A Concept for Operationally Responsive Space Mission Planning Using Aeroassisted Orbital Transfer

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Abstract

A concept for optimal mission planning of space assets for operationally responsive space is presented. The concept is based on the idea of transferring a small maneuverable spacecraft between two low-Earth orbits using an aeroassisted orbital transfer. The mission design is described schematically and representative initial and terminal orbits are given for which to execute the mission plan. Some high-level results are then shown along with the fuel requirements for these maneuvers. The aeroassisted maneuvers are compared qualitatively with typical all-propulsive maneuvers and it is shown that the aeroassisted concept produces a great savings in fuel over an all-propulsive maneuver. The results presented here show that the concept of using an aeroassisted orbital transfer is an excellent option for operationally responsive space.

1 Introduction

A growing area of current and future interest to the U.S. space community is the problem of *Operationally Responsive Space* (ORS) [1]. ORS refers to the ability to enhance capability, increase flexibility, and reduce execution time of operational spacecraft [1]. The problem of ORS and its importance is summarized eloquently in the following quote from U.S. Air Force Major Kendall K. Brown [1]:

> An operationally responsive space system could be an integral part of national defense by providing operational capabilities, flexibility, and responsiveness that does not exist today. Current space assets provide communication, navigation, and intelligence, surveillance, and reconnaissance (ISR) capabilities using satellites designed for long life and high reliability. Those life and reliability requirements are due in part to the high cost and limited availability of space launch. Current space systems require years to develop due to the complicated specialized design and manufacturing processes. The high cost of launching space assets, and competition with the commercial launch market, require launch scheduling years in advance. Moreover, once it has been scheduled on a launch vehicle, it may take several months to checkout and integrate into the launch vehicle, and several additional months to become operational once it is in space. This existing capability is not operationally responsive. An

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operationally responsive space system needs the capability to transport space assets to, through, and from space. The responsive satellites need operational capability immediately upon deployment for contingency constellation sustainment or augmentation.

In addition to being a problem of interest to the military, ORS is a potentially important technology for civilian applications. In particular, unmanned orbiting vehicles may need to be repositioned rapidly in order to quickly predict or track dangerous natural phenomena (e.g., hurricane or tsunamis) while human space flight missions (e.g., the using the new NASA Crew Exploration Vehicle) may need to be rapidly aborted in order to save the crew in case of a catastrophic failure. Because a key goal of ORS is to rapidly reposition space assets, a necessity for ORS is the development of spacecraft that can accomplish multiple distinct missions. In order for multiple-mission capability to become a reality, it is necessary to develop approaches to rapidly design or re-design space missions.

A key component of mission design is *trajectory planning* (equivalently, *path planning* or *guidance*). Trajectory planning for space missions is essentially equivalent to determining the trajectories and controls that transfer a spacecraft between two orbits (i.e., orbital transfer). Due to the high cost of resources (e.g., fuel) or constraints on the duration of time over which a mission must be accomplished, orbital transfer is generally planned in an *optimal* or *near-optimal* manner. Designs for orbital transfer generally fall into one of two major categories: *all-propulsive* transfers (where the orbit is changed completely using on-board fuel) or transfers that combine propulsive maneuvers with *atmospheric flight* maneuvers (where a portion of the orbital transfer is accomplished using propulsion while the remainder is accomplished using aerodynamic control via flight through the atmosphere). The latter category of orbital transfer is called *aeroassisted orbital transfer*. In the case of small satellites, the on-board fuel constraints will render all-propulsive maneuvers infeasible for many missions, thereby requiring the use of atmospheric flight maneuvers. Because aeroassisted orbital transfer has long been known as a way to decrease the amount of fuel consumption, it is important to study the problem of aeroassisted orbital transfer for ORS.

Optimal aeroassisted orbital transfer for high-mass lifting bodies has been studied extensively over the past several decades. Much of the early work used analytical methods and is summarized in the survey paper of Refs. [2] and [3]. More recently, optimal aeroassisted orbital transfer has been considered for vehicles subject to high heating rate constraints [4, 5, 6, 7]. In addition, multiple-pass aeroassisted orbital transfer from geostationary orbit to low earth orbit with desired inclination change, subject to heating rate constraints, has been studied in Ref. [8] and [9]. In these studies, several types of aeroassisted maneuvers, such as aerocruise with propulsive maneuvers and aeroglide without propulsive maneuvers, have been discussed. It has been found that the heating rate constraint is one of the key parameters in determining the performance of the aeroassisted orbital transfer (i.e., the sustainable heating rate directly affects the amount of inclination change that can be achieved by the aeroassisted maneuver) and the overall mission cost (i.e., the amount of fuel required for the mission).

In this paper, we present a concept for mission planning for ORS. The mission planning problem is conceived as an aeroassisted orbital transfer problem between two strategic low-Earth orbits (LEOs) with a potentially large inclination change using a small, highly maneuverable, orbiting vehicle. In addition, the vehicle is subject to constraints on acceleration load and heating rate. The orbiting vehicle is chosen to be of a size that can be launched on a modern day small launch vehicle (e.g., Falcon or Minotaur). The aeroassisted orbital transfer problem is posed as a three-phase optimal control problem and is solved numerically using a multiple-phase version of the *Gauss pseudospectral method* [11]. The trajectories obtained in this study provide insight into the structure of the optimal mission plans and show that the aeroassisted transfers consume significantly less fuel as compared to an optimal all-propulsive transfer. The approach developed in this research shows the viability of the proposed mission plans for rapid mission planning for operationally responsive space.

2 Motivation for ORS

The current paradigm for space mission planning is extremely costly and time-consuming. For example, mission planning for a U.S. Space Shuttle launch takes years and, without any delays, the day-of-launch cost is $500 million. While other uninhabited spacecraft are less costly than the shuttle, these other missions still cost up to $100 million to launch. The prohibitive cost of mission planning makes it important to consider less costly alternatives. In particular, launching smaller spacecraft with less costly launch vehicles (e.g., small launch vehicles with solid fuel as opposed to larger launch vehicles with liquid fuel) can significantly reduce the individual mission cost. In this context, ORS will have a great impact on the availability of space access to organizations and countries who would otherwise lack the funding. In addition, the ability to launch on short notice can aid in being able to perform rapid intelligence, surveillance, and reconnaissance or analyze acute weather phenomena such as tropical storms and hurricanes.

3 Small Spacecraft Aeroassisted Orbital Transfer Problem

In this research we consider the following orbital transfer problem. A small spacecraft (of mass approximately 1720 kg) is initially placed into a low-Earth orbit (LEO). Furthermore, the initial mass of the vehicle, denoted m_0, is the sum of the dry mass, m_{dry}, and the propellant mass, m_{fuel}, i.e.,

$$m_0 = m_{\text{dry}} + m_{\text{fuel}} \tag{1}$$

The initial mass is assumed to be small enough that it can be delivered to the initial orbit using a *small* launch vehicle (SLV), where the SLV is similar to a vehicle such as a Falcon or a Minotaur. The objective is to transfer the spacecraft to a terminal orbit such that the propellant consumed during the transfer is minimized. Because we will consider problems with significant changes in inclination, in this study we focus on problems that use atmospheric flight maneuvers (i.e., *aeroassisted* maneuvers) to change the orbit of the vehicle. Thus, the vehicle considered in this research must be aerodynamically maneuverable in hypersonic flight.

4 Trajectory Design

In this study we consider the flight of an orbital transfer vehicle assuming a point mass model in motion over a spherical nonrotating Earth. In addition, we assume all thrust is impulsive, i.e.,

$$\Delta V = g_0 I_{sp} \exp(m^+/m^-) \tag{2}$$

where ΔV is the impulse, g_0 is the sea level gravity, I_{sp} is the engine specific impulse, and m^+ and m^- represent the mass of the vehicle just before and after the application of the impulse. The flight of the vehicle is divided into three phases: (1) deorbit coast, (2) atmospheric glide, and (3) reorbit coast. The deorbit phase consists of an initial impulse, ΔV_1, along with an exo-atmospheric flight segment. The deorbit phase terminates at the edge of the measurable atmosphere (i.e., the transfer orbit attains an altitude h_{atm} where the atmosphere is first sensible by on-board instrumentation). The atmospheric glide begins upon atmospheric entry and terminates at atmospheric exit (i.e., when the vehicle re-attains the altitude h_{atm}). The reorbit phase consists of a second impulsive maneuver, ΔV_2, along with a second exo-atmospheric flight segment. This second exo-atmospheric flight segment terminates at the apogee of the terminal orbit. Finally, upon reaching apogee, a third impulse, ΔV_3, is applied to place the vehicle into the final circular orbit. A schematic of the trajectory event sequence is shown in Fig. 1. [][]Phase 1[][]Phase 1

In all phases of flight, the differential equations of motion for the vehicle are given in spherical coordinates as [12]

$$\begin{aligned}
\dot{r} &= v \sin\gamma \\
\dot{\phi} &= \frac{v\cos\gamma\cos\psi}{r\cos\theta} \\
\dot{\theta} &= \frac{v\cos\gamma\sin\psi}{r} \\
\dot{v} &= -D - g\sin\gamma \\
\dot{\gamma} &= \frac{1}{v}\left[-L\cos\sigma - \left(g - \frac{v^2}{r}\right)\cos\gamma\right] \\
\dot{\psi} &= \frac{1}{v}\left[-\frac{L\sin\sigma}{\cos\gamma} - \frac{v^2}{r}\cos\gamma\cos\psi\tan\phi\right]
\end{aligned} \tag{3}$$

where r is the geocentric radius, ϕ is the latitude, θ is the longitude, v is the speed, γ is the flight path angle, ψ is the heading angle, σ is the bank angle, $g = \mu/r^2$ is the gravitational acceleration, μ is the Earth gravitational parameter, D is the drag acceleration, and L is the lift acceleration. The drag and lift accelerations are given, respectively, as

$$\begin{aligned}
D = qSC_D/m \\
L = qSC_L/m
\end{aligned} \tag{4}$$

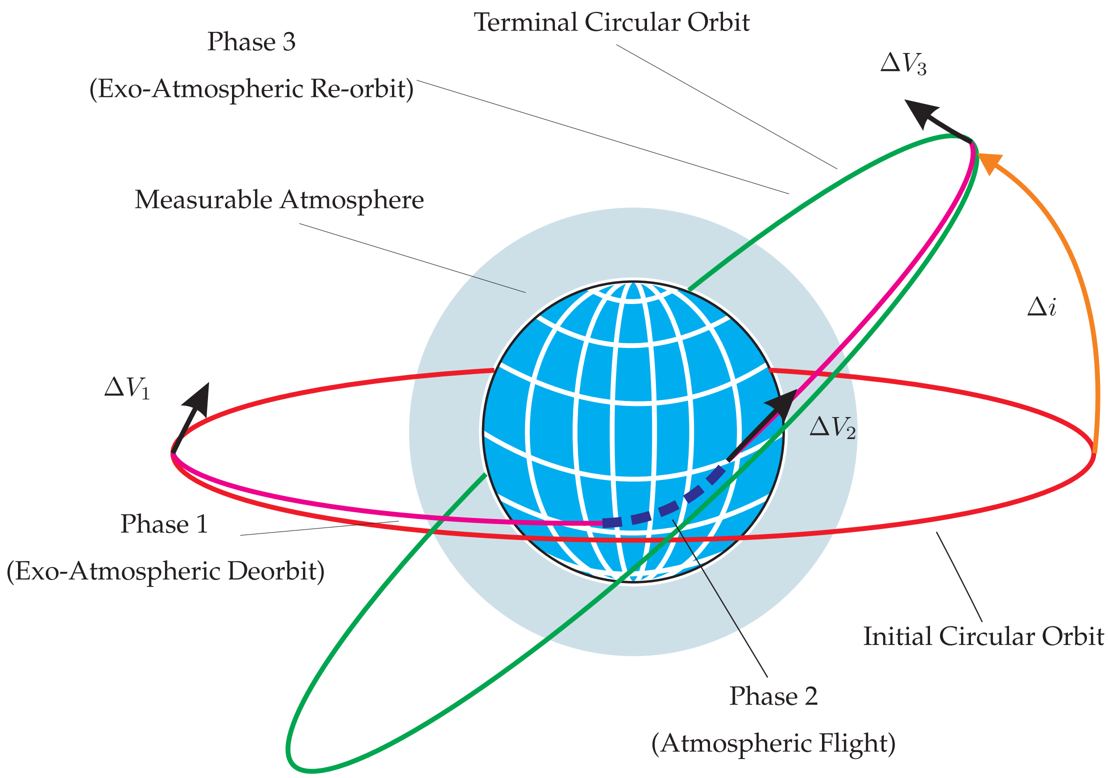

Figure 1: Schematic of trajectory design for LEO to LEO small satellite aeroassisted orbital transfer problem.

where $q = \rho v^2/2$ is the dynamic pressure and ρ is the atmospheric density. The density is computed as

$$\rho = \rho_0 \exp(-h/H) \tag{5}$$

where ρ_0 is the sea level density, $h = r - R_e$ is the altitude, R_e is the radius of the Earth, and H is the density scale height. The aerodynamic model used in this study is a drag polar of the form

$$\begin{aligned} C_D &= C_{D0} + K{C_L}^2 \\ C_L &= C_{L,\alpha}\alpha \end{aligned} \tag{6}$$

where α is the angle of attack, C_D and C_L are the coefficients of lift and drag, respectively, C_{D0} is the zero-lift drag coefficient, $C_{L,\alpha}$ is the lift slope, and K is the drag polar parameter. Finally, during atmospheric flight the vehicle is subject to the heating rate constraint [13]

$$\dot{Q} = \dot{\bar{Q}}(v/v_e)^{3.15}\sqrt{\rho/\rho_0} \tag{7}$$

where $\dot{\bar{Q}}$ is a constant. The aerodynamic and physical constants used in this study are shown in Table 1 and are taken from Ref. [7] (with the exception of the mass values, which, as mentioned, represent a vehicle that is capable of being placed into LEO via a small launch vehicle such as a Falcon or Minotaur). Finally, it is noted, that during *exo-atmospheric flight*, the drag and lift are omitted from the dynamics of Eq. (3) (i.e., $D = L = 0$ during exo-atmospheric flight). Because all propulsion is impulsive, the thrust does not appear in the differential equations of Eq. (3).

4.1 Trajectory Optimization Problem

The trajectory optimization problem is now stated as follows. Determine the trajectory $(r, \theta, \phi, v, \gamma, \psi)$ and the corresponding controls (α, σ) that transfer a vehicle from an initial circular orbit Θ_0 to a terminal circular orbit Θ_f while minimizing

$$\sum_{i=1}^{3} \Delta V_i \tag{8}$$

Table 1: Vehicle and Astrodynamic Data.

Quantity	Numerical Value
m_0	1720 kg
m_{dry}	516 kg
I_{sp}	310 s
S	11.69 m^2
C_{D0}	0.032
K	1.4
$C_{L,\alpha}$	0.5699
$\dot{Q}$	11.357 kw/m^2
μ	3.986×10^{14} m^3/s^2
ρ_0	1.225 kg/m^3
H	7200 m
R_e	6378145 m
v_e	$\sqrt{\mu/R_e}$

In all cases it is assumed that the initial circular orbit is equatorial while the terminal circular orbit is of the same size, but has a different inclination.

5 Results

The aforementioned trajectory optimization problem was solved using a multiple-phase version of the *Gauss pseudospectral method* [11]. The problem was scaled canonically from SI units to the following system of units:

$$\begin{aligned} \text{LENGTH} &= \text{Units of } R_e \\ \text{SPEED} &= \text{Units of } v_e \\ \text{TIME} &= \text{Units of } R_e/v_e \\ \text{MASS} &= \text{Units of } m_0 \end{aligned}$$

For the aforementioned vehicle and trajectory design, Fig. 2 shows a plot comparing the minimum ΔV required for both an aeroassisted and all-propulsive transfer between two circular orbits of altitude 185.2 km. The all propulsive ΔV maneuver was calculated using $\Delta V = 2v_{circ} \sin i/2$ (see Ref. [14]), where Δi is the desired inclination change, and v_{circ} is the speed of a spacecraft in circular orbit. It is seen that the amount of fuel consumed by the all-propulsive maneuver increases significantly faster than does the aeroassisted maneuver and that only a small amount of inclination change is required for the aeroassisted maneuver to outperform the all-propulsive maneuver. Moreover, the all-propulsive and aeroassist ΔV grows rapidly as the terminal inclination increases. In this study the maximum achievable inclination change was found to be 58 deg. With this inclination change capability and an initial equatorial circular orbit, the vehicle can potentially be transferred to orbits that provide viewing of a variety of strategic points over the Earth. Furthermore, because of the relatively small size of the fully loaded spacecraft, a small launch vehicle such as a Minotaur IV can potentially deliver the vehicle to the required initial orbit. In particular, a Minotaur IV can place a 1720 kg payload into a 185 km orbit with an inclination of 28.5 deg when launched from Cape Canaveral Air Force Station (CCAFS). Table 2 shows the inclination change required to reach several strategic targets from a 28.5 deg inclination circular orbit. One key assumption in this table is that the inclination of an orbit that would place a spacecraft over a strategic target is equal to the latitude of the target. This assumption can be made for a circular orbit over a spherical earth. Examining Table 2 in more detail, it is seen that every strategic location can be observed from an initial 28.5 deg inclination. A more detailed look at the key features of the

Table 2: Inclination change required to reposition a small maneuverable spacecraft from a 28.5 deg inclined circular orbit to orbits that cover strategic locations around the globe.

Location	Pyongyang	Tehran	Beirut	Beijing	Fallujah	Moscow
Δi	10.54 deg	6.95 deg	5.04 deg	11.42 deg	4.93 deg	27.26 deg

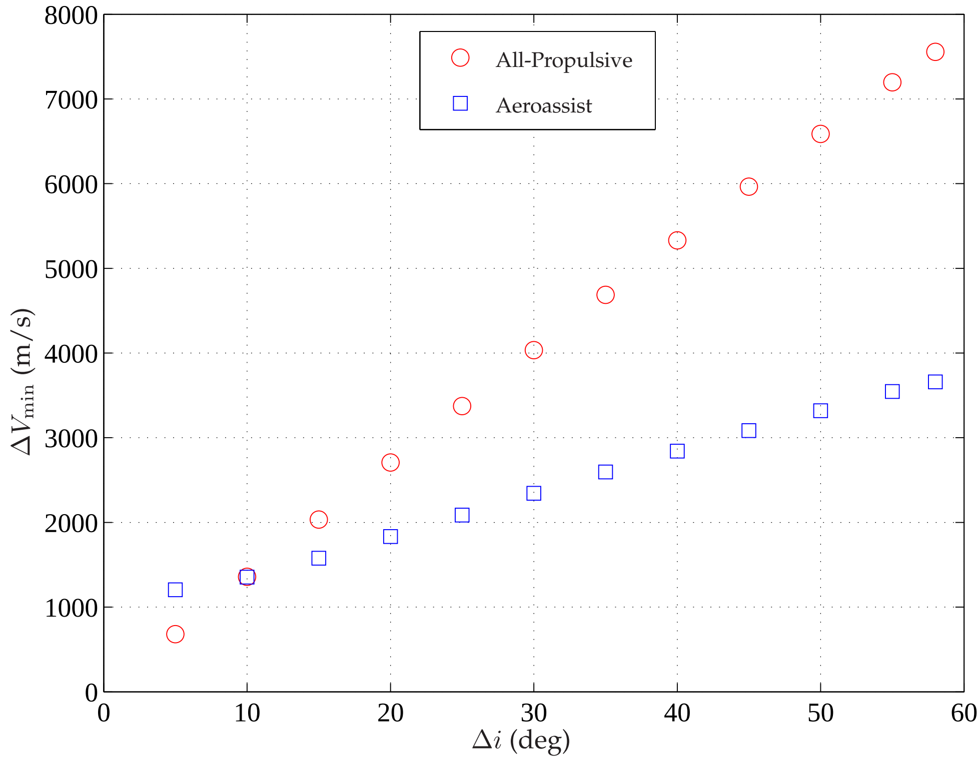

Figure 2: Comparison of optimal aeroassisted ΔV and propulsive ΔV as a function of inclination change, Δi required to transfer a vehicle between two circular orbits of altitude 185.2 km

solution is shown in Figs. 3–5 where the altitude and speed are shown, respectively, for various final inclinations as a function of time for a maximum allowable heating rate of 4543 kw/m^2 and different values of final inclination. As expected, Figs. 3 and 4 show that the vehicle dives further into the atmosphere and loses more speed as the required inclination change increases. Fig. 5 is a color-gradient plot of altitude vs. groundtrack of the vehicle for a terminal inclination of 50 deg as the vehicle moves from its initial equatorial orbit to its terminal inclined orbit. Finally, Fig. 6 is a parametric study of the minimum ΔV as a function of heating rate for different required inclination changes. It is seen from Fig. 6 that the minimum ΔV increases as the maximum allowable heating rate decreases and also increases as the required inclination change increases. It also can be seen from Fig. 6 that $\partial \Delta V / \partial i_f$ is much larger than $\partial \Delta V / \partial \dot{Q}_{\max}$, thereby indicating that the minimum ΔV is much more sensitive to changes in inclination than it is to changes in maximum allowable heating rate.

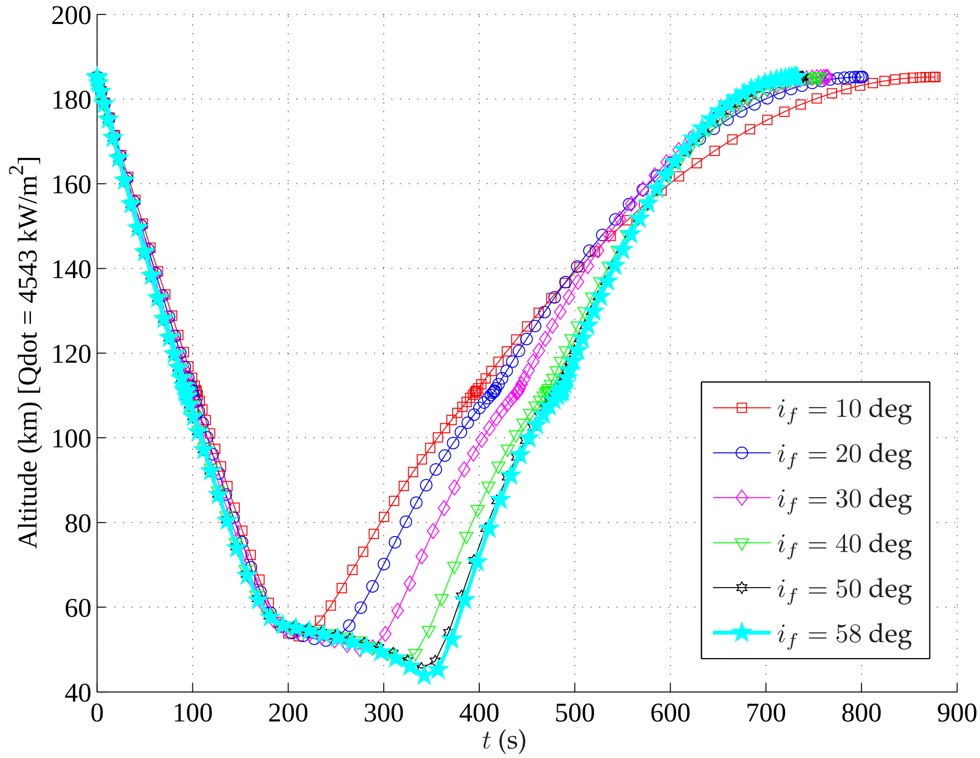

Figure 3: Comparison of altitude vs. time for trajectories of varying inclination change.

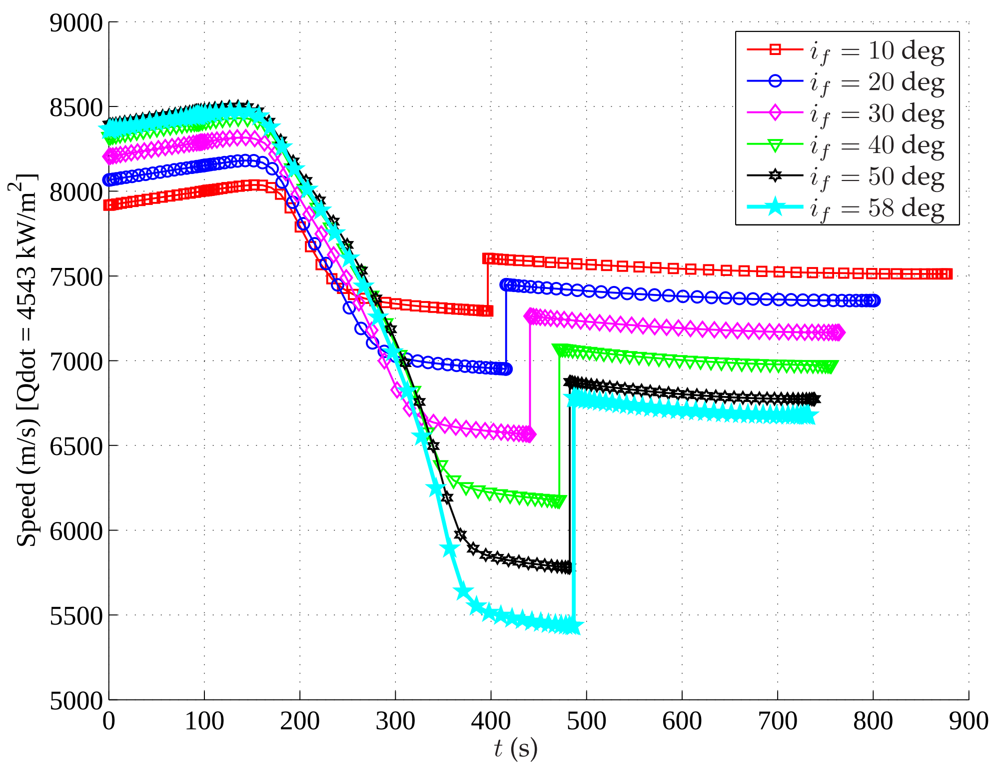

Figure 4: Comparison of altitude vs. time for trajectories of varying inclination change.

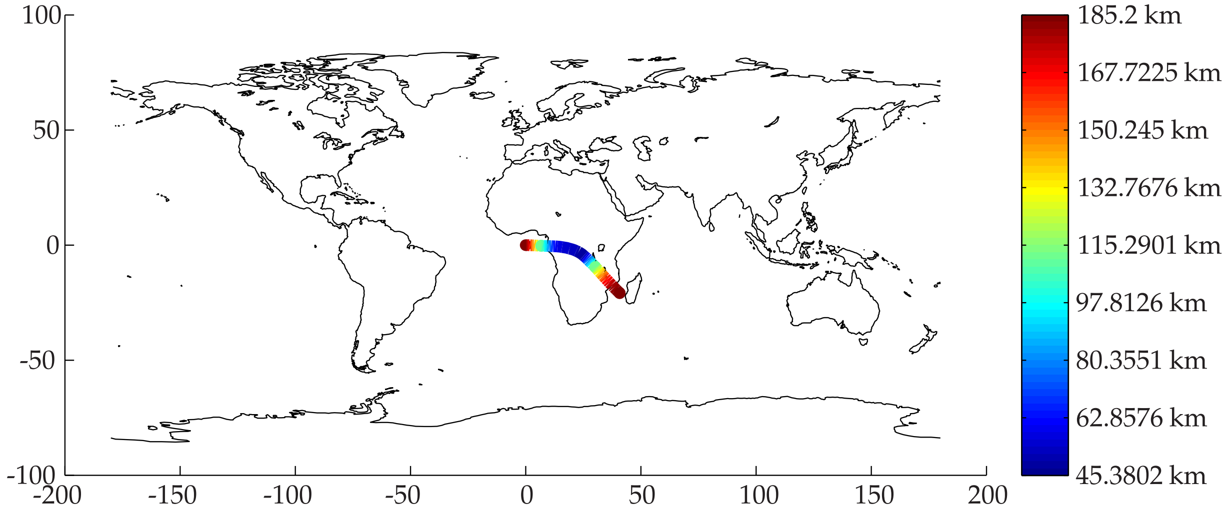

Figure 5: Color-gradient plot of altitude vs. groundtrack showing the motion of the trajectory for a transfer between a circular equatorial LEO orbit and a circular LEO orbit with inclination 50 deg.

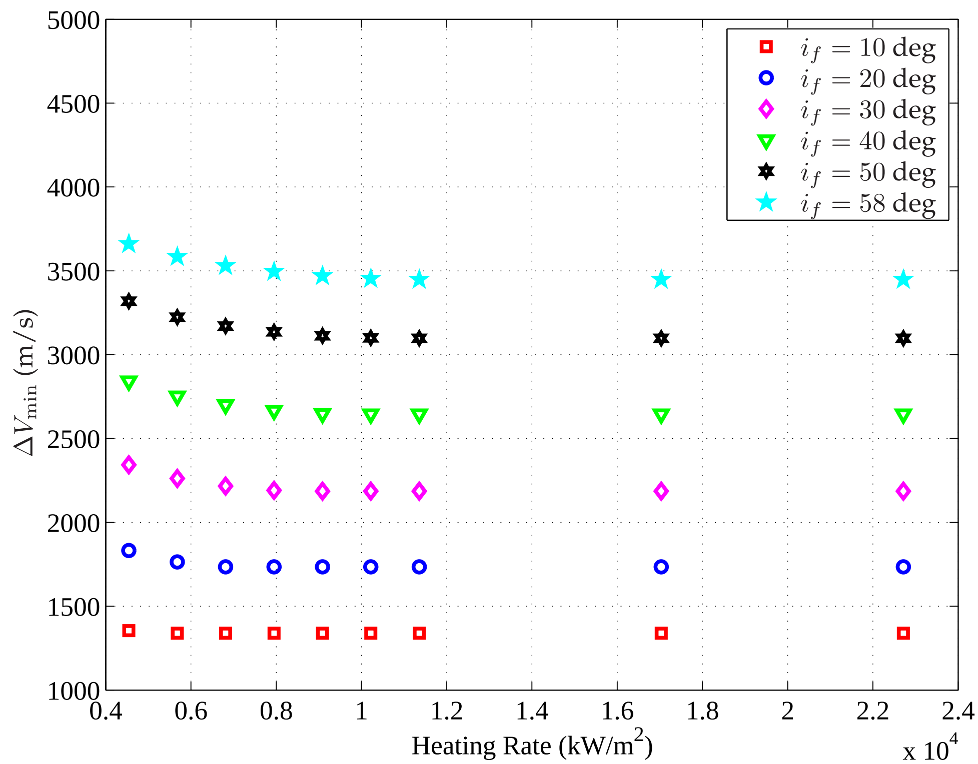

Figure 6: Comparison of $\Delta V_{\min}$ for a variety of heating rates and inclination changes.

6 Future Work

Future research on aeroassisted orbital transfer will focus on the design and deployment of an operational asset that is capable of meeting all the requirements of a demonstration mission. The hybrid vehicle design will address aerodynamics, attitude determination and control, structural layout, imaging payload design, propulsion, power and communications. The challenges of packaging a vehicle within the fairing of a Minotaur IV will also be addressed.

7 Conclusions

A concept for optimal mission planning of space assets for operationally responsive space has been presented. The mission design has been conceived as an aeroassisted orbital transfer problem between two low-Earth orbits. The aeroassisted orbital transfer problem has been described schematically and representative initial and terminal orbits are given for which to execute the mission plan. Some high-level results were shown along with the fuel requirements for these maneuvers. The aeroassisted maneuvers were compared with typical all-propulsive maneuvers and it was found that the aeroassisted maneuvers consumed significantly less fuel as compared to all-propulsive maneuvers. The results presented here show that the concept of using an aeroassisted orbital transfer is an excellent option for operationally responsive space.

Acknowledgments

The authors gratefully acknowledge support for this research by the NASA Florida Space Grant Consortium.

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