for a ground launch [Anderson, 1996]. The reusability of the aircraft segment may incur reduced costs for high launch rates. However, based on the current market trend described in Section 1.1, it is unlikely that this will materialize. The additional maintenance costs of the carrier aircraft segment are expected to dominate. Additionally, if the carrier aircraft is not leased, substantial capital will be required. The launch system's transportation costs between the manufacturing, test and recovery sites will likely be smaller for an air launch

since the system is assumed to exist as opposed to a ground launch where a larger carrier must be identified [Donahue, 2004]. According to [Anderson, 1996], the support hardware reduction, the launch site flexibility along with the improved range safety of an air launch will promote lower recurring costs. It should be noted that the air launch's mobility eliminates the refurbishing costs of the ground infrastructure in the case of a catastrophic ground launch.

Since the motors in this study were assumed to be developed, the air launch is at a disadvantage given the development costs associated with the carrier aircraft's modifications such that the air launched vehicle can be carried and released at the prescribed conditions depicted in Section 2.3.2 [Karabeyoglu, 2005]. In the case of a ground launch, the ground infrastructure is a significant non-recurring cost [Bonometti, 2006].

4.13.2 Business Aspects

In order to reduce launch costs beyond engineering challenges, [Jurist, 2005] proposed developing a private sector orbital space vehicle based upon a commercially successful suborbital industry. Critical factors in developing a suborbital industry include an appropriate business plan, streamlined management operations and well designed safety considerations [Jurist, 2005].

Selected necessary conditions for an air launch to yield substantial benefits are described. Although a commercial aircraft was recommended based on a first order analysis in

Section 2.3.2, [Bonometti, 2006] indicated that this solution may not be optimal if considerable modifications are required. Alternatively, efforts should be focused on reducing ground crews, devising more efficient rocket manufacturing processes, eliminating rocket shipping costs, simplifying inspections along with promoting lower insurance premiums via multiple abort and recovery procedures. Also, due to the current low launch rate, approximately 60 per year as discussed in Section 1.1, continuous utilization of the carrier aircraft is critical. From a technical perspective, this same source proposed carrying the launch vehicle under the carrier aircraft, along its centreline. In turn, this is expected to simplify servicing and inspection, balance the carrier aircraft's flight dynamics and provide positive separation upon release. Also, it introduces the opportunity for the carrier aircraft to conduct unrelated tasks such as carrying outsized commercial cargo which promotes continuous utilization. In terms of systems engineering, selected subsystems should be offloaded to the carrier aircraft; examples include fire suppression, power supplies along with tracking and guidance units [Bonometti, 2006]. It should be noted that an air launch could potentially benefit from regional expertise if the launch is conducted in such an area as compared to a ground launch and this could minimize the business and technical risks. Table 4.3 outlines recommendations for a microsatellite launch system.

Table 4.3: Microsatellite launch system recommendations [Bille, 2003]

Recommendation	Description
Government and/or internal financing	Other financing sources are not practical given unproven market for investors
Rely on proven technologies	New technologies may lower costs but these must be balanced against associated risks
Maximize operability	Maximize operability even at expense of additional research and development
Lean organization	Lean, dedicated organization with a committed budget
Design to cost	Design to cost rather than maximizing performance and/or for mass production
Early failure safety net	Ability to survive an early failure (ex: Falcon 1)

In addition to the factors listed in Table 4.3, technical and business risks can be mitigated with proven technology, intact aborts along with demonstrated off-the-shelf components [Sarigul-Klijn, 2000]. The Orbital Express benefits from these factors, as noted in Section 2.1, since its development relied upon proven, off-the-shelf technology [Hughes, 1996]. These approaches along with others to minimize the cost of launch missions are addressed in the following section.

4.13.3 Low Cost Mission Approaches

The foregoing discussion focuses on orbital injection accuracy and mission philosophies. Although the cases herein assumed the vehicle was already built, these approaches could be applied in developing or integrating systems such as a launch pad in the case of a ground launch and/or the couplings to the carrier aircraft in the case of an air launch.

Orbital Injection Accuracy

Two low cost LEO injection alternatives are described to achieve relatively high pointing accuracies. The launch vehicle could purposely deliver the spacecraft into a higher orbit. In turn, the mission lifetime would be extended to fulfil the objectives and could potentially introduce a relatively low cost design margin increase. Although the additional propellant required to achieve the higher orbital altitude would lead to increased costs, in the event where challenges are experienced with the initial checkout, the probability of losing the mission to a malfunction which could not be remedied until large orbit decay would be diminished. Alternatively, the spacecraft could be inserted into a frozen orbit where the Earth's oblateness effects on the eccentricity and argument of perigee are eliminated and the spacecraft's orbital shape is preserved. Also, the spacecraft undergoes minimal variations in altitude. Examples of missions in a frozen orbit include RADARSAT, LANDSAT and SPOT [Aorpimai, 2003].

Mission Philosophies

Optimizing for minimum cost is proposed. In this case, the maximum performance and minimum mass design philosophy is set aside for one where the vehicle cost and mass are optimized while maintaining quality and reliability standards. Reducing the part count is another cost minimizing approach which incidentally also promotes reliability. A simplified design with less components are expected to promote improved manufacturing procedures such that costs can be reduced [Wertz (b), 1999]. Small launch vehicle costs can be decreased by competition, commonality and simplified systems. Another

alternative is to emphasize cost over performance which yields a higher design margin. Despite this approach's shortfall of higher mass, it could potentially deliver lower recurring costs. By promoting design margins rather than redundant systems, it is expected that both complexity and cost will decrease and a higher reliability will ensue. Assembly line produced launch vehicles, characterized by relatively low cost and redundancy along with automated launch pad operations have been demonstrated by the former Soviet Union countries [Weeks, 2004].

Another approach is to use commercial off-the-shelf products (COTS). Although the foregoing discussion focuses on spacecraft development, parallels can be made to launch systems. It has been shown that COTS devices can be successfully utilized for LEO microsatellite missions. According to [Underwood, 2000], selected benefits of COTS products include low cost, ready availability, low power and high packing density. Ultimately, the optimal approach would likely be a compromise of the above-described methodologies.

4.14 Assessment

An air launch clearly benefits from a strategic launch location and has the ability to be responsive. An air launch can be selected based on suitable weather, ascent over water trajectory clear of aircraft, ships and other launch systems at the latitude equal to the desired orbital inclination with appropriate phasing particularly in rendezvous missions to minimize electrical power requirements and maximize payload mass [Sarigul-Klijn,

2001]. Although an air launched alternative can be independent of a dedicated launch facility, limited ground infrastructure is still required. A ground launch's payload capability is not subject to the limitations by the carrier aircraft's capabilities given its size and mass constraints as is the case for an air launch. With regards to the vehicle design, given the smaller change in velocity needed to achieve an orbit in the case of an air launch, the effect of the inert mass fraction is mitigated which introduces design flexibility and/or added safety margins. Although an air launch can potentially reduce range and tracking fees, careful consideration must be given to the regulatory concerns of utilizing a foreign range along with the requirement for a restricted or unoccupied airspace. The significant ground launch heritage and know-how which currently exists in comparison to that of air launches suggest that an air launch is in fact more complex. An air launch benefits from simplified abort modes due, in part, to the relatively high launch altitude. However, the air launch must consider the safety of onboard crew, if any, but this could be avoided with uninhabited aircraft launches. In the case of a catastrophic failure, this study surmised that an air launch would likely necessitate less time to recover and return to normal operating conditions in situations where the significant infrastructure of the ground launch calls for extensive refurbishing. With regards to insurance costs, which account for a relatively significant expense for a launch campaign, although a typical ground launch will likely incur reduced insurance premiums due to the larger pool of know-how, the strategic launch location and simplified abort modes of an air launch could lead to lower insurance costs provided flight heritage is established. In terms of recurring and non-recurring costs, one of the most significant cost reductions for

the air launch is the fact that it has less expendable hardware and infrastructure. However, an air launch must account for the recurring aircraft and runway costs along with the development costs associated with the carrier aircraft's modifications.

In order to address the factors discussed above within the context of a Canadian micro class access to space, this study proposes to begin with a ground launched vehicle which can be converted to an air launch [Labib, 2004]. In fact, the air launch's operational benefits could complement the larger payload capability of a ground launched system. In such a case, the air launched vehicle would cater to lighter payloads which require a responsive service [Sarigul-Klijn, 2001].

4.15 Summary

Operational aspects of a ground and air launch have been addressed. These facets were wide ranging including responsiveness, launch infrastructure, air traffic, safety and cost considerations. Operational commonalities and differences of both launch systems were described and favourable circumstances for ground and air launch were discussed. A summary of these findings, along with the analytical results of Chapter 3 is provided in Chapter 5.

CHAPTER 5

CONCLUSIONS

The results of a comparative investigation of a ground and air launched vehicle have been described. The study addressed both analytical and operational aspects of the launch architectures. Analytical considerations were largely substantiated by ASTOS in addition to in-house numerical models and were based on the Orbital Express conceptual launcher, a joint Bristol and Microsat effort. The operational segment assessed key factors from integration to launch. Ultimately, this project identified favourable ground

and air launch circumstances while accounting for a multitude of technical and logistical factors.

5.1 Current State of Small and Micro Launcher Initiatives

Successful, low cost and effective missions have been demonstrated by small and microsatellites. However, the launch cost is consistently too high in comparison to the low cost of such satellites. These satellites have often been launched using the former Soviet Union ICBMs and relatively inexpensive Western launchers. Despite the current oversupply of launch capacity in the world, dedicated launch vehicles for small and microsatellites are very limited. In order to minimize launch costs, such satellites are often launched as a secondary payload to the detriment of mission flexibility, multi-payload integration risks, potential scheduling delays along with the need to find another customer to share the launch. As a result, there is a strong demand for a low cost dedicated launcher.

Since the current launch figures can easily be satisfied by the world's current fleet of launch vehicles, it will be difficult for the new small and micro launch vehicles to break into the world market in the near future. However, an important observation was the US military's clear shift towards small and microsatellites along with rapid launch. This market is also stimulated by an unmet demand for low cost launchers by universities mainly fuelled by the growing capability of small and microsatellites. Today, many countries are developing low cost small launch vehicles.

5.2 Ground and Air Launch Investigation

The Orbital Express was originally a four staged solid propellant-based launch vehicle. Within the scope of this study, the ground launched vehicle was not modified but the air launched system was comprised of the upper three stages. The air launched vehicle was assumed to be characterized by a 90° initial pitch angle from 11 km in altitude and -50 m/s initial flight path speed. The ground and air launched vehicles had a mass of 16,270 kg and 4730 kg with 55 kg and 35 kg payload capability respectively to a 600 km altitude and 60° inclination circular orbit from the Churchill launch range. In turn, this translated into a payload to take-off mass ratio of 0.34% and 0.74% respectively.

In order to validate the ASTOS results and gain insight into the commercial software, an in-house MATLAB model with the appropriate equations of motion was developed for both launch methods. The MATLAB ground and air launched models were within 4.59% and 9.18% respectively of the ASTOS results.

The analytical results indicated that at constant inclination and SSO, a ground launch is better suited for microsatellites up to approximately 55 kg. In addition, such a launch method was determined to be capable of accessing relatively low inclinations for which the air launch was limited. However, this is only applicable where an air launch's inherent strategic launch location is not incorporated as was the case herein for comparative purposes along with the operational limitations of an air launch discussed below. Otherwise, by capitalizing on this feature, an air launch would typically yield higher

payload capabilities and access to lower altitudes and inclinations. Based on this study's assumptions, an air launched system may be warranted for payloads up to approximately 40 kg, predominantly for low altitudes. An air launch was increasingly favourable if the given mission required a service which promoted responsiveness. In this case, schedule, weather and transportation challenges could be mitigated by the air launch's operational flexibility. It should be noted that although an air launch can be independent of a dedicated launch facility, practical considerations typically require a launch base. This relatively limited infrastructure can be portable but can potentially limit the degree of a strategic launch. The results at constant altitude indicated that accessing low inclinations with higher altitude became more challenging for both launch architectures. The ground launch was typically advantageous, most notably as the altitude increased, while an air launch was more appropriate for low inclinations with higher altitude. In fact, similarly to the above description, despite its lower payload capability, an air launch may occasionally be preferred, particularly for inclinations above the corresponding launch latitude, since it is conducive to responsiveness. In regards to the initial launch velocity of an air launch, the results revealed that the added complexity of attempting to provide a higher initial velocity to the launcher outweighed the added payload mass in the case of a 90° launch pitch angle. Also, significant payload, safety and flexibility improvements could be realized with one or more hybrid engines given their throttling capability. This was mainly noted for the ground launched case; although similar benefits could be delivered for an air launched vehicle, the higher volume of such systems compared to SRMs introduce challenges given the carrier aircraft's geometrical restrictions.

An air launch avoids the large infrastructure and recurring expenses surrounding a ground launch campaign but this launch architecture must account for the development costs associated with the carrier aircraft's modifications along with the maintenance costs. This study surmised that these factors coupled to the additional ground launch know-how will lead to lower costs in the ground launched case compared to its air launched counterpart. However, if a stronger air launch heritage is developed, lower costs could be realized. Although a typical ground launch will likely incur reduced insurance premiums given its stronger flight heritage, the simplified abort modes of an air launch could decrease insurance costs provided a sufficient successful flight record is established. Also, despite the fact that an air launch can potentially reduce range and tracking fees, careful consideration must be given to the regulatory concerns of utilizing a foreign range along with the requirement for a restricted or unoccupied airspace. In the case of a catastrophic failure, this study deduced that an air launch would likely require less time to recover and return to normal operating conditions in cases where the significant infrastructure of a ground launch requires extensive refurbishing.

Within the scope of a Canadian micro class access to space development program, an air launch's operational benefits could complement the higher payload capabilities offered by a ground launch [Sarigul-Klijn, 2001]. Hence, this study proposes the operation of a ground launched vehicle followed by an air launched system as the project matures, once an adequate flight heritage and a market have been established.

5.3 Summary of the Major Findings and Contributions

A concise summary of the major findings and contributions from this report are listed below:

- The need exists for a dedicated small and micro launch vehicle.
- A ground and air launched vehicle have been proposed based on the Orbital Express.
- The study recommended that the air launched vehicle be mounted under a carrier aircraft with a 90° initial pitch angle and an aft crossing trajectory.
- First order investigations indicated that the CRJ705 would be a suitable carrier aircraft for an air launch at 11 km altitude and -50 m/s initial flight path speed.
- MATLAB models were successfully created to validate the results and gain insight into the ASTOS commercial software.
- The initial mass and payload capability of the ground and air launched vehicles to 600 km circular altitude and 60° inclination were 16,270 kg and 4730 kg along with 55 kg and 35 kg respectively. These correspond to a payload to take-off mass ratio of 0.34% and 0.74% for the ground and air launched vehicles respectively.
- A hybrid motor would significantly improve the payload capability and mission flexibility of both launch architectures, particularly the ground launched system.
- Favourable ground and air launch scenarios from both analytical and operational perspectives were identified.
- In developing a Canadian micro class access to space, a ground launched vehicle was proposed followed by the introduction of an air launched system.

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