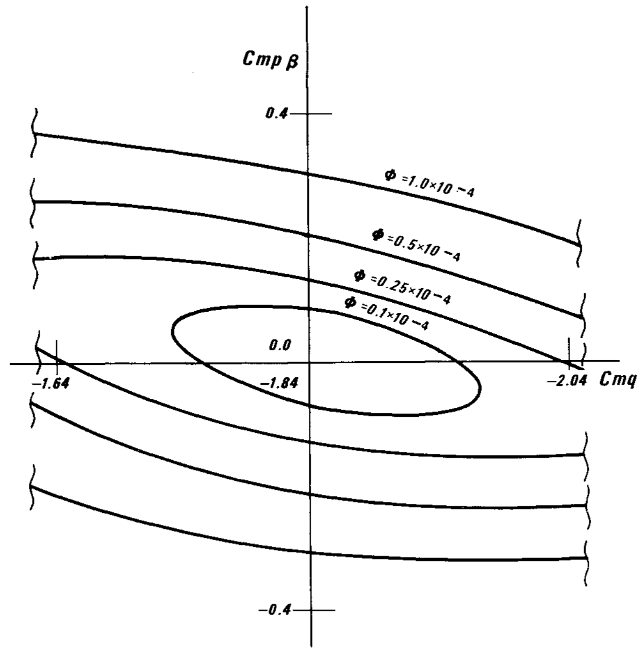

Fig. 1c Contours of constant Φ_c for $C_m\alpha = -0.12$ for re-entry vehicle, example.

$$\bar{\alpha} = (A_1 \cos\omega_1 t + B_1 \sin\omega_1 t)e^{\lambda_1 t} + (A_2 \cos\omega_2 t + B_2 \sin\omega_2 t)e^{\lambda_2 t}$$

$$\bar{\beta} = (B_1 \cos\omega_1 t - A_1 \sin\omega_1 t)e^{\lambda_1 t} + (B_2 \cos\omega_2 t - A_2 \sin\omega_2 t)e^{\lambda_2 t}$$

and with $\zeta \equiv pI_x/2I$,

$$\lambda_{1,2} = \left(\frac{M_q + M_{\dot{\alpha}}}{2I}\right)\left[1 \pm \frac{\zeta}{(\zeta^2 - M_\alpha/I)^{1/2}}\right] \pm \frac{\zeta M_{p\beta}/I_x}{(\zeta^2 - M_\alpha/I)^{1/2}}$$

and $\omega_{1,2} = \zeta \pm (\zeta^2 - M_\alpha/I)^{1/2}$.

The coefficients to be determined are $C_{m\alpha}$, C_{mq}, and $C_{mp\beta}$ where $C_{m\dot{\alpha}}$ is incorporated into C_{mq}.

As a particular example a small reentry vehicle entering the Earth's atmosphere is presented with the q' curve from Ref. 2. The aerodynamic coefficients of the vehicle are $C_{m\alpha} = -0.12$, $C_{mq} = -1.84$, $C_{mp\beta} = 0$. The time interval was 2.5 sec, which represents $\sim$3 cycles of motion. The initial value of α was 2° and p was 18.8 rad/sec. A case was run with $C_{m\alpha}$ and C_{mq} assumed known and $C_{mp\beta}$ to be determined. The sensitivities of Φ to coefficient variations for this model can be seen in Fig. 1. Simulated random noise was added to the actual solution (shown in Fig. 2), and the coefficient that gave a minimum Φ was determined. The effect of the noise is to increase $\Phi_{\min}$ from 0 to 0.0016 and $C_{mp\beta}$ from 0 to 0.6. The significance of such a difference in coefficient value must be judged in terms of the subsequent application of the results in other regions where the solution is sensitive to the coefficient value (e.g., resonant motion).

Another example is taken directly from Ref. 1. The vehicle is a finned rocket with $C_{mq} = -1400$ and $C_{mp\beta} = 0$. The data was generated directly from Eq. (1) using an α dependent $C_{m\alpha} = -31 + \epsilon$, where $\epsilon = 0.01(10 - |\xi|)^2$ over an interval of $\sim$8 cycles with $\xi = 10°$ at t_1. The minimum Φ of 0.448 was achieved with the computed parameters $C_{m\alpha} = -29.8$ and $C_{mp\beta} = -40$, where C_{mq} was held constant at -1400. The average amplitude error was only 0.238°, or 2.4%. It should be noted that with a Φ of this size for this example the values calculated for $C_{m\alpha}$ and $C_{mp\beta}$ could be further from the actual values depending on the error form. The sensitivity of Φ to coefficient variation is similar to the previous examples. The data used in the examples have much less distortion than typical experimental data.

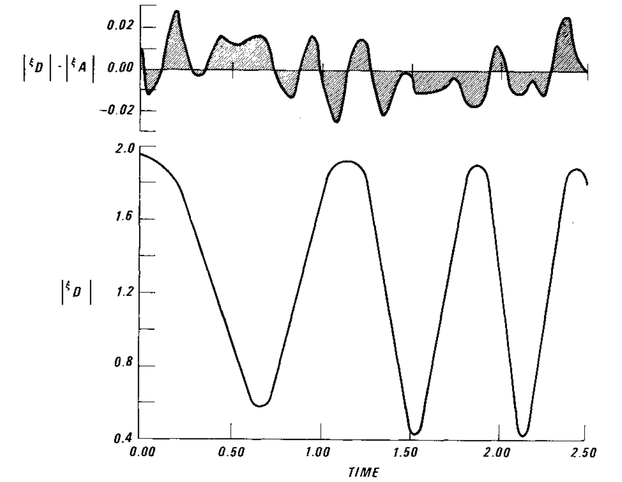

Fig. 2 Plot of data in degrees and error in degrees vs time.

Conclusions

Use of a sensitivity analysis of a performance index Φ has shown that it may be difficult to determine aerodynamic coefficients accurately from experimental data because of the insensitivity of the solution to variations in these parameters. In addition, techniques which obtain coefficient values should be carefully evaluated to make certain that they do not use one insensitive parameter in the determination of another parameter. A perfect fit ($\Phi = 0$) can never be achieved because of variation in the data from the assumed solution. These variations or errors come from data noise, modeling errors, and time-varying parameters. Experimental data should be obtained in regions of maximum coefficient sensitivity.

References

[1] Vaughn, H. R., "A Detailed Development of the Tricyclic Theory," SC-M-67-2933, Feb. 1968, Sandia Labs., Albuquerque, N. Mex.

[2] Vaughn, H. R., "Spin-up and Roll Reversal of Reentry Vehicles," SCRR 68-219, May 1968, Sandia Laboratories, Albuquerque, N. Mex.

[3] Unruh, D. R., "Determination of Optimal Parameters for Dynamical Systems," Ph.D. dissertation, 1970, Oklahoma State University, Stillwater, Okla.

Aerodynamic Viscous Effects on a New Space Shuttle Vehicle

M. I. Kussoy,* C. C. Horstman,† and D. A. Stewart‡
NASA Ames Research Center, Moffett Field, Calif.

Introduction

A NEW configuration for a reusable space shuttle vehicle (SSV) was presented and discussed by Faget.[1] Plan views of this vehicle, taken from Ref. 1, are shown in Fig. 1. The short-range shuttle will spend between 150 and 200 sec of flight time descending from the entry altitude of 400 K to 300 K ft at a Mach number around 25. Previous results for blunt lifting reentry bodies,[2] such as the Apollo, have indicated significant viscous effects on the forces over these

Received July 8, 1970.
* Research Scientist. Member AIAA.
† Research Scientist. Associate Fellow AIAA.
‡ Research Scientist.

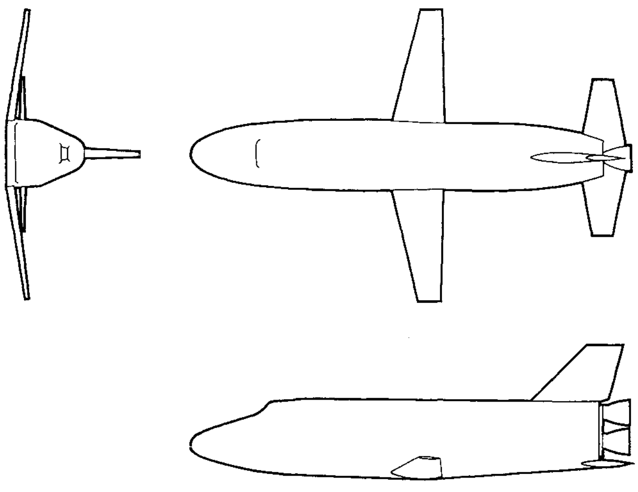

Fig. 1 Plan views of MSC-orbiter configuration for present tests.

bodies flying at these altitudes and speeds. At present, it is extremely difficult to theoretically predict the influence of viscous effects on the aerodynamic forces of complicated shapes. So experimental measurements are necessary. The purpose of this note is to present some experimental-force data for a typical SSV design in the hypersonic viscous flow regime, and assess its effects on an entry trajectory.

Experimental Procedure

This investigation was conducted in the Ames 42-in. shock tunnel, the calibration and operation of which has been

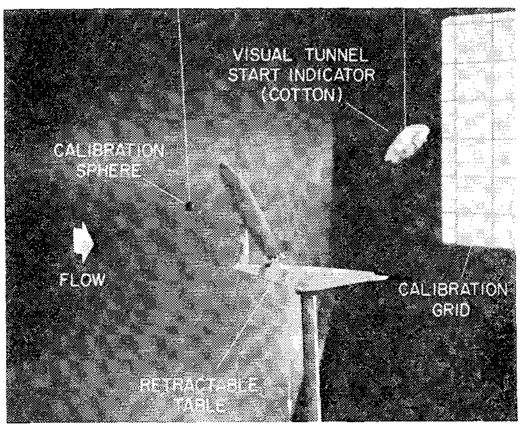

Fig. 2 Typical test-section arrangement for SSV free-flight tests in NASA Ames 42-in. shock tunnel.

previously discussed.[3] A free-flight technique similar to that of Geiger[4] was used. A typical test setup is illustrated in Fig. 2. A 4-in.-long balsa wood model of the vehicle rests on a table at a preset angle of attack (α). The table is retracted 2 msec before tunnel start, leaving the model suspended at the

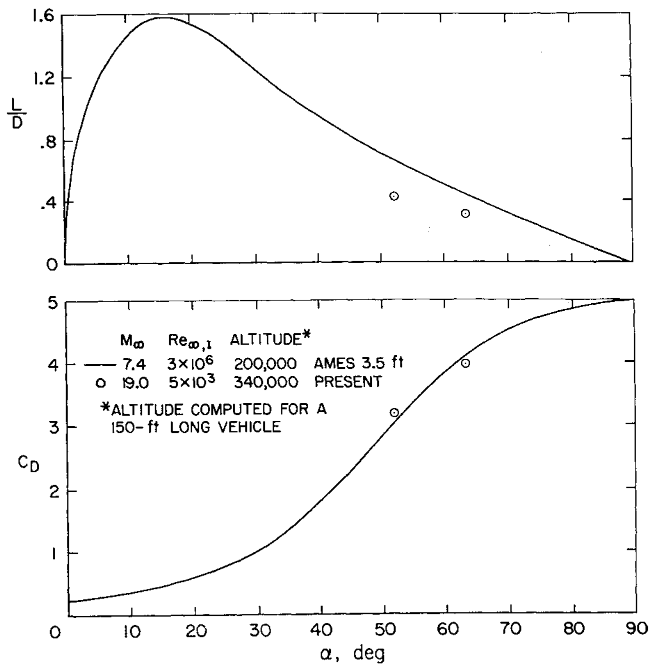

Fig. 3 C_D and L/D vs α for MSC Space Shuttle orbiter.

initial α. A cotton wad provides a visual indication of tunnel start. A calibrating sphere of known drag coefficient (C_D) is flown to the side of the test body to obtain the tunnel dynamic pressure. When properly trimmed, the model will remain at the initial α throughout its flight. High-speed motion pictures are taken, and using the background grid as a reference, lift and drag are obtained from the horizontal and vertical model accelerations determined from the time histories of model displacement. Tests were made in air at Mach number 19, trim angle of attack of 52 and 65°, and Reynolds number based on length, $Re_{\infty,L}$ of 5000 (altitude ~340,000 ft for a 150 ft vehicle).

Results and Discussion

The measurements are shown in Fig. 3, plotted as lift-drag ratio (L/D) and C_D vs α. (The reference area used in C_D was the total projected wing area.) These measurements are compared with sting-mounted data obtained by John Axelson of the Ames Research Center in the Ames 3.5-ft wind-tunnel, at $M_\infty = 7.4$, and $Re_{\infty,L} = 3 \times 10^6$ (altitude ~200,000 ft). The drag results agree; however, the present L/D ratios, and hence lift coefficients, are 30–40% lower than the results of Axelson. This reduction at low Reynolds number is attributed to increased skin friction, which significantly decreases the lift vector. C_D is not measurably affected, because of the high-model angle of attack. These results are consistent with those of Ref. 2, where similar decreases in lift at low Reynolds numbers were measured on the Apollo and Gemini vehicles.

To assess the effects of these significant decreases in L/D with decreasing Reynolds number, several typical entry

Table 1 SSV entry trajectories: (M/C_DA) slugs/ft² = 0.70

					Maximum heating conditions		
lift/drag	Roll angle, deg	Maximum g load	Longitudinal range, naut miles	Cross-range, naut miles	Heating rate for a 1 ft sphere, Btu/ft² °R	altitude, ft	Re_∞/ft
0.50	45°	2.51	2759	227	58.1	257,000	4691
0.26 to 0.50[a]	45°	2.50	2698	215	60.0	256,000	5000
0.50	0°	1.92	3631	...	51.6	263,000	3612
0.26 to 0.50[a]	0°	1.95	3480	...	53.8	262,000	3924

[a] See text for explanation of variation.

trajectories were calculated. These computations used identical sets of initial conditions, and a trim angle of 60° with either continuum aerodynamics or low Reynolds number aerodynamics. For the continuum case, a lift-drag ratio of 0.5 was used for the entire trajectory; for the low Reynolds number case, the L/D was varied linearly from 0.26 at 400,000 ft altitude to 0.5 at 250,000 ft, and then kept constant for the remaining trajectory. The calculations were made for an azimuth angle of 32°, an Earth reference velocity of 25, 134 fps, and the lift vector in the positive vertical position (roll angle $\phi = 0°$) or inclined (roll angle $\phi = 45°$). The results of these computations are presented in Table 1. In all cases, the low density effects produce only minor changes in range, g loads, and maximum heating conditions.

In summary, lift and drag forces were measured on a typical SSV design in the hypersonic flow regime using a free-flight technique. Significant decreases in lift due to low Reynolds number effects were measured for this space shuttle vehicle. However, calculated entry trajectories are essentially the same whether or not these low density effects are taken into account.

References

[1] Faget, M., "Space Shuttle: A New Configuration," *Astronautics and Aeronautics*, Vol. 8, No. 1, Jan. 1970, p. 52.

[2] Horstman, C. C. and Kussoy, M. I., "Free-Flight Measurements of Aerodynamic Viscous Effects on Lifting Re-Entry Bodies," *Journal of Spacecraft and Rockets*, Vol. 4, No. 8, Aug. 1967, p. 1064.

[3] Kussoy, M. I. and Horstman, C. C., "Cone Drag in Rarefied Hypersonic Flow," *AIAA Journal*, Vol. 8, No. 2, Feb. 1970, p. 315.

[4] Geiger, R. E., "Slender Cone, Zero Angle of Attack Drag in Continuum and Noncontinuum Flow," AIAA Paper 69-711, San Francisco, Calif., 1969.

Drag of Vane-Type Vortex Generators in Compressible Flow

J. C. Westkaemper* and Jerry W. Whitten†
The University of Texas at Austin, Austin, Texas

Nomenclature

AR = aspect ratio
b = vortex generator span
b_T = generator trailing edge span
C_D = drag coefficient
C_f = skin-friction coefficient
C_p = pressure coefficient
c = generator chord length
D = drag force
d = leading edge thickness
l = wedge centerline length
M = Mach number
q = dynamic pressure
S = reference area
x = distance along wedge axis
θ = leading-edge sweep angle
α = angle of attack
δ = nominal boundary layer thickness
ϵ = rake angle
ϕ = wedge total angle
γ = ratio of specific heats

Subscripts

a = boundary-layer average
g = vortex generator
i = incompressible
n = leading-edge nose
w = wedge
∞ = freestream

Received June 11, 1970; revision received July 15, 1970. This work was sponsored by the Naval Air Systems Command, under Subcontract 181471 with the Applied Physics Laboratory of The Johns Hopkins University.

* Research Engineer and Associate Professor of Aerospace Engineering, Applied Research Laboratory. Associate Fellow AIAA.

† Research Engineer, Applied Research Laboratory; now with the Boeing Company, Seattle, Wash.

Introduction

THE use of vane-type vortex generators in subsonic flows is widespread. Boundary-layer separation can be delayed or prevented on aircraft wings and stabilizers, in jet-engine inlets and similar diffusers. There has been little use of the generators in the supersonic speed regime, however. This may be in part the result of the lack of information on their performance in compressible flow. Reference 1 reports the results of flight tests of generators used to control separation on a flapped wing on a missile, primarily at transonic speeds. Limited data at Mach numbers up to 1.35 indicated that some configurations improved and some degraded the wing performance. No systematic investigation was made, however. In Ref. 2, in tests of a supersonic inlet, the vortex generators were in supersonic flow for some test conditions; here they "continued to function properly" in improving pressure recovery. The maximum Mach number at the generator location was 1.45. Again no systematic study was made of generator configurations.

The present work is part of a wind-tunnel test program to determine the usefulness of vortex generators in preventing separation of turbulent, compressible boundary layers. This Note covers an investigation of the drag of vane-type generators. Vane planforms were limited to simple shapes, since no data were available to aid in predicting an optimum shape.

Apparatus and Procedure

The tests were made in a blowdown-type wind tunnel having test section dimensions of 6 × 7 in. with fixed nozzle blocks giving a test Mach number of 4.9. A stagnation temperature of 660 ± 5° R was used to preclude moisture condensation and air liquefaction. A stagnation pressure of 260 ± 5 psig was employed, resulting in an average freestream Reynolds number of 1.08 million/in.

The vortex generators were mounted on the flat plate shown in Fig. 1. This aluminum plate was 6 in. wide, 17.5 in. long, and 0.875 in. thick. The leading edge had a wedge angle of 14°. A half-in.-wide strip of 80-grit emery cloth was cemented to the test surface, $\frac{1}{2}$ in. downstream of the leading edge, to serve as a boundary-layer trip. This resulted in a

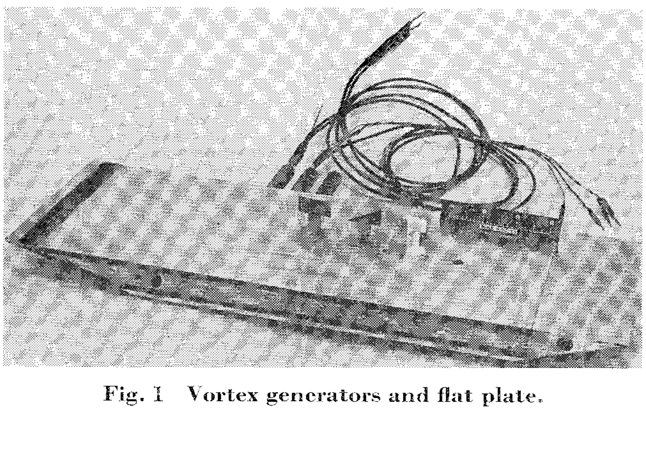

Fig. 1 Vortex generators and flat plate.