CHAPTER 1

INTRODUCTION

In recent years, the space sector has seen a significant shift in the paradigm of space launch system design. The sector has moved towards privatisation, with new and innovative launch systems competing to offer the most cost-efficient and reliable launches. The sector has also seen a split between those who produce large satellite launchers and those who produce small satellite launchers. For large payload launchers, reusability is a major focus in the design of new launch systems, with the purpose of making a launch system cost efficient over multiple launches[5]. For small payload launchers, reusability is more complex than for large launchers, as the additional systems necessary for reusability add a larger fraction of system mass, and require a proportionally larger fuel mass. Consequently, the focus of small launch system design is currently on producing expendable launch systems as cheaply and efficiently as possible, using state of the art technologies such as 3D printing to expedite the process and minimise cost[6]. However, if reusability is able to be successfully integrated into small launch system design, it has the potential to increase the cost efficiency and launch flexibility, potentially opening up the small satellite market significantly.

A potential candidate for integrating reusability into small satellite launch systems is the use of airbreathing engines[7, 8]. Airbreathing engines produce higher specific impulse than rockets, and do not require oxidiser to be carried on-board a launch vehicle[9]. The higher efficiency and reduced propellant mass of airbreathing vehicles allows the additional mass of the systems necessary for reusability to be mitigated[10]. An airbreathing vehicle can be designed in a similar fashion to a conventional aircraft, with wings, stabilisers and ailerons[11, 12]. A vehicle designed in this fashion has a high lift-to-drag ratio, and good manoeuvrability, allowing for a return flight and landing on a conventional landing strip[12]. This style of return removes the need for transport, enabling a fast turn-around and cost-efficient re-use.

The primary airbreathing engines in consideration for launch vehicles are ramjet and scramjet engines, and rocket-based combined cycle engines that combine multiple engine cycles for operation over a wider range of flight conditions[13, 14]. These engines offer good efficiency and have operational regimes that allow them to effectively accelerate a launch vehicle over a range of Mach numbers. Ramjets and scramjets rely on the high speed of the aircraft to compress the flow of air entering the engine before combustion. Ramjets slow the air to subsonic speeds before combustion and are limited to operation at low Mach numbers, whereas scramjets keep the flow supersonic throughout, and operate within the hypersonic regime, above Mach 5. These engines have limited operational regimes, and require atmospheric flight in order to take oxidiser from the air. These operational constraints mean that a launch system cannot be solely powered by airbreathing engines. Rocket power is necessary for at least the exoatmospheric portion of the trajectory. As a result, the designs of airbreathing launch systems require rocket stages at reach orbit[7], and if high-speed airbreathing engines are used in the launch system, rocket power is also desirable for accelerating scramjet accelerator to minimum operational speed, as the alternative is using different types of lower-speed airbreathing engines sequentially[7], which is weight and cost intensive.

These various propulsion systems may all be integrated into a single stage-to-orbit spaceplane that is capable of launching, placing payload in orbit, reentering, and returning to a suitable landing site[15–22]. Alternatively, they may be separated into multiple stages, similarly to typical, fully rocket-based launch systems[23–32]. Single stage-to-orbit launch vehicles are fully-reusable, and have the potential to be extremely cost-efficient, if they are able to be reused for many missions[33]. However, their development cost is likely to be very high, and they are generally suited for launching large amounts of payload-to-orbit[15–22], a market which is now extremely competitive thanks to the relatively recent advent of large, partially reusable, rocket-based launch systems. Multi-stage airbreathing launch systems, however, have the potential to bring cost-efficient reusability to small payload launchers[12], particularly if they are able to stage at high speeds[33]. Airbreathing systems scale more efficiently than rockets, meaning that the systems needed for reusability in a small launcher do not require the launcher system to be dramatically increased in size, one of the primary limitations on reusability in small rocket-based launch systems. For this reason, partially-reusable two and three-stage airbreathing launch systems are being investigated for small satellite launch[12]. To date, two-stage launch systems have primarily been studied over three-stage launch systems, however, three stage launchers have recently been espoused as potentially offering advantages in reusable mass and mass-to-orbit efficiency[12].

There are as of yet no airbreathing launch systems that have progressed past the research and very early design stages[15–32]. The development and analysis of trajectories is a crucial part of this early design process, and is a complex task for an airbreathing launch system if the efficiency of the launch system is to be maximised. A trajectory must be calculated that allows the launch system to achieve its objective of placing the maximum payload into orbit, while recovering any reusable stages and adhering to the structural, heating and propulsive limitations of the vehicle[34]. In order to maximise the efficiency of the launch system, and thus the payload-to-orbit, there are complex trade-offs in the performance of the launch system that must be taken into account: the airbreathing engines of a ramjet or scramjet-powered stage require high dynamic pressure to operate effectively, and airbreathing stages are generally designed for high lift-to-drag. Conversely, rocket-powered stages operate more efficiently at higher altitude, and are generally designed for weight and cost efficiency. For airbreathing launch systems, the various distinct operation modes and phases involved during launch require trade-offs in engine efficiency and thrust generation, stage mass, and vehicle aerodynamics. These factors require the launch trajectory of the system to be thoroughly simulated and optimised, to ensure that the launch vehicle is operating effectively.

For single-stage-to-orbit airbreathing launch systems, trajectories have been developed and studied in detail[15–22], in particular due to a multitude of studies in the 1980s and 90s, the most prominent of which was the National Aerospace Plane (NASP), which led, in part, to the Hyper-X program and the X-43 flight experiment[33]. These studies show complex trade-offs that occur in a maximum efficiency airbreathing vehicle trajectory, including trade-offs between the operational efficiencies of the various engine modes. These trajectories often show flight at maximum dynamic pressure or maximum thermal loading, to maximise the efficiency of there airbreathing engines, before a pull-up to orbit, sometimes initially under airbreathing power. These trajectory analyses generally assume that return is possible, due to the high speed and manoeuvrability of the vehicles allowing flexible reentry, and their on-board low-speed engines allowing propulsion and cruise in-atmosphere.

Compared to single-stage airbreathing systems, multi-stage airbreathing launch system trajectories are far more complex, with even more complex trade-offs between the operational modes due to these modes being separated into mechanically distinct stages[34]. The trajectories of two-stage-to-orbit airbreathing launch systems that have been studied generally exhibit constant dynamic pressure flight to maximise the operation of the airbreathing engines, and a pull-up at the end of the airbreathing trajectory. The return trajectories of two-stage-to-orbit launch systems are generally performed under cruise power, utilising turbojet, or sometimes ramjet engines[23, 25, 29, 30], or land at a point downrange[26]. There is considerable disparity between studies over whether constant dynamic pressure flight is optimal, and if a pull-up under airbreathing power is optimal[23–32]. No studies have attempted a detailed investigation into the trade-offs between the stages, and the optimal trajectory is not well understood. For three-stage-to-orbit launch systems the launch trajectory is more complex again, due to there being two separation points at which to consider trade-offs, and each stage generally only having a single engine type. Three-stage-to-orbit systems are by far the least studied of the airbreathing launcher configurations, and have to this point only been designed around prescribed trajectory shapes that constrain the hypersonic airbreathing stages to constant dynamic pressure[31, 32]. It is currently unknown what the optimal trajectory shape for a three-stage launcher is, and if it is possible to return the second stage of a launch system of this type to its initial launch location. The presence of a first-second stage separation, high second-third stage separation speeds, lack of low speed engines on the second stage, and small third stage rocket make the calculation of an optimal trajectory shape for a three-stage airbreathing launcher a unique problem, with extremely complex trade-offs. The study of two-stage-to-orbit trajectories is somewhat useful in this regard, however, the disparities between the various studies that have been conducted, and the lack of clear studies on the trade-offs between the performance of the launch system stages makes the development of an maximum efficiency trajectory solely from previous studies untenable.

Figure 1.1: The SPARTAN scramjet-powered accelerator[35].

This work aims to expand our knowledge on the operation of airbreathing launch systems, by developing a trajectory for a partially-reusable rocket-scramjet-rocket small satellite launch system, of a design based on the SPARTAN launch system under development by The University of Queensland and Hypersonix[12]. This trajectory is developed using modern and robust optimal control theory techniques, that are able to calculate the optimised trajectory profile for a launch vehicle in a robust and computationally efficient manner, allowing a trajectory to be calculated in which the flight path of each individual stage is considered simultaneously to produce a maximum-payload trajectory[36]. Optimal control is able to produce an optimised trajectory that satisfies the specific structural and flight constraints of the vehicle being simulated, allowing the physical limitations of the vehicle, such as heating and structural loading limits, to be imposed[36]. This optimal trajectory profile is investigated to give insights into the trade-offs between the rocket and airbreathing stages that are unique to this type of launch system. This manner of optimal trajectory analysis allows for generalities to be made about the nature of the trajectory shape, and for the understanding of the performance of this type of launch system to be improved. This trajectory analysis is intended to both aid in the ongoing design process of multi-stage launchers, by characterising the performance needs of a rocket-scramjet-rocket launch system throughout its trajectory, as well as to stand on its own merit by indicating the best possible trajectory shape for this type of launch system.

CHAPTER 8

CONCLUSIONS

The purpose of this work was to design and investigate the optimal launch trajectory of a partially-reusable, rocket-scramjet-rocket, small satellite launch system. The trajectory of this launch system was optimised for maximum payload-to-orbit, and characterised in order to determine the key performance parameters of the launch system. This aim was achieved through the completion of the set of objectives detailed as follows:

Development of a detailed design and aerodynamic simulation for a rocket-scramjet-rocket launch system.

In order to create a representative model for a trajectory simulation, the design of a rocket-scramjet-rocket launch system was developed. This launch system was designed around the SPARTAN scramjet-powered accelerator, which is in development by The University of Queensland and Hypersonix. A first stage rocket was designed, to accelerate the scramjet accelerator to its minimum operating speed of Mach 5. This first stage was based upon the Falcon-1e, scaled down lengthwise to 9.5m. A third stage rocket was designed, based around the Kestrel upper stage rocket motor for cost effectiveness. This third stage was sized to fit within the fuselage of the scramjet accelerator, to be 8.7m long, and 1.1m wide. The heat shield necessary for atmospheric flight, and the internal fuel tanks of the third stage were sized to a total mass of 3300kg. The fuel tanks of the scramjet accelerator were resized, to accommodate this redesigned third stage.

The aerodynamics of the launch system were calculated using Cart3D, an inviscid CFD package, and modified using a viscous correction for accuracy. The aerodynamics of the launch system were calculated across the operable regimes of the vehicles, which for the scramjet accelerator included both engine-on and engine-off conditions, across a range of Mach numbers from 0.2 to 10. The control surfaces of the scramjet accelerator were modelled, and the aerodynamics of the scramjet accelerator simulated with flaps deployed. A variable centre of gravity model was created for the launch system, to model the changes in the vehicle dynamics during flight. The aerodynamics of the scramjet accelerator were calculated at multiple centre of gravity positions, and a trimmed aerodynamic database was created.

Calculation of the maximum payload-to-orbit trajectory for a rocket-scramjet-rocket launch system using optimal control, with and without fly-back.

This study represents the first time that an optimal launch trajectory has been calculated for an airbreathing three-stage launch system. In order to calculate the maximum payload-to-orbit trajectory of the launch system, a software package was created to simulate and optimise launch system trajectories, designated LODESTAR. LODESTAR utilises GPOPS-2, a pseudospectral method optimal control solver, and simulates the trajectory of each stage of the launch system in a geodetic rotational reference frame. LODESTAR optimises the entire trajectory of the launch system simultaneously, so that the performance trade-offs between the stages are captured accurately.

A mission profile has been developed for the representative rocket-scramjet-rocket launch system for which to calculate this optimal trajectory, launching a satellite to sun synchronous orbit from the Northern Territory, Australia. Initially, the trajectory of the launch system was developed with the assumption that the scramjet accelerator lands at a location downrange. A mission case was developed in which the scramjet accelerator stage of the launch vehicle was constrained to flight at its maximum dynamic pressure. This is a common assumption for the maximum efficiency trajectory for airbreathing launch systems; because it maximises the thrust of the airbreathing stage, and as such it provides a useful a baseline trajectory case for comparison. This constant dynamic pressure trajectory was found to be capable of delivering 98.3kg to sun synchronous orbit. The maximum payload-to-orbit trajectory of the launch system was then calculated. It was found that, when flying the payload-optimised trajectory, the launch system is capable of delivering 156.4kg of payload to sun synchronous orbit, an increase of 59.1% over the simulation with the scramjet accelerator constrained to constant dynamic pressure. Three key features were observed in the trajectory; a higher first stage-scramjet accelerator separation point, an altitude raising manoeuvre in the centre of the scramjet accelerator's trajectory, and a pull-up before scramjet accelerator-third stage separation. The altitude raising manoeuvre in the centre of the scramjet accelerator's trajectory was observed occur in a region of homogeneity in the performance of the scramjet accelerator, increasing the efficiency of the scramjet accelerator by +0.3%. The improvement in payload-to-orbit was found to result primarily from the stage separation conditions, as a consequence of favourable trade-offs between the efficiencies of the stages of the launch system. The higher first stage-scramjet accelerator separation point was found to decrease the amount of turning which the first stage must perform, reducing the necessary throttling, and increasing the efficiency of the first stage. Similarly, a pull-up before the scramjet accelerator-third stage separation decreases the amount of turning which the third stage must perform, and enables the third stage to gain altitude much more rapidly, causing it to spend significantly less flight time at high dynamic pressure. This reduced exposure to high dynamic pressure reduces the energy losses due to the aerodynamic drag of the third stage, as well as reducing the amount of energy imparted upon the heat shield, by allowing it to be jettisoned earlier. The altitude increasing manoeuvres at the stage separations were found to result in the exergy efficiency of the scramjet accelerator decreasing by -0.719%η (-14.04%). However, this reduction in the efficiency of the scramjet accelerator is a trade-off for increases in the exergy efficiencies of the first and third stages, of +0.623%η (+9.96%) and +6.013%η (+63.20%) respectively, resulting in a significantly higher overall efficiency.

The mission definition was adjusted, to include a constraint of the scramjet accelerator flying back to the initial launch site after the separation of the third stage. The optimised maximum payload-to-orbit trajectory profile was calculated, and it was found that the launch system is capable of delivering 132.1kg of payload to sun synchronous orbit, while returning the scramjet accelerator to the initial launch site. This result shows that it is feasible to return a scramjet-powered accelerator to its initial launch site, with only a -24.3kg (-15.5%) reduction in the payload mass-to-orbit. The inclusion of the fly-back of the scramjet accelerator was found to alter the shape of the ascent trajectory significantly. When the fly-back was included, the first stage was found to initially pitch towards the east. The scramjet accelerator was then observed to bank heavily, executing a heading angle change manoeuvre during its acceleration. No altitude raising manoeuvre was observed during this banking acceleration, due to the higher angles of attack while banking resulting in flight at the scramjet accelerator's maximum dynamic pressure being optimal. When the fly-back was included, the scramjet accelerator was still observed to perform a pull-up manoeuvre before third stage separation, of a similar magnitude to the pull-up manoeuvre performed with no fly-back. The optimal fly-back of the scramjet accelerator was found to require the ignition of the scramjet engines, and was observed to exhibit three distinct phases, an initial turn, a boost-skip, and an approach. During the initial turn, the bank angle of the scramjet accelerator is increased rapidly, in order to manoeuvre the heading angle of the scramjet accelerator back towards its initial launch site. After this initial turn, the boost-skip phase is initiated, consisting of multiple skipping manoeuvres. These skipping manoeuvres serve both to increase the range of the scramjet accelerator during its return, minimising the fuel necessary for the fly-back, as well as to improve the specific impulse of the scramjet engines. The scramjet engines were observed to be ignited at the trough of each skip, as soon as the scramjet accelerator accelerates above the minimum operable Mach number of the C-REST engines. At this point of the skipping manoeuvres, the specific impulse of the scramjet engines is highest, so that igniting the scramjet engines at this point minimises the fuel necessary for the return flight. After the scramjets were ignited a total of three times, three unpowered skips were performed, decreasing in size sequentially. Finally, the skips ceased entirely, beginning a steady descent and approach to the landing site. In total, 257.8kg of fuel was used during the fly-back, 16.5% of the scramjet accelerator's total fuel mass.

These maximum payload-to-orbit trajectory profiles, that have been calculated using LODESTAR, are non-intuitive, and involve complex trade-offs between the efficiencies of each stage of the launch system, as well as the fly-back of the scramjet accelerator. The design of these optimised flight paths is made possible through the use of the pseudospectral method of optimal control, coupled with accurate propulsion and aerodynamic modelling. These trajectory profiles improve the performance of the launch system, and can assist in directing future design decisions for partially-airbreathing, multi-stage launch systems. Particularly of interest is the optimal pull-up of the scramjet accelerator, before the release of the third stage. The pull-up, as well as directly increasing payload-to-orbit, significantly lowers the dynamic pressures experienced by the third stage rocket, an important factor when designing the thermal protection and structure of the third stage. It is likely that the mass of the heat shielding of the third stage will be able to be reduced significantly due to the reduced thermal and structural loading, and that the control of the third stage will be made more simple due to the decreased aerodynamic forces and moments experienced. In addition to the significant altitude-raising manoeuvre at the first stage-scramjet accelerator separation, these deviations from maximum dynamic pressure indicate that it is imperative to design the scramjet engines of a three-stage launch system to be operable at low dynamic pressure, and at high angle of attack. The multiple restarts observed during the return trajectory also indicate the importance of being able to restart the scramjet engines rapidly.

Analysis of the sensitivity of the maximum payload-to-orbit trajectory to variations in key design parameters of the launch system.

Eight key design parameters of the launch system were modified, and the sensitivities of the maximum payload-to-orbit trajectory were studied. The parameters varied were: the maximum dynamic pressure of the scramjet accelerator, the fuel mass within the scramjet accelerator, the drag of the scramjet accelerator , the specific impulse of the scramjet accelerator, the mass of the scramjet accelerator, the drag of the third stage, the specific impulse of the third stage, and the mass of the third stage. These parameters were varied for trajectories both with, and without, scramjet accelerator fly-back. When fly-back was included, the sensitivity to most design parameters was observed to decrease, due to the fly-back distance counteracting some of the effects of the design variations. However, the sensitivity to maximum dynamic pressure was observed to significantly increase, as the complex coupled nature of the ascent and fly-back meant that the maximum dynamic pressure had a more significant effect on the overall trajectory.

It was found that the ability of the first stage to pitch, determined by the acceleration of the launch system, is the primary driver of the first stage-scramjet accelerator separation conditions. The first stage-scramjet accelerator separation altitude was observed to increase, and the separation angle was observed to decrease, when the acceleration during first stage operation was increased, due to the better pitching ability of the first stage after accelerating more rapidly. When the efficiency of the scramjet accelerator was increased, the efficiency of the third stage was also observed to increase, and vice versa. This increased efficiency trend was due to the increased speed at the scramjet accelerator-third stage separation point, which improves the propulsive efficiency of the third stage rocket. The exception to this trend was found when the fuel mass of either stage was varied, resulting in a larger mass, and decreased efficiency, but a higher total exergy. Variations in the efficiency of the third stage past a certain point were found to produce no significant variation in the trajectory of the first stage or scramjet accelerator, but had by far the highest relative influence on the payload-to-orbit, in part due to the particular importance of the specific impulse during the Hohmann transfer. However, at low specific impulses, the third stage was observed to have difficulty exiting the atmosphere, requiring a very low and level release point in order to use its aerodynamics effectively. This trend was also observed in the third stage mass sensitivity, where payload-to-orbit was observed to increase until a distinct cutoff point as the mass of the third stage was increased, after which it decreased drastically, because the third stage once again experienced difficulties exiting the atmosphere. This indicates that achieving a suitable thrust-to-mass ratio for atmospheric exit after release from the end of a pull-up manoeuvre is crucial for efficient third stage operation.

The sensitivities of all significantly coupled design parameters were compared, and their relative quantities assessed to provide insights for future launch system designs. Of these comparisons, the relationship between the maximum dynamic pressure and the structural mass of the scramjet accelerator was found to be of particular interest. It was found that the sensitivity of the launch system to the maximum dynamic pressure of the scramjet accelerator is relatively low, indicating that it may be advantageous to design the scramjet accelerator to fly at a lower maximum dynamic pressure, in order to reduce heat shielding and structural mass. It was found that if the mass of the scramjet accelerator can be reduced by greater than -68.6kg per -1kPa reduction in maximum dynamic pressure (or -106.7kg per 1kPa when fly-back is included) then a larger payload-to-orbit will be achieved.

This investigation into the sensitivity of the optimised trajectory to variations in the design parameters of the launch system has provided insights into the shape of the optimised trajectory, and allowed the effects of the modified design parameters to be quantified. These findings can be used to predict the maximum payload-to-orbit trajectories of future launch systems, as well as how design changes may affect the performance of the launch system utilised in this study.