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Reduction of Coolant Fuel Losses in Hypersonic Flight by Optimal Trajectory Control

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Engine cooling demands of a hypersonic carrier vehicle propelled by a turbo/ramjet combination are considered for a three-dimensional range trajectory with a return to the launch site. The fuel flow required for cooling the engine is included in an overall fuel optimization problem, so that the flight phases during which more fuel is required for engine cooling than for producing thrust are adequately accounted for. It is shown that a significant fuel saving can be achieved compared with the classical trajectory optimization approach that takes into consideration only the fuel required for producing thrust.

Nomenclature

A	= cross-sectional area, m^2
C_D	= drag coefficient
C_L	= lift coefficient
c_p	= heat capacity, J/(kgK)
D	= drag, N
f_α	= thrust factor for angle-of-attack dependency
g	= acceleration due to gravity, m/s^2
h	= altitude, m
L	= lift, N
M	= Mach number
m	= mass, kg
m_f	= fuel mass consumed, kg
Nu	= Nusselt number, $\alpha_q x/\lambda$
n	= load factor
Pr	= Prandtl number, $\eta c_p/\lambda$
p	= pressure, N/m^2
$\bar{q}$	= dynamic pressure, $(\rho/2)V^2$, N/m^2
Re	= Reynolds number, $V\rho x/\eta$
r_e	= radius of the Earth, m
S	= reference area, m^2
T	= thrust, N
$\tilde{T}$	= temperature, K
t	= time, s
t_f	= final time, s
V	= speed, m/s
x	= distance along surface, m
α	= angle of attack, deg
α_q	= heat transfer coefficient, W/(m^{2}K)
β	= shock wave angle, deg
γ	= flight-path angle, deg
Δ	= geocentric latitude, deg
δ_T	= throttle setting
ε_T	= thrust vector angle, deg
η	= dynamic viscosity, kg/(ms)
θ_0	= angle between zero lift direction and a body reference line, deg
κ	= specific heat
Λ	= geographic longitude, deg
λ	= heat conductivity, J/(msK)
μ_a	= velocity bank angle, deg
ρ	= atmospheric density, kg/m^3
σ	= specific fuel consumption, s/m
ϕ_f	= equivalence ratio
χ	= velocity azimuth angle, deg
ω_e	= angular velocity of the Earth, 1/s

Subscripts

cooling	= with m_f consumption for engine cooling
t	= with p total pressure
thrust	= with m_f consumption for thrust production
0	= with g at zero altitude

Introduction

NEW concepts of aerospace plane type vehicles are currently being considered as a means for expanding the usable flight regime to hypersonic speeds and for providing an improved space transport capability. There are various differences among these concepts. However, airbreathing propulsion and lifting capability represent common features.

Hypersonic flight poses challenging problems. This particularly holds for the propulsion system. One problem concerns the hot environment that the engine is exposed to at hypersonic speed. A sophisticated thermal management system is required to cope with high-temperature problems.[1,2] A technique for efficiently solving this problem is to use hydrogen fuel in a cooling circuit before it is fed to the engine for producing thrust. At each instant along the flight path there are two instantaneous fuel rates of interest: the fuel rate required for thrust production and the fuel rate required for cooling. The actual fuel rate must be the larger of the two; thus, if the cooling requirement exceeds the thrust requirement, then unburned fuel is considered to be dumped. This problem is addressed, and it is shown that an appropriate trajectory optimization provides a means to substantially reduce the fuel losses caused by engine cooling.

Temporary storage of the excess fuel in phases with a higher fuel demand for cooling than for producing thrust and for use for thrust production at a later time may be a possibility for avoiding a fuel loss. This possibility is not considered as feasible because additional installations would be required, which are rather complex. Furthermore, there may be safety risks because of the hot temperature of the excess fuel after the cooling process.

Optimization techniques for trajectory control are necessary to fully exploit the performance capability of hypersonic vehicles. Because of extreme performance requirements for these vehicles, fuel minimization is a primary goal in trajectory optimization. Papers on trajectory optimization have so far considered only fuel consumption necessary for thrust production,[3–14] without accounting for the fuel required for engine cooling. However, cooling requirements can be so large that a need may exist for reducing the fuel loss caused by cooling. It will be shown that the overall fuel consumption can

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be significantly reduced if fuel losses due to engine cooling are explicitly accounted for in a trajectory optimization formulation.

Heating problems and fuel-rate requirements for cooling the engine/airframe structure have been addressed in recent research. In Ref. 15 a method for determining the near-optimal operation of the propulsion system of a single-stage-to-orbit vehicle is presented.

The trajectory optimization technique proposed for reducing cooling fuel losses is applied to a hypersonic vehicle consisting of an airbreathing carrier and a rocket propelled orbital stage. The airbreathing propulsion system of the carrier consists of a turbo/ramjet combination that uses hydrogen fuel. The flight path of the carrier is considered, including a range requirement for releasing the orbital stage at a specified location.

Cooling Fuel for Engine Thermal Management

Hot engine components require adequate cooling, for which a concept is shown in Fig. 1. There are two different cooling circuits: the first uses an indirect technique and the second a direct one. The first circuit provides air for various components: 1) cooling air for the intake, 2) cooling air for the turbo engine, and 3) sealing air for the intake and nozzle flaps. The sealing air is needed to avoid damage of the flap mechanism due to the hot air-flow passing the gaps. The air extracted from the flow in the subsonic diffusor intake is passed through a heat exchanger that is used for reducing the air temperature to a sufficiently low level. Then, a compressor feeds the air through the engine components. After cooling the components, most of the air is again transferred to the mainstream flow of the engine.

The second circuit concerns the hydrogen fuel used for cooling, which offers a large heat-absorption capacity for cooling purposes, not only due to its temperature ($T = 20$ K), but also due to its mass. The hydrogen is heated to 900 K and then is fed to the engines or blown off. The heat transferred to the fuel may be considered to increase the thermodynamic efficiency. However, there is a corresponding cooling effect on the airflow through the engine. Since the air and the fuel are mixed in the burner, it is assumed that the heating effects balance each other. Therefore, no increase in thermodynamic efficiency is assumed for the overall process.

One part of the fuel is used for cooling the air in the heat exchanger. The other part is used for the direct cooling of engine components: the ramduct, the ramburner, and the nozzle. A mathematical model is developed for describing the cooling requirements. Each engine component is treated separately, according to its individual cooling demands. As an example, the model for calculating the airflow required for cooling the intake is presented in detail.

As a starting point, reference flight conditions (Fig. 2), which cover the flight parts where the engines are in the ramjet mode, were used with fixed combinations of M, h, α, and δ_T. For these flight conditions, material for describing the cooling demands of all engine components was provided by an industry company. This is used as reference data for developing a more general mathematical model. As an example, the cooling demands of the intake are shown

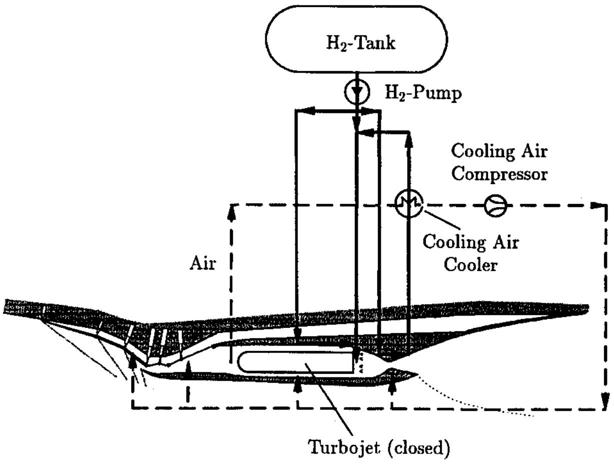

Fig. 1 Thermal management concept,[2] ramjet mode.

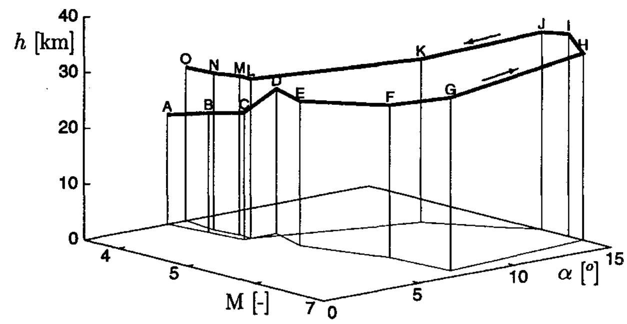

Fig. 2 Reference flight conditions for cooling model.

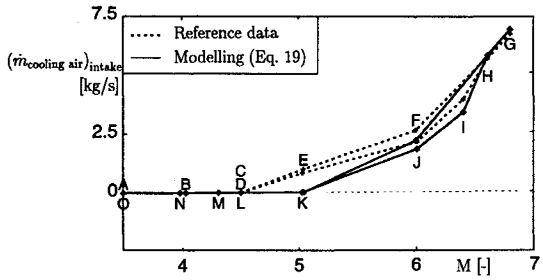

Fig. 3 Airflow required for intake cooling.

in Fig. 3 for the reference flight conditions (with letters A–O used for cross reference with Fig. 2).

Based on the reference values, a mathematical model was developed that allows the determination of the cooling demands for a wide range of M, h, α, and δ_T. This model includes several computational steps, which are described in the following for the intake.

The flow conditions at the intake are described with the use of approximative formulas. The oblique shock wave angle at the aircraft nose can be approximately expressed as[16]

$$\beta \approx \theta \cdot \{(\kappa + 1)/4 + \sqrt{[(\kappa + 1)/4]^2 + (1/M^2\theta^2)}\} = g_1(M, \alpha) \quad (1)$$

where

$$\theta = \theta_0 + \alpha \quad (2)$$

Using β, the flow condition ahead of the intake shock system (denoted by the subscript 1) can be computed. The deceleration of the flow to subsonic conditions in the intake (denoted by the subscript 2) leads to a total pressure loss. With the total pressure in the intake

$$p_{t2} = g_2(M_1)p_{t1} \quad (3)$$

the subsonic flow conditions in the intake can be computed. The function $g_2(M_1)$ describes the pressure loss at the shock system of the ramps and the intake configuration.[17]

The coefficient for the heat transfer from the air to the intake walls can be expressed as[18]

$$\alpha_q = Nu(\lambda/x) \quad (4)$$

For computing the heat transfer, the flow is considered turbulent. Assuming flat plate characteristics, the local dimensionless heat transfer coefficient is related to the flow and fluid properties by[18,19]

$$Nu = 0.0296 Re^{0.8} Pr^{0.33} \quad (5)$$

With the use of Eqs. (4) and (5), the heat transfer coefficient can be expressed as

$$\alpha_q = 0.0296 \frac{(V_3\rho_3)^{0.8}\lambda_3^{0.67}c_{p3}^{0.33}}{\eta_3^{0.47}x^{0.2}}$$

where subscript 3 denotes the flow conditions at the wall and x is the local coordinate in flow direction. With the air mass flow

$$\dot{m}_{\text{air}} = V_3\rho_3 A$$

the expression for α_q can be rewritten as

$$\alpha_q = \frac{0.0296}{x^{0.2}A^{0.8}} \cdot \frac{\dot{m}_{\text{air}}^{0.8}\lambda_3^{0.67}c_{p3}^{0.33}}{\eta_3^{0.47}} \tag{6}$$

From Eq. (6) it follows that

$$\alpha_q \propto \frac{\dot{m}_{\text{air}}^{0.8}\lambda_3^{0.67}c_{p,3}^{0.33}}{\eta_3^{0.47}} \tag{7}$$

The following functional relationships hold:

$$\lambda_3 = \lambda_3(\tilde{T}_3, p_3) \tag{8}$$

$$c_{p,3} = c_{p,3}(\tilde{T}_3) \tag{9}$$

$$\eta_3 = \eta_3(\tilde{T}_3) \tag{10}$$

$$\dot{m}_{\text{air}} \propto (\dot{m}_f)_{\text{thrust}} \tag{11}$$

where

$$(\dot{m}_f)_{\text{thrust}} = (\dot{m}_f)_{\text{thrust}}(M, \rho, \alpha, \delta_T) \tag{12}$$

$$\tilde{T}_3 = \tilde{T}_3(M, \rho, \alpha) \tag{13}$$

$$p_3 = p_3(M, \rho, \alpha) \tag{14}$$

$$\rho = \rho(h) \tag{15}$$

As a result, α_q can be described as a function of flight Mach number, altitude, angle of attack, and throttle setting:

$$\alpha_q = \alpha_q(M, h, \alpha, \delta_T) \tag{16}$$

The heat flux through the intake walls is assumed to be equal to the heat flux transferred to the cooling air, so that the following proportionality exists:

$$(\dot{m}_{\text{cooling air}})_{\text{intake}} \Delta\tilde{T}_{\text{cool}} \propto \alpha_q(\tilde{T}_3 - \tilde{T}_w) \tag{17}$$

where ΔT_{cool} is the temperature difference of the cooling air before and after cooling and T_w is the maximum allowable wall temperature (here, $T_w = 1600$ K).

From Eqs. (16) and (17) it follows that a mathematical model can be generated for the cooling airflow:

$$(\dot{m}_{\text{cooling air}})_{\text{intake}} = k \cdot \alpha_q \cdot \frac{\tilde{T}_3 - \tilde{T}_w}{\Delta T_{\text{cool}}} \tag{18}$$

Equation (18) is scaled with a factor k for best agreement with the reference data (Fig. 3). To sum up, the cooling airflow can be expressed as a functional relationship

$$(\dot{m}_{\text{cooling air}})_{\text{intake}} = f_1(M, h, \alpha, \delta_T) \tag{19}$$

Similarly, models for the other engine components using air for cooling are developed. These components are intake flaps, $(\dot{m}_{\text{cooling air}})_{\text{intake flaps}} = f_2(M, h, \alpha)$; turbo engine, $(\dot{m}_{\text{cooling air}})_{\text{turbo}} = f_3(M, h, \alpha, \delta_T)$; and nozzle flaps, $(\dot{m}_{\text{cooling air}})_{\text{nozzle flaps}} = f_4(M, h, \alpha)$.

The overall airflow required for the addressed engine components is cooled in a heat exchanger with hydrogen fuel. The hydrogen fuel flow necessary for operating the heat exchanger can be expressed as a functional relationship

$$(\dot{m}_f)_{\text{indirect}} = K_{\text{exch}} \sum_{i=1}^{4} f_i(M, h, \alpha, \delta_T) = f_5(M, h, \alpha, \delta_T) \tag{20}$$

where K_{exch} represents a factor describing the heat exchange characteristics of the heat exchanger:

$$K_{\text{exch}} = f_{\text{exch}}(M, h) \tag{21}$$

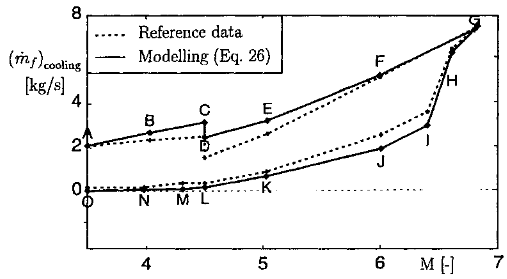

Fig. 4 Hydrogen fuel flow required for engine cooling.

The mathematical model developed for describing the fuel flow for direct ramduct cooling basically shows a functional relationship

$$(\dot{m}_f)_{\text{ramduct}} = f_6(M, h, \alpha, \delta_T) \tag{22}$$

The other engine components directly cooled with hydrogen fuel can be modeled in a similar way, yielding

$$(\dot{m}_f)_{\text{burner}} = f_7(M, h, \alpha, \delta_T) \tag{23}$$

$$(\dot{m}_f)_{\text{nozzle}} = f_8(M, h, \alpha, \delta_T) \tag{24}$$

The overall demand of hydrogen fuel for cooling the engine is the sum of the described contributions for the individual components. Thus,

$$(\dot{m}_f)_{\text{cooling}} = \sum_{i=5}^{8} f_i(M, h, \alpha, \delta_T) \tag{25}$$

As a result, the overall fuel flow for engine cooling can be described by a functional relationship

$$(\dot{m}_f)_{\text{cooling}} = f(M, h, \alpha, \delta_T) \tag{26}$$

This model is compared with data of the reference flight conditions in Fig. 4, which shows good agreement (with letters A–O used for cross reference with Fig. 2).

From Eq. (26) it follows that the cooling fuel flow depends on state and control variables. This relationship is incorporated in the mathematical and computational process for describing the dynamics of the system and its optimization.

Dynamics, Aerodynamics, and Propulsion Model

Modeling of vehicle dynamics is based on the equations of motion with reference to a spherical rotating Earth. The equations of motion read[20] (Fig. 5)

$$\begin{aligned}
\dot{V} &= (1/m)[T\cos(\alpha + \varepsilon_T) - D] - g\sin\gamma \\
&\quad + \omega_e^2 r\cos\Delta(\sin\gamma\cos\Delta - \cos\gamma\sin\Delta\cos\chi) \\
\dot{\gamma} &= (1/mV)[T\sin(\alpha + \varepsilon_T) + L]\cos\mu_a \\
&\quad + \cos\gamma[(V/r) - (g/V)] + 2\omega_e\cos\Delta\sin\chi \\
&\quad + \left(\omega_e^2 r/V\right)\cos\Delta(\cos\gamma\cos\Delta + \sin\gamma\sin\Delta\cos\chi) \\
\dot{\chi} &= \frac{1}{mV}[T\sin(\alpha + \varepsilon_T) + L]\frac{\sin\mu_a}{\cos\gamma} \\
&\quad + \frac{V}{r}\cos\gamma\sin\chi\tan\Delta + \frac{\omega_e^2 r}{V\cos\gamma}\sin\Delta\cos\Delta\sin\chi \\
&\quad - 2\omega_e(\tan\gamma\cos\Delta\cos\chi - \sin\Delta) \\
\dot{\Delta} &= \frac{V\cos\gamma\cos\chi}{r}, \qquad \dot{\Lambda} = \frac{V\cos\gamma\sin\chi}{r\cos\Delta} \\
\dot{h} &= V\sin\gamma, \qquad \dot{m} = -\dot{m}_f
\end{aligned} \tag{27}$$

where

$$r = r_e + h, \qquad g = g_0(r_e/r)^2 \tag{28}$$

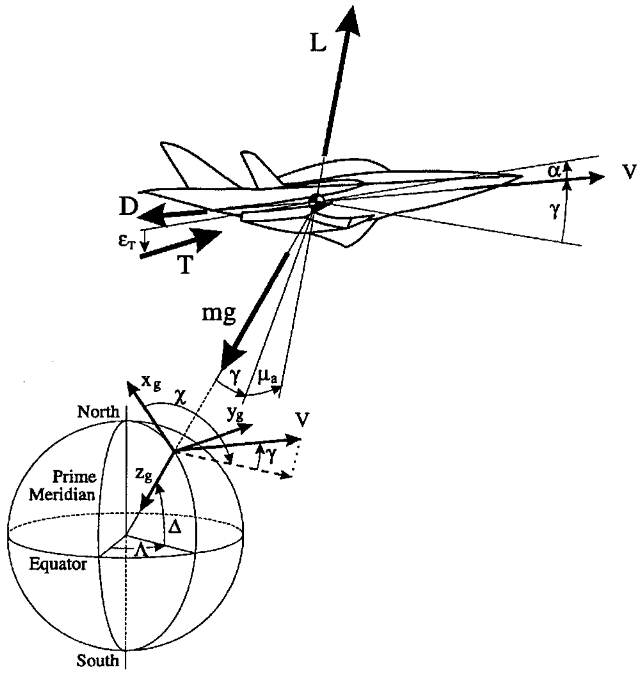

Fig. 5 Forces acting on the vehicle and coordinate systems.

A complex mathematical model involving multifunctional dependencies for describing aerodynamics and powerplant characteristics of the vehicle is applied. Emphasis is put on a realistic modeling. The aerodynamics model describes the lift and drag forces, which can be expressed as

$$L = C_L(\rho/2)V^2S, \qquad D = C_D(\rho/2)V^2S \tag{29}$$

where (Fig. 6)

$$C_L = C_L(\alpha, M), \qquad C_D = C_D(\alpha, M) \tag{30}$$

The powerplant consists of a turbo/ramjet combination. The mathematical model of this system describes rather complex thrust and fuel consumption characteristics, which depend on throttle setting, Mach number, and altitude. As a unique effect of hypersonic flight, angle of attack also exerts an influence. The mathematical model developed for describing the addressed effects reads

$$\begin{aligned} T &= \delta_T T^*(M, h) f_\alpha(M, \alpha) \\ (\dot{m}_f)_{\text{thrust}} &= \phi_f(\delta_T, M)\sigma^*(M, h)T^*(M, h) f_\alpha(M, \alpha) \end{aligned} \tag{31}$$

where T^* and σ^* denote reference values at stoichiometric combustion (Fig. 7). The engine model accounts for nonzero fuel consumption at idling and the possibility of overfueled combustion ($\phi_f > 1$) in the ramjet mode. Switching from turbo to ramjet operation and vice versa is modeled as a linear process between Mach 3 and 3.5. An afterburner mode is assumed to be used for turbojet operation at transonic and supersonic Mach numbers.

Control variables for the problem are angle of attack, throttle setting, and bank angle.

A realistic modeling involves constraints that exert a substantial effect on the performance of hypersonic vehicles. There are various reasons for the existence of constraints in hypersonic flight, such as aerodynamics and engine and structural limitations. The numerical values of the applied constraints are presented in Table 1. The boundary conditions for the return-to-base flight problem are shown in Table 2. The flight condition at the separation is considered to be fixed. The values chosen for the separation condition are based on an overall trajectory optimization for the combined flight of the carrier and the orbital stage.[14]

The mission type, which is of interest, consists of a trajectory from the base to a location for releasing the orbital stage and then back to the base again. The optimization problem is to minimize the overall fuel consumption $m_f(t_f)$, where t_f is the final time, which is

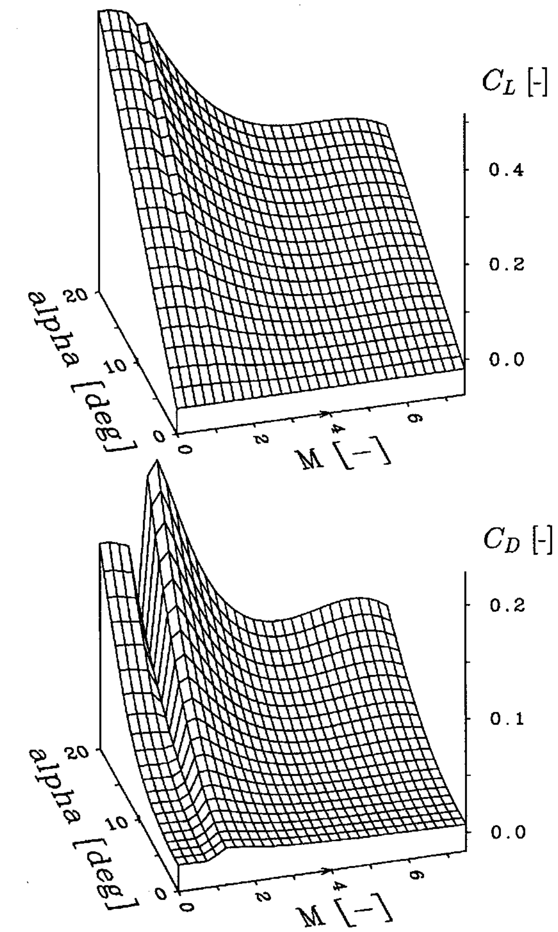

Fig. 6 Lift and drag coefficients.

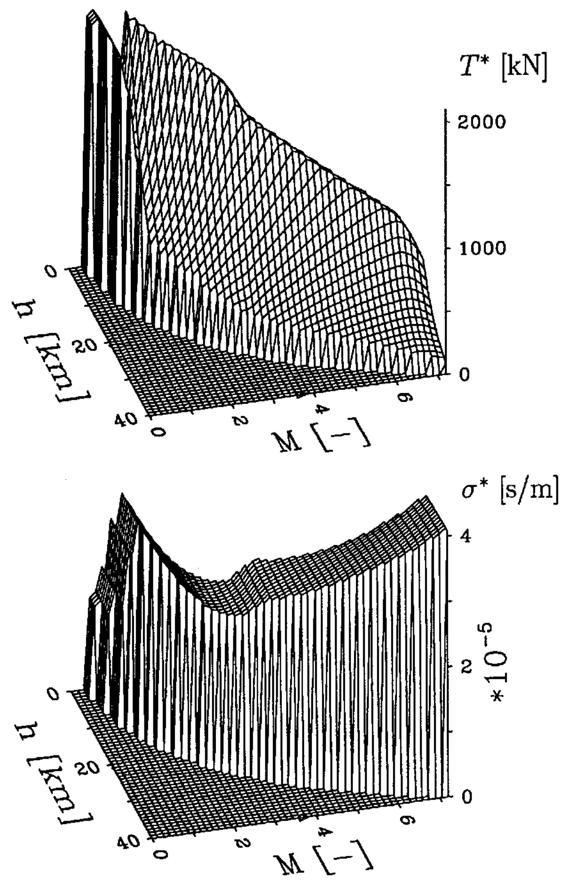

Fig. 7 Reference thrust T^* and thrust-related fuel consumption σ^*.

Table 1 Constraints

	Minimum	Maximum
α, deg	−1.5	20
ϕ_f (ram)	$\phi_f(M, \delta_{T\,\min})$	3
δ_T (turbo)	0	1
δ_T (ram)	0	$\delta_T(M, \phi_{f\,\max})$
$\lvert\mu_a\rvert$, deg	free	varied
$\bar{q}$, kPa	10	50
n	0	2

Table 2 Conditions at the initial and final points of the trajectory, and at separation

	$t = 0$	Separation	$t = t_f$
h, m	500	33,800	500
M	0.44	6.8	0.44
γ, deg	3	8.7	−3
χ, deg	——	90	——
Δ, deg	43.5	16.5	43.5
Λ, deg	0	——	0
m, kg	340,000	——	——

treated as free. A performance criterion for describing this problem may be formulated as

$$J = m_f(t_f) \tag{32}$$

For computing fuel consumption, the following relation is applied:

$$\dot{m}_f = \max[(\dot{m}_f)_{\text{thrust}}, (\dot{m}_f)_{\text{cooling}}] \tag{33}$$

This relation means that the fuel rate required for thrust production and the fuel rate required for cooling are compared, and the larger of the two is used for the trajectory optimization computations.

The optimization problem is to find the control histories that minimize the fuel consumed for the trajectory in question. 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 procedure, which was successfully applied, is a parameter optimization technique.[21]

Results

As a reference, an optimal trajectory is considered first where the fuel needed for producing thrust is minimized while the fuel for cooling the engine is not included in the optimization process. This is the classical approach to fuel-optimal flight-path optimization. Results are presented in Figs. 8–11. Figure 8 provides a perspective view for the type of trajectory in mind. The trajectory basically consists of a three-dimensional motion where the turn constitutes a significant part of the overall flight.

The histories of state and control variables are presented in Figs. 9 and 10. They show that the flight trajectory is rather unsteady and, thus, differs from usual aircraft cruise, which is practically a steady-state flight. The separation maneuver, which the carrier vehicle performs for releasing the orbital stage, is a highly dynamic motion.

Fuel consumption characteristics that are of particular concern are given in Fig. 11, which shows the fuel rates necessary for thrust production and for cooling. There are two phases during which more fuel is required for cooling than for thrust production. The difference, which represents a significant portion of the overall fuel consumption, is regarded as a loss since it is considered to be blown off.

When the fuel flow for engine cooling is included in the optimization process, as described in a previous section, a significant saving can be achieved. This is illustrated in Fig. 12, which shows fuel flow characteristics for the overall optimization case. As a main result, only a small phase after the separation maneuver exists, during which more fuel is required for engine cooling than for thrust production (i.e., where there is no need for more thrust). The loss has been reduced to an insignificant level. It may be of interest to note that there is some increase in the fuel required for thrust production, when compared with the thrust-fuel optimal case. However, the overall amount of fuel consumed is significantly smaller.

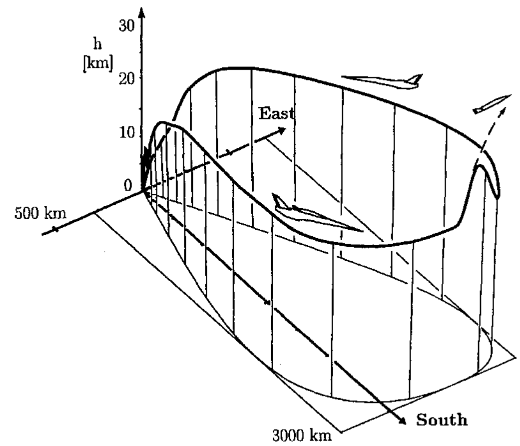

Fig. 8 Perspective presentation of a three-dimensional cruise trajectory for a two-stage hypersonic flight system.

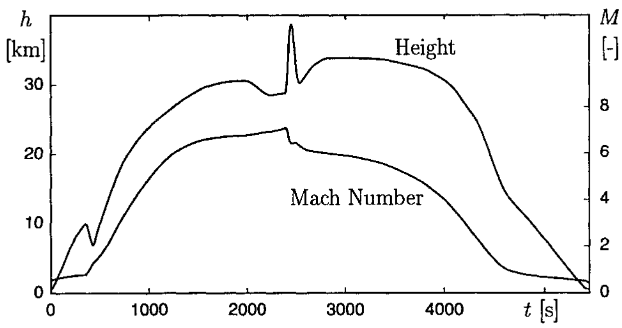

Fig. 9 State variables for thrust-fuel optimal trajectory.

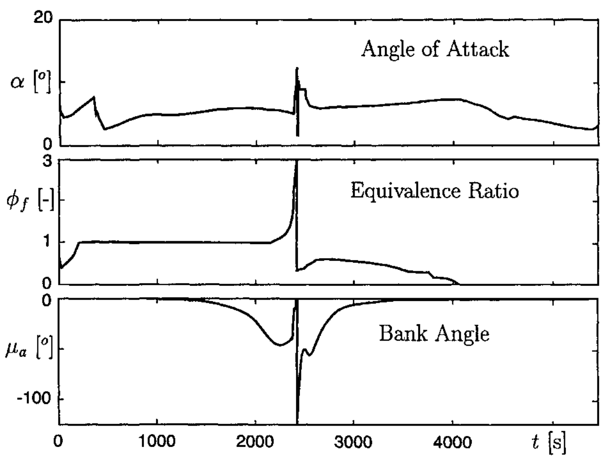

Fig. 10 Control variables for thrust-fuel optimal trajectory.

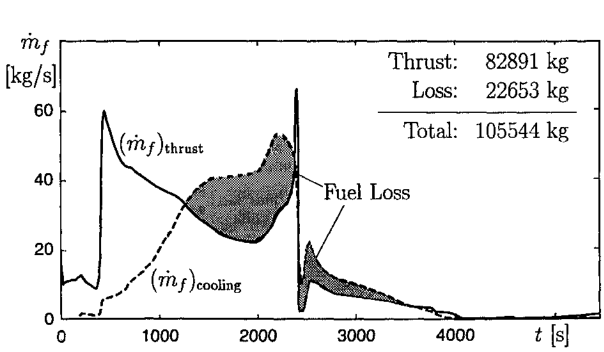

Fig. 11 Fuel flow for thrust-fuel optimal trajectory.

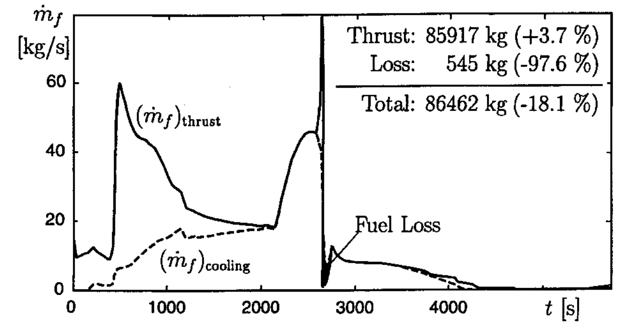

Fig. 12 Fuel flow for overall fuel optimal trajectory.

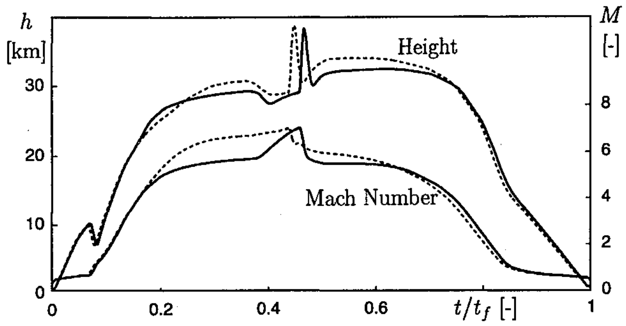

Fig. 13 State variables: ——, overall fuel optimal trajectory, t_f = 5680 s; and - - - -, thrust-fuel optimal trajectory, t_f = 5452 s.

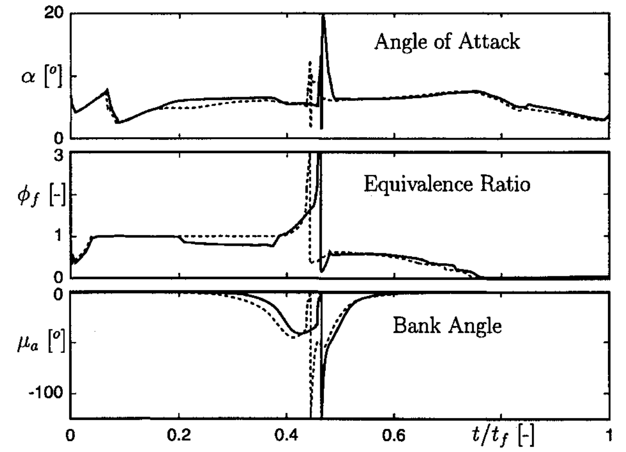

Fig. 14 Control variables: ——, overall fuel optimal trajectory, t_f = 5680 s; and - - - -, thrust-fuel optimal trajectory, t_f = 5452 s.

Further insight into fuel-optimal flight that includes the cooling requirements is provided in Figs. 13 and 14, which show the history of state and control variables. There is a significant reduction in speed during the phases where the thrust-fuel optimal trajectory shows a fuel loss. Since speed is a significant factor for cooling demands, a decrease in speed contributes to a reduction of cooling fuel flow.

Additional insight into the mechanism for reducing the cooling fuel flow by an appropriate control of the trajectory can be provided by considering the cooling demands of the various engine components. The trajectory for minimizing only the fuel flow for thrust production is considered first (Fig. 15). In the first phase (before separation), the directly cooled components show a higher fuel demand than the indirectly cooled components. In the second phase (after separation), the overall cooling fuel demand is significantly reduced. The directly cooled engine components again require the greater portion.

In the case of the overall fuel-optimal trajectory, the cooling demands show remarkable changes (Fig. 16). There is a significant reduction that primarily concerns the directly cooled engine components.

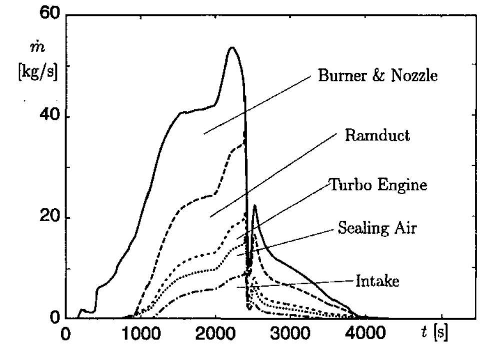

Fig. 15 Fuel flow for cooling the engine components for thrust-fuel optimal trajectory.

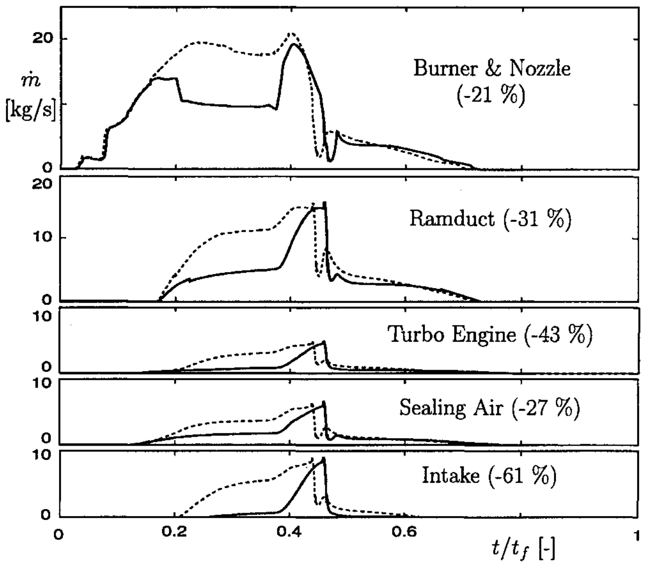

Fig. 16 Fuel flow for cooling the engine components: ——, overall fuel optimal trajectory, t_f = 5680 s; and - - - -, thrust-fuel optimal trajectory, t_f = 5452 s.

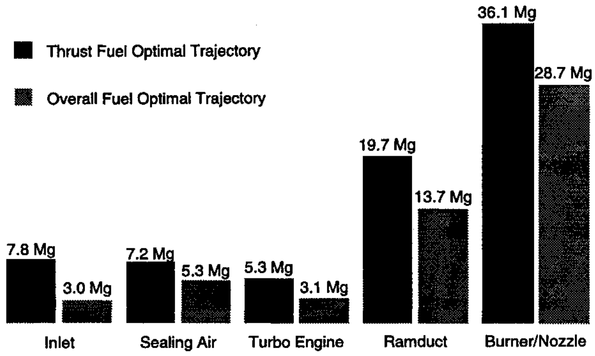

Fig. 17 Fuel consumption for cooling the engine components.

Further information is provided in Fig. 17, which shows the amount of fuel for each engine component. The major cooling demands are due to ramduct, burner, and nozzle. These engine components show the greatest savings when the cooling demands are included in the optimization process. Note that the greatest relative saving concerns the intake.

Conclusions

The fuel demand for cooling the engine of a hypersonic vehicle is considered and included in an overall trajectory optimization problem. A mathematical model for describing the cooling fuel demand of the individual engine components is developed and combined with the model for describing the dynamics of a hypersonic flight system. This system is a combination of a turbo/ramjet propelled carrier vehicle and a rocket driven orbital stage. A three-dimensional range flight is considered, during which the carrier vehicle releases the orbital stage at a specified location and then returns to its launching site.

It is shown that the overall fuel-optimal trajectory provides a significant saving compared with a classically optimized trajectory (for which only the fuel consumed for thrust production is minimized). In particular, it is shown that the optimization approach presented yields practically no flight phases of fuel loss where more fuel is required for cooling the engine than for producing thrust. An insight into the mechanism of reducing fuel flow for cooling is also presented.

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