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Comparison of Methane and Propane Rockets

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Introduction

THE potential benefits of hydrocarbon fuels and dual-fuel propulsion for future Earth-to-orbit vehicles have been shown in a number of studies.[1-6] Of the several hydrocarbon fuels considered, methane and propane are the favored candidates. In a previous study,[5] methane and propane were compared for use in a single-stage, Earth-to-orbit vehicle with dual-fuel rocket propulsion. The results indicated a significant advantage for propane, and further optimizations were conducted using this fuel. The purpose of this Note is to show how the comparison changes when both methane and propane vehicles are optimized. The analysis is the same as that described in Ref. 5.

Results

The results are shown in Figs. 1 and 2. In Fig. 1, the dry mass characteristics of vehicles with both fuels are shown. The flagged symbols at a hydrocarbon thrust fraction of 0.8 and a hydrocarbon propellant fraction of 0.79 are the same results shown in preliminary screening.[5] As the vehicles are optimized, the difference is reduced slightly, but the conclusions do not change. Propane is still the preferred hydrocarbon fuel to

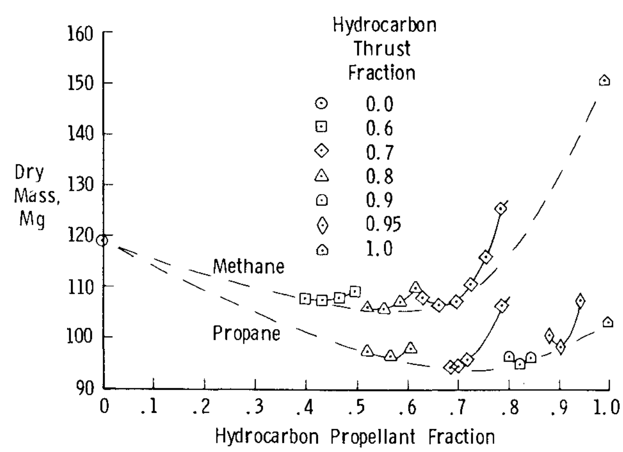

Fig. 1 Effect of fuel on vehicle dry-mass optimization.

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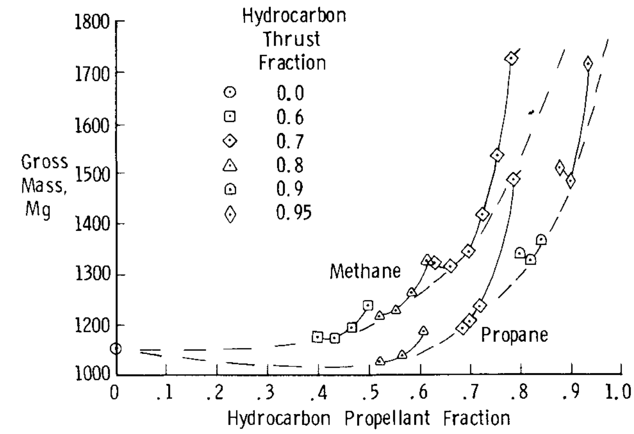

Fig. 2 Effect of fuel on vehicle gross-mass optimization.

minimize dry mass. Figure 2 shows the corresponding results for vehicle gross mass. In this figure, the difference between propane and methane decreases noticably when moving from the flagged symbols to the points with minimum gross mass. Propane is the fuel which minimizes gross mass. The results with methane indicate that a vehicle with only hydrogen fuel (a hydrocarbon fuel fraction of 0.0) has a gross mass as low as any dual-fuel vehicle.

Conclusion

When single-stage-to-orbit vehicles with methane and propane fueled rockets are optimized, the preferred fuel for minimum dry or gross mass is propane.

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Second-Order Roll Damping of Rolling Wings at Supersonic Speeds

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Nomenclature

b = wing span
B = $(M^2-1)^{1/2}$

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C = chord length of the midspan section of the wing
Cl_p = roll moment derivative due to rolling velocity
C_p = pressure coefficient
k = tangent of sweep-back angle
K = reduced frequency
M = freestream Mach number
n = total number of fins and/or wings
N = parameter $(\gamma+1)(M^2/2B^2)$
S = area of fin or wing panel
U = velocity of undisturbed flow
x,y,z = nondimensional coordinates referred to C
α = angle of attack
γ = ratio of specific heat
ω = rolling velocity, rad/s
Φ = nondimensional velocity perturbation potential referred to CU
ϕ_0 = linear part of Φ
ϕ_1 = second-order part of ϕ

Subscripts

x,y,z = partial derivatives with x,y,z

Introduction

THE lifting pressure distribution and damping in roll are calculated on the basis of a second-order theory similar to the one developed by Van Dyke[1,2] and Martin and Gerber.[3] But the airfoils treated in Ref. 3 are two-dimensional swept-back wings with an arbitrary symmetrical cross section. The estimate of the effect of thickness was made by increasing the Cl_p of the linearized theory by the same percentage as the thickness increases the Cl_p for an infinite wing.

The work by Van Dyke[1,2] indicates that second-order solutions of the partial differential equation of steady and unsteady supersonic flow can be obtained by the use of particular solutions. The partial differential equation considered in this Note is not the equation of steady supersonic flow, but it is similar, and we shall give the particular solution of the rolling term. Then, we get a real three-dimensional method to calculate the unsteady supersonic flow of a rolling wing by means of the second-order theory. Hence, the series of results from the unsteady second-order theory can be used for rolling wings.[4,5]

For engineering practice, we apply strip theory to study the problem of stability of a rolling wing of arbitrary planform and cross-sectional shape at a certain angle of attack in a supersonic stream. Calculating the roll damping of a wingbodytail rocket, this method can give a second-order correction to the calculation of the linearized theory, which is systematically summarized in Ref. 6. The results shown agree well with available experimental data.

Theory

The partial differential equation to be used is a special case of the three-dimensional time-dependent equation for the potential function of a nonviscous compressible fluid. Using a method of analysis applied to the three-dimensional problem,[7] it is found that in the sufficient condition: $(M\delta)^2<1$, $\mu^2 \ll \sigma^2$, and $K\sigma/\mu \leqslant$ or $=\mathrm{O}(\delta)$, where the dimensionles parameter δ is the thickness ratio of wing, σ is a measure of the lateral extent of the boundaries, and μ is another one to define the vertical extent of this neighborhood, the potential equation [Ref. 3, Eq.(4)] reduces to

$$\Phi_{zz}-B^2\Phi_{xx}=-2KM^2y\Phi_{xz}+M^2[(N-1)B^2\Phi_x^2+\Phi_y^2+\Phi_z^2]_x+M^2(\gamma-1)\Phi_x(\Phi_{zz}+\Phi_{yy}-B^2\Phi_{xx}) \tag{1}$$

To solve the nonlinear equation for Φ, we use an iteration procedure.[1,3,4] The first-order equation is

$$\phi_{0zz}-B^2\phi_{0xx}=0 \tag{2}$$

The second-order partial differential equation is

$$\phi_{1zz}-B^2\phi_{1xx}=-2KM^2y\phi_{0xy}+M^2[(N-1)B^2\phi_{0x}^2+\phi_{0y}^2+\phi_{0z}^2]_x \tag{3}$$

The equation of the surface of the body may be expressed by $z=H(x,y)$. The boundary conditions for the first-order solution [Ref. 6, Eqs. (9) and (10)] are $\phi_0(x,y,z)=0$ upstream of the wing and

$$\phi_{0z}(x,y,0)=-Ky+H_x(x,y) \tag{4}$$

Similarly, the boundary conditions for the second-order solution are $\phi_1(x,y,z)=0$ upstream of the wing and

$$\phi_{1z}(x,y,0)=\phi_{0x}(x,y,0)H_x(x,y)+\phi_{0y}(x,y,0)H_y(x,y)-\phi_{0zz}(x,y,0)H(x,y) \tag{5}$$

The coordinate axes are chosen as indicated in Fig. 1. The first-order solution is

$$\phi_0=-Ky(z-x/B)-(1/B)H(x-Bz,y) \tag{6}$$

The second-order potential function ϕ_1 must satisfy the nonhomogeneous equation (3). The particular integral of Eq.

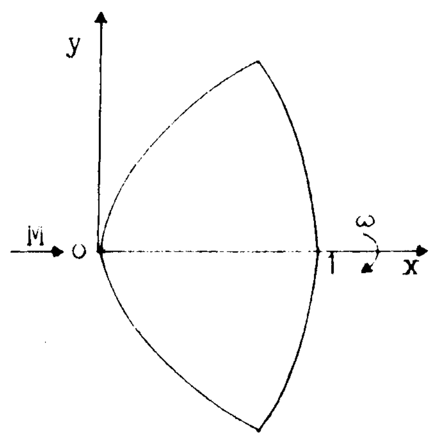

Fig. 1 Coordinates for rolling wing.

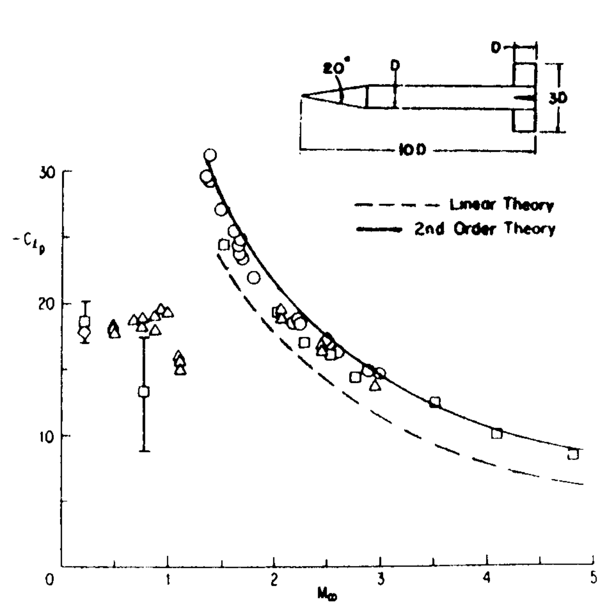

Fig. 2 Comparison of the second-order damping in roll coefficient with experimental values.[8]

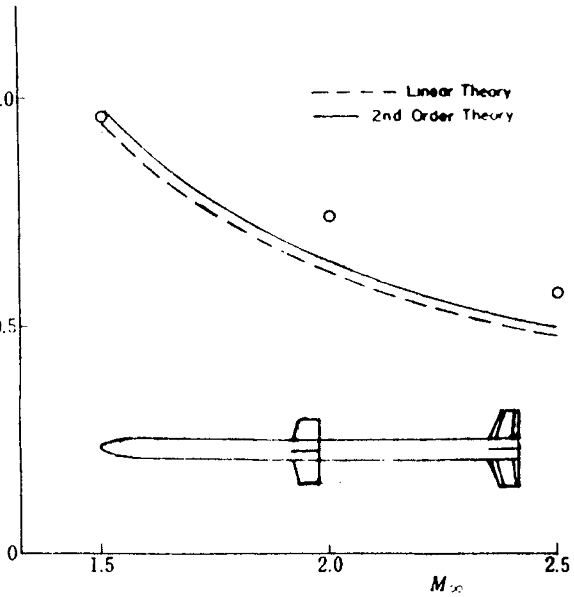

Fig. 3 Another comparison of the second-order damping moment with experimental values.[9]

(3) was found to be

$$\phi_1^R = -2Kzy\phi_{0x} + M^2\phi_0\phi_{0x} + M^2Nz\phi_{0x}\phi_{0z} \tag{7}$$

Now we can consider the solution ϕ_1 as the sum of the particular solution Eq. (7) and the complementary solution ϕ_1^c that satisfies Eq. (2) and subjects to the boundary condition

$$\phi_{1z}^c(x,y,0) = \phi_{1z}(x,y,0) - \phi_{1z}^R(x,y,0) \tag{8}$$

The damping in roll coefficient Cl_p can be written as

$$Cl_p = (2C^4/Sb^2) \iint_{\substack{\text{wing}\\ \text{area}}} y\partial\Delta C_p/\partial K \mathrm{d}x\mathrm{d}y \tag{9}$$

Calculations and Discussion

We have calculated the roll damping moments of two rocket models using the linearized and second-order theories. The effects of wingbody interference are estimated by the linearized theory.[6]

Figure 2 is a study of the effect of freestream Mach number on the damping moment of the basic finner in roll. The cross section of its fins is an 8% thick wedge. It presents the values of the Cl_p estimated in this manner as compared to the experimental values presented in Ref. 8. Figure 2 indicates that the agreement between theory and experiment has been improved considerably.

Figure 3 shows the roll damping moment of a rocket model. The calculated results of the linearized and second-order theories were compared with the experimental values presented in Ref. 9. Because of the small thickness of the wings (only about 3%), the use of the second-order theory did not result in much improvement.

Conclusions

Because an exact particular solution Eq. (7) has been found in this Note, a real three-dimensional second-order theory may be used to study the problem of stability of a rolling wing of an arbitrary planform and cross-sectional shape in a supersonic stream. The results show that the second-order theory can give a slight improvement in the roll damping of a wingbody rocket based on the results of the linearized theory.

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Dynamic Stability Tests on Finned Bodies at Hypersonic Mach Number

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Nomenclature

C_m	= pitching moment coefficient = $M_{\text{aero}}/\frac{1}{2}\rho V^2 Sd$
$C_{m\alpha}$	= pitching aerodynamic stiffness derivative = $\partial C_m/\partial\alpha$
$C_{m\dot\alpha}$	= pitching moment derivative due to rate of change of angle of attack = $\partial C_m/\partial(\dot\alpha d/2V)$
C_{mq}	= pitching moment derivative due to rate of pitching = $\partial C_m/\partial(qd/2V)$
d	= model centerbody diameter
L	= total model length, nose to cylinder base
M	= freestream Mach number
M_{aero}	= aerodynamic moment
q	= pitch rate of oscillating model
Re_d	= Reynolds number based on cylinder body diameter
S	= cylinder area = $(\pi d^2/4)$
V	= flow speed
X_{cg}	= axial distance from the nose tip to the oscillation axis
α	= angle of attack, deg
$\dot\alpha$	= rate of change of α w.r.t. time
ρ	= flow density

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