OPTIMAL MISSION ABORTS OF A WINGED ORBITAL STAGE

Michael Mayrhofer and Gottfried Sachs
Institute of Flight Mechanics and Flight Control
Technische Universität München
Boltzmannstraße 15, 85748 Garching bei München, Germany
phone: +49 89 289 16076, fax: +49 89 289 16058
email: mayrhofer@lfm.mw.tum.de

Abstract

In recent years, two-stage winged flight systems for space transportation have been investigated in order to improve the space transportation capability and to reduce costs. These flight systems use the atmosphere to produce lift and thrust. The carrier stage is equipped with airbreathing turbo-ramjet engines, while the orbital stage uses rocket engines. An issue of primary concern is flight reliability and safety which may also be improved by the new space transportation concepts. An important safety aspect is the capability for safe aborts in case of emergency.

A primary reason for a mission abort is a main engine failure shortly after separation. Optimization results are presented which show the maximal achievable range of the orbital stage in case of a mission abort. A particular aspect of such a mission termination is the release of fuel for reducing weight prior to landing. Computations show that there is a range increase when fuel is optimally released during the flight.

Another important abort scenario is the mission termination from orbit. Results for the two basic abort scenarios, the Direct Abort and the Abort to Launch Site, are presented and discussed.

Key words: Mission Abort, Trajectory Optimization, Reentry, Winged Orbiter, Fuel Draining.

Nomenclature

a	exponent of exponential atmospheric model
c	speed of sound
g	acceleration due to gravity
h	altitude
l	specific lift
m	mass of the flight system
$\dot{m}_f$	fuel mass flow
q	heat load
$\bar{q}$	dynamic pressure, $\bar{q} = (\rho/2)V^2$
$\dot{q}$	heat flux
r_E	radius of the Earth
s	range
C_L	lift coefficient
C_L^*	C_L at maximum $(C_L/C_D)_{max}$
C_D	drag coefficient
D	drag
I_{sp}	specific impulse
L	lift
M	Mach number
S	reference area
T	thrust
V	velocity
V^*	velocity at maximum $(C_L/C_D)_{max}$
α	angle of attack
δ_T	throttle setting
ϵ_T	thrust vector angle
γ	flight path angle
μ_a	bank angle
ρ	air density
χ	azimuth angle
ω_E	angular velocity of the Earth
Δ	geocentric latitude
Λ	geographic longitude

Introduction

Important aspects for developing modern space transportation systems are economy and reliability. For yielding an improvement, new concepts for space transportation systems are proposed and investigated in various countries.

A promising concept is a two-stage hypersonic vehicle with a winged carrier stage propelled by airbreathing turbo-ramjet engines and a winged orbital

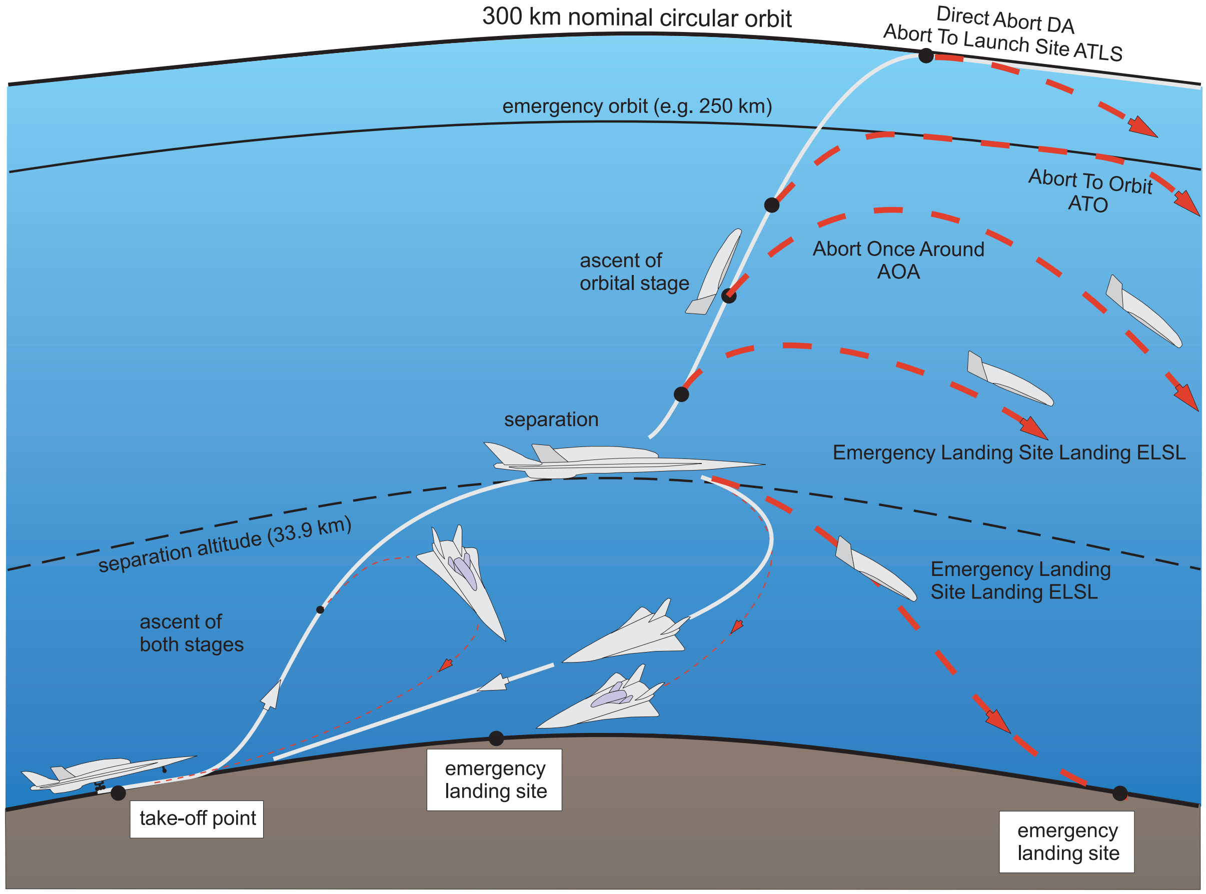

Figure 1: Abort scenarios

stage powered by rockets. This new space transportation concept features inherent abort capabilities which may be superior when compared to current systems, so that an improvement in overall safety can be achieved. Since such capabilities are a critical issue, they should be known as early as possible in the conceptual phase.

There are many orbital stage abort strategies which have to be investigated. An overview of basic abort scenarios is provided by Fig. 1:

- mission abort during ascent
  - intact abort
    - Emergency Landing Site Landing (ELSL)
    - Abort Once Around (AOA)
    - Abort To Orbit (ATO)
  - contingency abort
- mission abort from orbit
  - intact abort
    - Direct Abort (DA)
    - Abort To Launch Site (ATLS)
  - contingency abort

First of all there is a significant difference in respect to the total energy of the orbital stage between an abort during the ascent and an abort from orbit. A higher amount of total energy implies a greater performance for a mission abort.

There are two basic types of abort modes: intact aborts and contingency aborts. Intact aborts are designed to provide a safe return of the orbiter to a nominally planned landing site. Contingency aborts are designed to permit flight crew survival when an intact abort is not possible.

Three different strategies are possible for an intact mission abort during ascent of the orbital stage.

The abort to orbit ATO scenario is used to boost the orbital stage to a safe orbital altitude when performance has been lost and it is impossible to reach the planned orbit. The abort once around AOA is used in cases in which the loss of vehicle performance is too high to achieve a circular orbit. If the loss of performance is even too high for an AOA, the only way to do an intact abort is an emergency landing site landing ELSL.

The intact mission abort from orbit is subdivided into the direct abort, which makes it possible to land within the next 2 hours , and the abort to launch site ATLS. In the worst case the ATLS needs 24 hours from the start of mission abort to touch down at launch site.

MODEL OF FLIGHT SYSTEM

For the trajectory optimization problem, a mass point modelling is applied for describing the flight system dynamics. With reference to a rotating, spherical Earth, the equations of motion can be expressed as (Fig. 2):

$$\dot{V} = \frac{T\cos(\alpha+\epsilon_T) - D}{m} - g\sin\gamma + \omega_E^2(r_E+h) \cdot \cos\Delta\,(\sin\gamma\cos\Delta - \cos\gamma\sin\Delta\cos\chi)\,,$$

$$\dot{\gamma} = \frac{T\sin(\alpha+\epsilon_T) + L}{mV}\cos\mu_a + \cos\gamma\left(\frac{V}{r_E+h} - \frac{g}{V}\right) + 2\omega_E\cos\Delta\sin\chi + \frac{\omega_E^2(r_E+h)}{V}\cos\Delta \cdot \cdot\,(\cos\gamma\cos\Delta + \sin\gamma\sin\Delta\cos\chi)\,,$$

$$\dot{\chi} = \frac{T\sin(\alpha+\epsilon_T) + L}{mV\cos\gamma}\sin\mu_a + \frac{V}{r_E+h}\cos\gamma\sin\chi\tan\Delta - 2\omega_E(\tan\gamma\cos\Delta\cos\chi - \sin\Delta) + \frac{\omega_E^2(r_E+h)}{V\cos\gamma}\sin\Delta\cos\Delta\sin\chi, \tag{1}$$

$$\dot{\Delta} = \frac{V\cos\gamma\cos\chi}{r_E+h},$$

$$\dot{\Lambda} = \frac{V\cos\gamma\sin\chi}{(r_E+h)\cos\Delta},$$

$$\dot{h} = V\sin\gamma,$$

$$\dot{m} = -\dot{m}_f$$

with

$$g = g_0\left(\frac{r_E}{r_E+h}\right)^2. \tag{2}$$

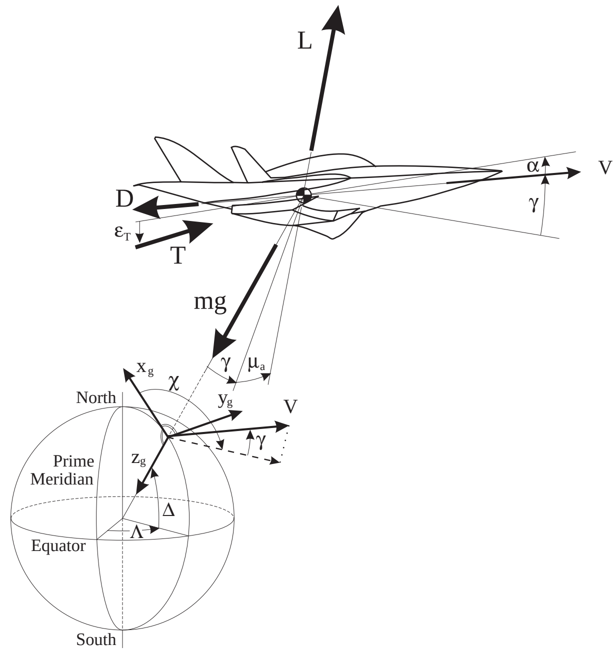

Figure 2: Forces on the overall flight system

These equations are basically valid for the overall system as well as for the single stages.

The orbital stage considered in this paper is similar to the upper stage HORUS of the SÄNGER-concept. The orbital stage is a winged vehicle propelled by rockets. In addition, it is equipped with a propulsive system considered for orbital maneuvering (OMS).

The aerothermodynamic model can be described as

$$\begin{aligned} L &= C_L\bar{q}S, \\ D &= C_D\bar{q}S \end{aligned} \tag{3}$$

with $C_L = C_L(\alpha, M)$ and $C_D = C_D(\alpha, M)$. The models for C_L and C_D were computed from data fields to obtain smooth functions. The model of the main rocket propulsion of the orbital stage can be expressed as

$$T = \delta_T T_{max}, \qquad (4)$$
$$\dot{m}_f = \frac{T}{g_0 I_{sp}}.$$

OPTIMIZATION PROBLEM

A major aim of trajectory optimization is to find a control law which makes it possible to transfer a flight system from a starting to a final point, subject to boundary conditions and path constraints, with a minimum cost function.

The abort trajectories for achieving a safe mission termination and landing are treated as an optimal control problem.

The controls are angle of attack α, roll angle μ_a, throttle setting δ_T and fuel mass flow for draining $\dot{m}_f$.

The initial conditions for the orbital stage are given by the conditions at the separation from the carrier stage. The separation is optimized for a minimum fuel consumption of the overall system.[1] The state variables at separation are presented in Table 1.

state variable	value at separation
h	33.9 km
M	6.8
γ	8.71 deg
χ	90 deg
Δ	16.5 deg
Λ	3.44 deg
$m_{orbital\ \ stage}$	96 Mg

Table 1: Initial conditions of mission aborts

The nominal orbit for the orbital stage is circular, at an altitude of 300 km and an inclination of 16.5 deg.

The final conditions for an intact mission abort are given by $h = 1\,\mathrm{km}$ and $V = 135\,\mathrm{m/s}$ with a flight path angle of $\gamma = -6\,\mathrm{deg}$.

Realistic contraints and vehicles conditions are considered:

- strength and trajectory related constraints
- constraints resulting from the failure
- landing configuration (e.g. fuel draining for mass decrease)

The path constraints are shown in Table 2. The minimum dynamic pressure constraint is considered to be valid only at altitudes lower than 45 km. The specific lift l is a reference for describing the structure loading. It is defined as

$$l = \frac{\sqrt{L^2 + D^2}}{S}. \qquad (5)$$

The maximum reference heat flux at the stagnation point $\dot{q} = 375\,\mathrm{kW/m^2}$ is taken from Refs.[2] and Ref.[3] . The dynamic pressure constraints are from Ref.[4] .

	Minimum	Maximum
α [deg]	0	45
δ_T [-]	0	1
μ_a [deg]	-90	90
$\bar{q}$ [kPa]	5	50
l [N/m^2]	0	8000
$\dot{q}$ [kW/m^2]	-	375

Table 2: Path constaints of orbital stage

For solving this type of optimal control problem, efficient numerical optimization methods and computational techniques are required which are capable of coping with complex functional relationships including various kinds of constraints.

The procedures which were successfully applied in this paper are a parameterization optimization technique[5] with the graphical environment GESOP[6] .

RESULTS

The first result concerns mission abort capabilties for the orbital stage of the considered flight system if ignition of the main rocket engine fails (ignition occurs after separation of the orbital stage from the carrier). The related abort scenarios are called Emergency Landing Site Landing ELSL, because an intact abort enables only landing at an emergency landing site.

ELSL with Fuel Draining

A first approach to work out optimal abort strategies for main engine failures is a glide flight of the orbital stage with finally landing at an emergency landing site. The gear of the orbital stage is designed for a touch down with almost dry mass. As a consequence, it is necessary to release fuel before landing. As a

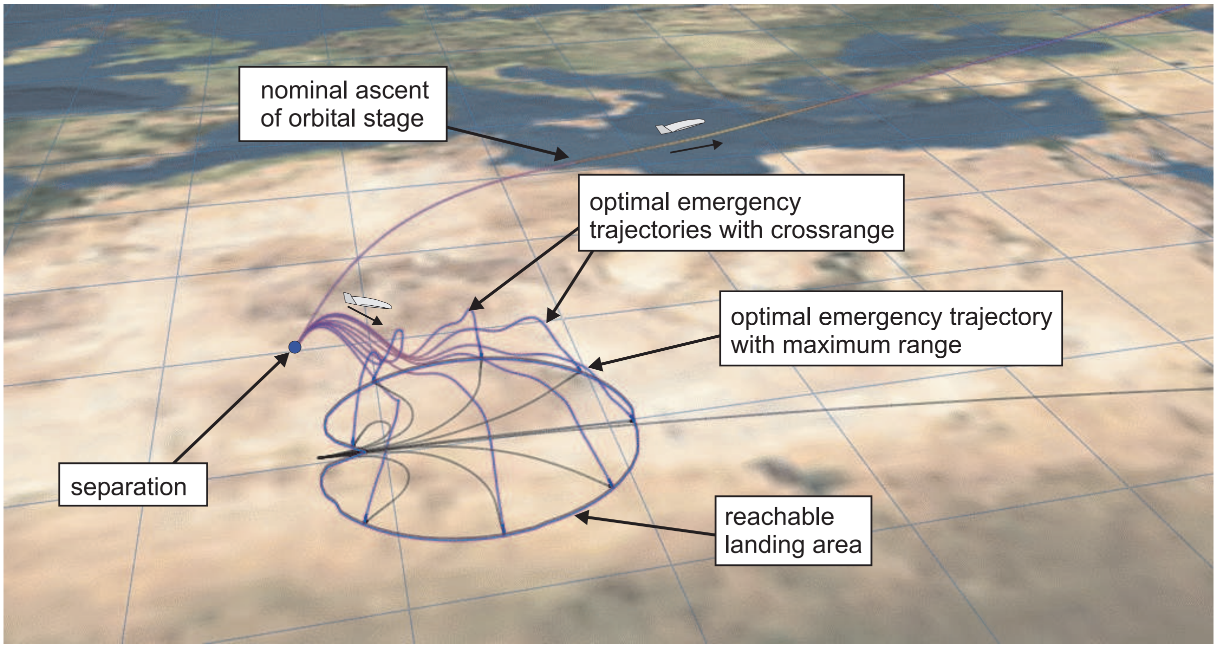

Figure 3: Flight path of orbital stage without propulsion for an ELSL

result of optimization fuel draining should occur as soon as possible for increasing the range. Therefore this flight is modelled as a two phase problem. In the first phase the turbo pumps drain the fuel at a constant mass flow and in the second phase no fuel draining takes place.

To find out the reachable emergency landing sites for such mission aborts it is useful to determine optimal flight paths for a maximized range performance function.

The corresponding optimal flight path is shown in Fig. 3 and the control history is presented in Fig. 4.

The maximum range for a gliding flight from the separation point with the fuel pumped off is 813 km. The final point is at 16.24 deg geocentric latitude and at 11.05 deg geographic longitude. This mission abort lasts about 21 min.

Fig. 3 shows the reachable landing area for this kind of mission termination.

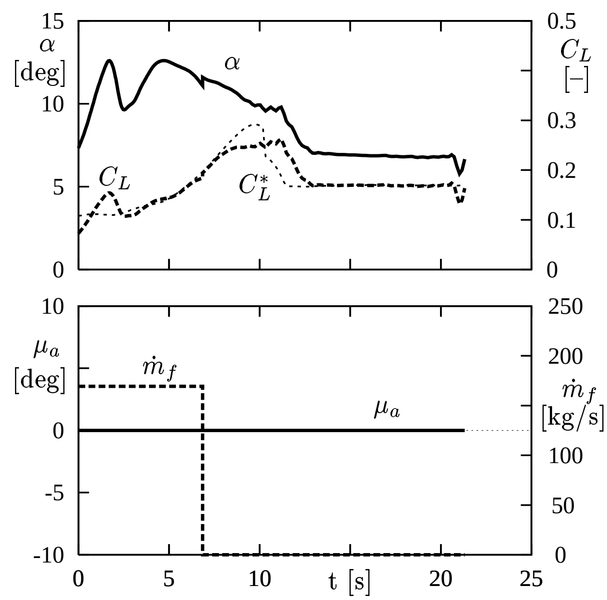

Figure 4: Controls of orbital stage without propulsion for an ELSL

Range Increase by Fuel Draining

The above results show that proper fuel draining increases the range of the vehicle in gliding flight. In the following, a physical explanation for this effect is provided.

The velocity for a maximum range flight is approximately given by

$$V = V^* = \sqrt{\frac{2 \cdot m(t) \cdot g}{\rho(t) \cdot C_L^*(t) \cdot S}}. \tag{6}$$

Differentiation yields

$$\dot{V} = \frac{V}{2} \cdot \left(\frac{\dot{m}}{m} - \frac{\dot{\rho}}{\rho} - \frac{\dot{C}_L^*}{C_L^*} \right). \tag{7}$$

The air density changes can be estimated with the use of an exponential atmospheric density model

$$\rho = \rho_i \cdot e^{-a_i \cdot (h - h_i)}, \quad \dot{\rho} = d\rho/dt = -\rho \cdot a_i \cdot \dot{h}. \tag{8}$$

Considering the fuel draining at constant mass flow $\dot{m}_f$, the change of the flight system mass is

$$m = m_0 - \dot{m}_f \cdot t, \quad \dot{m} = dm/dt = -\dot{m}_f. \tag{9}$$

With the use of Eqs. (8) and (9), $\dot{V}$ can be rewritten as

$$\dot{V} = -\frac{V}{2} \cdot \left(\frac{\dot{m}_f}{m} - a_i \cdot \dot{h} + \frac{\dot{C}_L^*}{C_L^*} \right). \tag{10}$$

The changes of velocity and altitude are determined on the basis of constant energy during fuel draining ($dm = -dm_f$):

$$\begin{aligned} m\frac{V^2}{2} + mgh &= (m+dm)\frac{(V+dV)^2}{2} + dm_f\frac{V^2}{2} \\ &\quad + (m+dm)g(h+dh) + dm_f gh. \end{aligned}$$

Simplification by neglecting higher order terms yields

$$\dot{h} = -(g/V) \cdot \dot{V}. \tag{11}$$

With this relation, Eq. (10) may be rewritten as

$$\dot{h} = (\dot{m}_f/m) \cdot h_{ref} \tag{12}$$

with

$$h_{ref} = \frac{1}{\frac{g}{c^2} \cdot \left(\frac{2}{M^2} + \frac{1}{M \cdot C_L^*} \cdot \frac{dC_L^*}{dM} \right) + a_i}$$

For the aerodynamic model of the orbital stage C_L^* can be estimated as

$$\begin{aligned} M < 0.8: \quad C_L^* &= 0.165 \\ M > 1.2: \quad C_L^* &= \frac{0.285}{\sqrt{M^2 - 0.218}} + 0.0664. \end{aligned}$$

The resulting h_{ref} values are shown in Fig. 5

Since h_{ref} increases with Mach number, it follows from Eq. (12) that the largest altitude gain is achieved at the highest possible Mach number. Because h_{ref} does not change much at Mach numbers higher than 2.5 following simplified expression holds

$$h_{ref} = \frac{0.95}{a_{strato}} = const. \tag{13}$$

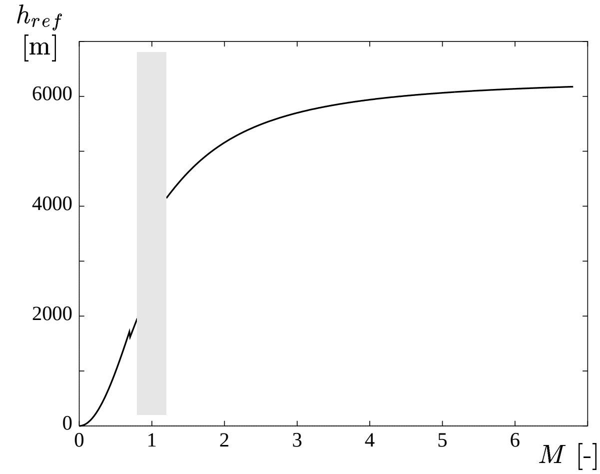

Figure 5: h_{ref} for investigated orbital stage

With this expression, integration of Eq. (12) yields

$$\Delta h = \frac{0.95}{a_{strato}} \cdot ln\frac{m_0}{m_1}. \tag{14}$$

It is assumed that the altitude increase described by Eq. (14) can be maintained until the final part of the trajectory when compared with a flight without fuel draining. Then, this altitude difference can be used to increase the range by continuing the glide at the subsonic $(C_L/C_D)_{max,sub}$ which is the highest (C_L/C_D) value for the hole Mach number range. The range increase can be expressed as

$$\Delta s = \Delta h \cdot (C_L/C_D)_{max,sub}. \tag{15}$$

Combining Eqs. (14) and (15) yields an expression for the estimated range increase of a hypersonic vehicle due to fuel draining

$$\Delta s = \left(\frac{C_L}{C_D} \right)_{max,sub} \cdot \frac{0.95}{a_{strato}} \cdot ln\frac{m_0}{m_1}. \tag{16}$$

Abort from Orbit

Other important mission termination procedures concern the different kinds of mission aborts from orbit. Basically these aborts are splitted up into two groups. The first is the Abort to Launch Site ATLS. On the other hand the Direct Abort DA is designed to provide an immediate mission termination from orbit.

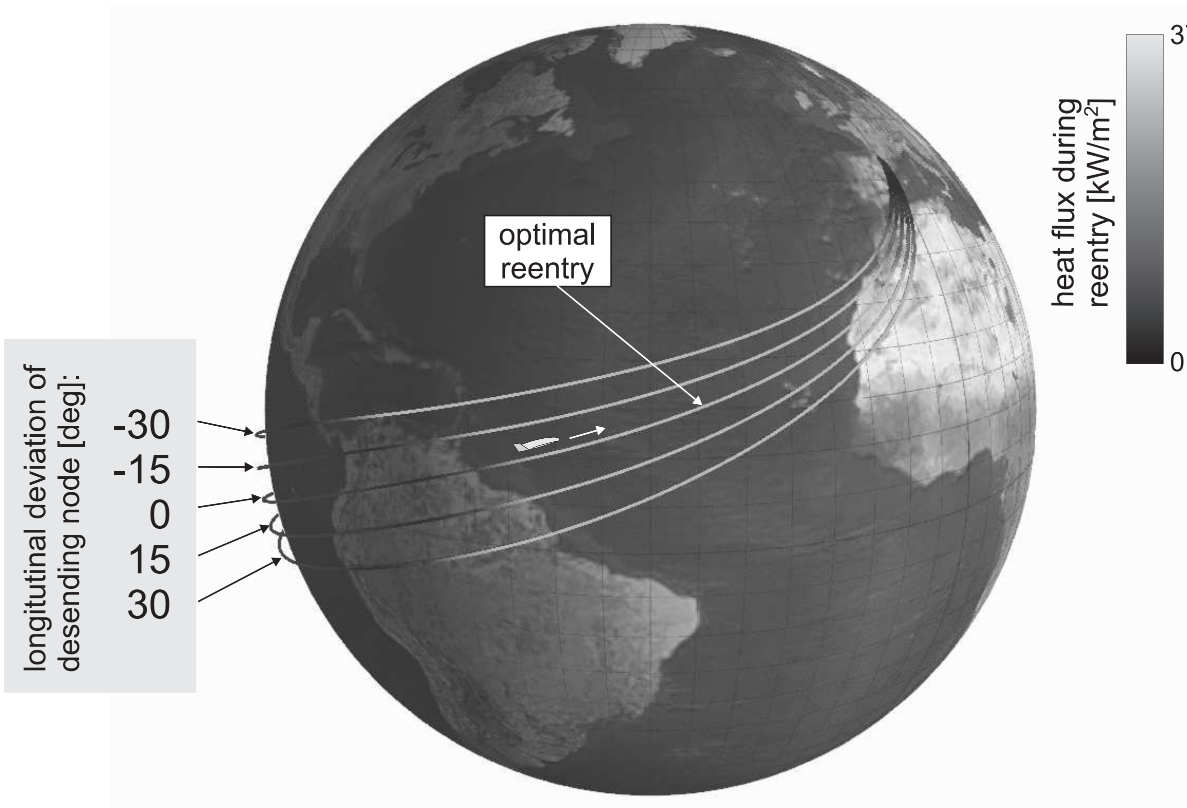

Figure 6: Optimal ATLS

The basic ATLS corresponds to the nominal reentry. This mission abort is shown in Fig. 6. The orbital stage must have a certain orbital position at the beginning of the abort. The optimal descending node for a minimum reentry heat load of 506 MJ/m^2 is at 106 deg eastern longitude. Therefore the corresponding deorbit point is at 15.88 deg southern latitude and at 175.13 deg eastern longitude. The control history for this high crossrange reentry is presented in Fig. 7. For this high crossrange reentry there are no bank reversals and the angle of attack does not exceed 35 deg.

The altitude Mach number relation shown in Fig. 8 reveals that the reentry can be divided into several phases. The first phase is the rather short deorbit maneuver. Then, the orbiter descends in a parabolic flight to an altitude of about 120 km where the transition to hypersonic flare begins. At the end of the hypersonic flare the maximum heat flux is reached. During this phase the orbiter is controlled along the maximum heat flux constraint. Thereafter, a phase takes place where no constraint is active till reaching the minimum dymanic pressure limit. The final phase is the end phase where the final landing area should be reached.

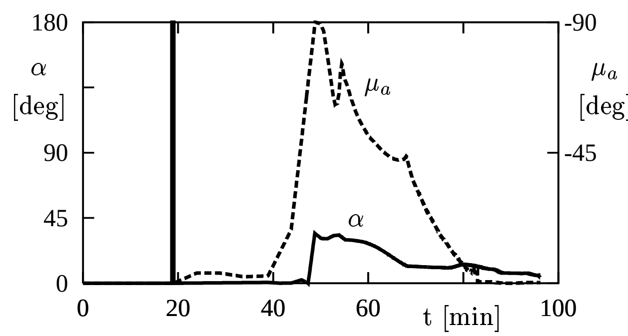

Figure 7: Control history for optimal nominal reentry

Every circle around the Earth on the nominal circular orbit the desending node deplaces 22.59 deg to

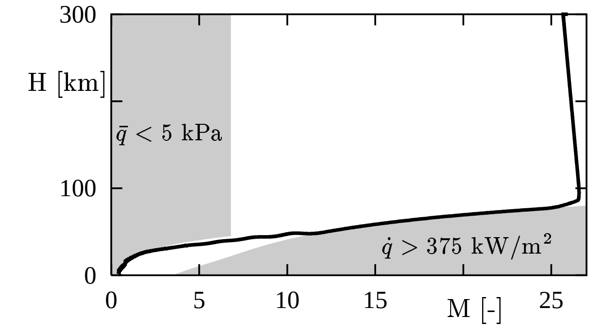

Figure 8: Flight envelope

the west, due to the Earth rotation. Therefore it is possible to get near the optimal descending node only once a day. Flight paths for the deviation of ±15 deg and ±30 deg from the optimal descending node are also shown in Fig. 6. The time flying along the maximum heat flux constraint raises which and a higher amount of reentry heat load results. A plot for this increasing reentry heat load with the descending node deviation is provided by Fig. 9.

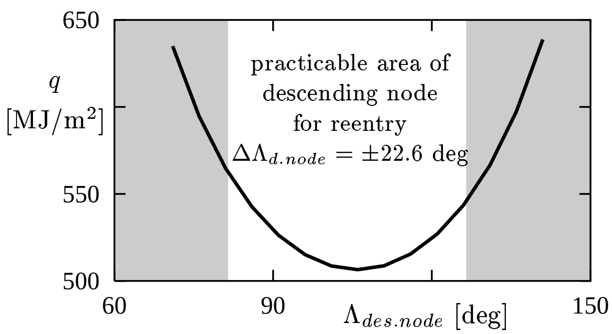

Figure 9: Reentry heat load versus deviation of descending node longitude

If there is not enough time for an ATLS, e.g. due to a system failure in the environment control and life support system ECLSS, the orbiter has to perform a Direct Abort DA. Due to the low inclination of 16.5 deg and the high crossrange cabability of about 28 deg only one emergency landing site is sufficient enough for a DA. The location of this landing site must be within a belt along the Equator with a width of about ±12 deg. Such an emergency landing site may also be used for the ELSL due to a failure after separation.

CONCLUSIONS

Mission aborts are considered for the orbital stage of a two-stage hypersonic vehicle. Intact abort trajectories for main engine failures shortly after separation are optimized. The performance can be improved by an optimal control of fuel draining. To maximize the abort performance the fuel should be released at the highest possible velocity.

Intact aborts from orbit are also discussed. A special Abort To Launch Site ATLS corresponds to the nominal reentry which has been optimized with respect to a minimum heat load. The influence of a deviation of the deorbit point to the thermal protection system has been determined. Another mission abort scenario from orbit which allows immediate mission termination is the Direct Abort DA. Results are presented for the area which can be reached by the vehicle for an emergency landing.

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