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SURFACE-FLOW, PRESSURE, AND HEAT-TRANSFER STUDIES ON TWO CONICAL DELTA WINGS AT A MACH NUMBER OF 6

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16. Abstract

An experimental investigation of the surface flow, pressures, and heat transfer on two conical delta wings having attached leading-edge shocks has been conducted at a Mach number of 6. The angle of attack was varied between 0° and 12°. The pressure data were compared with predictions obtained by the method-of-lines technique, and the heating data were compared with the heating levels predicted by the Spalding-Chi method.

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SUMMARY

An experimental investigation of the surface flow, pressures, and heat transfer on two conical delta wings having attached leading-edge shocks has been conducted at a Mach number of 6. The two 60° swept delta wings utilized a rhombic and a circular-arc cross section, respectively. The angle of attack was varied between 0° and 12°. The windward-surface flow on both delta wings was nearly two dimensional at the angles of attack tested; vortices were found on the leeward surfaces of the delta wings at angles of attack of 5° and 10°. The method-of-lines technique predicted the windward-surface pressures with the better agreement at the higher angles of attack where the viscous effects were small. Turbulent flow existed on most of the delta-wing surface. The Spalding-Chi method predicted the levels of turbulent heating reasonably well on the windward surfaces.

INTRODUCTION

Until recently the flow fields about conical bodies at incidence in supersonic and hypersonic flows had been solved only for the simplest cases or after linearization or other approximations to the governing equations. Even though the nature of the conical flow permitted the problem to be reduced from three to two dimensions, few exact solutions had been obtained. Recent advances in computer technology have spurred the development of numerical solutions to the full nonlinear equations to the extent that solutions are now becoming available for general conical-flow problems. For example, the method of lines originally reported in references 1 to 3 and modified in references 4 and 5 has successfully provided solutions to circular and elliptic cones at incidence and to the compression side of several conical delta wings with attached leading-edge shocks. Experimental data are now required to validate the numerical conical-flow solutions.

An abundance of experimental data exists for delta wings at supersonic and hypersonic speeds (e.g., refs. 6 to 12); however, these data with the exception of reference 12 are generated on configurations which have detached leading-edge shocks. Therefore, an experimental investigation of the flow field over two conical delta wings having attached leading-edge shocks was conducted at a Mach number of 6. The pressure data were compared with predictions obtained by the method-of-lines technique. Heat-transfer data were compared with heating levels predicted by the Spalding-Chi method for turbulent heating (ref. 13). Flow-visualization studies were conducted to describe the flow field and to help in the interpretation of the heating and pressure data.

The experimental investigation was conducted at a free-stream total pressure and total temperature of 2.9 MN/m^2 and 492 K, respectively (free-stream Reynolds number per meter of 2.33×10^7). The two 60^o swept delta wings utilized sharp leading edges and had a rhombic and a circular-arc cross section, respectively. The angle of attack was varied between 0^o and 12^o.

SYMBOLS

L	model length, centimeters
$N_{St,\infty}$	Stanton number based on free-stream conditions
p	static pressure
p_∞	free-stream static pressure
R	radial distance from apex, centimeters
R_∞	free-stream Reynolds number per meter
x	longitudinal distance from leading edge parallel with model center line, centimeters
x_{vo}	virtual origin, distance along surface to end of transition as determined from heating data, centimeters
y,z	lateral and vertical coordinates, respectively, centimeters (see fig. 1)
α	nominal angle of attack of model center line, degrees
α_T	actual angle of attack of model center line, degrees
ϕ	conical apex angle of body measured from model center line in horizontal plane, degrees (see fig. 1)

APPARATUS AND METHODS

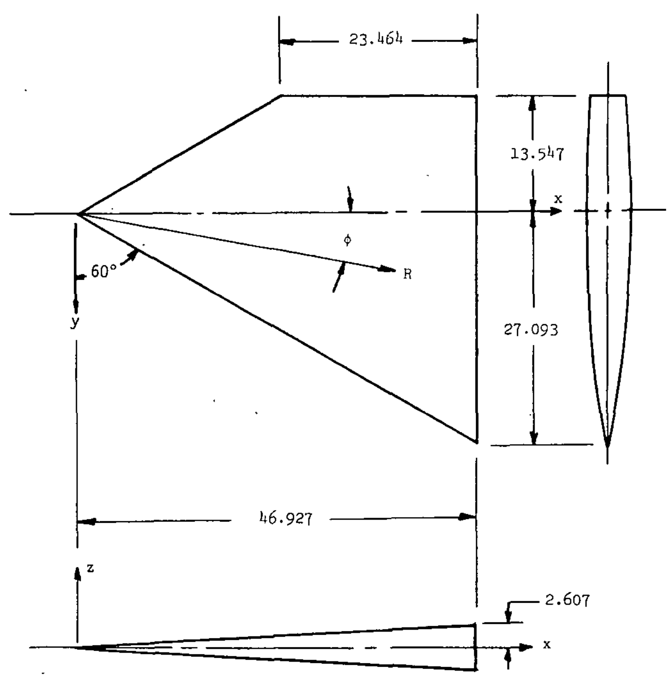
Wind Tunnel

The test program was conducted in the Langley 20-inch Mach 6 tunnel. A complete description of this facility and its calibration is given in the appendix of reference 14.

Models, Instrumentation, and Test Methods

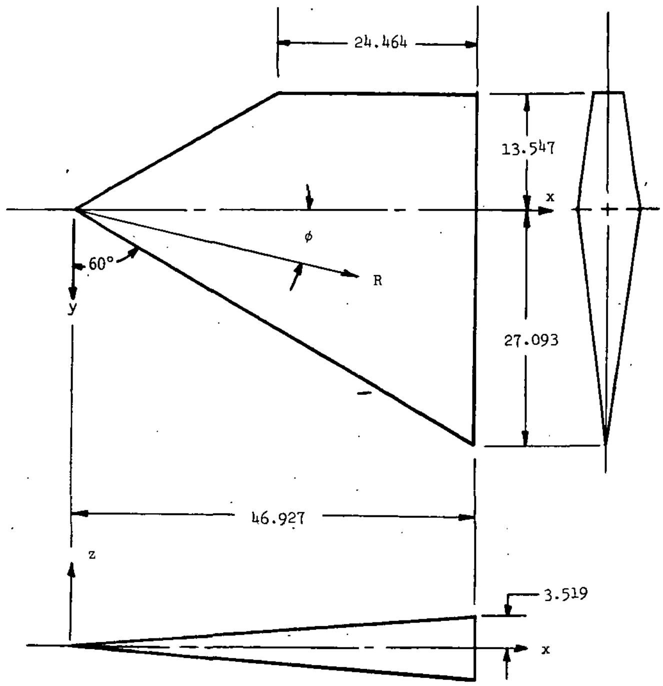
Two sting-mounted 60° swept delta wings were investigated, one with a rhombic cross section and the other with a circular-arc cross section. The leading edges of both wings were razor sharp with the sweep and opening wedge angles selected to insure an attached leading-edge shock for the test conditions. The two wings were designed so that at any longitudinal station the cross-sectional areas would be equal.

The heat-transfer and pressure models used in the study are shown with the important dimensions in figure 1. The heat-transfer and pressure models were clipped to enable the models to be injected into the test section. The oil-flow models were smaller (L = 30.80 cm) than the heat-transfer and pressure models and were not clipped. Oil-flow studies, not shown herein, indicated that clipping of the wing had no effect on the surface data on the model center line and on the unclipped half of the wing, provided that the leading-edge shock remained attached. The larger models were made of stainless steel



(a) Circular-arc delta wing. Equation of wing:
$y = \pm\sqrt{0.3333x^2 - 5.9444xz - z^2}$.

Figure 1.- Drawings and dimensions (cm) of models.



(b) Rhombic delta wing. Equation of wing:
$y = \pm(0.5774x - 7.6987z)$.

Figure 1.- Concluded.

and were equipped with both pressure orifices and iron-constantan thermocouples. (See tables I and II for location of instrumentation.) The oil-flow models were made of aluminum.

Pressure and heat-transfer data were obtained during different test runs with the desired angle of attack set prior to each run. Local surface pressures were obtained from pressure orifices connected to multirange capacitance-type transducers. The heat-transfer data were obtained from iron-constantan thermocouples located approximately 0.06 cm below the surface of the thin-skin model. The electrical outputs from both the transducers and the thermocouples were recorded on magnetic tape and processed by an electronic data-processing system.

Oil-flow studies were conducted to examine the surface-flow direction and relative shear on the windward and leeward surfaces. A mixture of silicon oil and lampblack was distributed in random dots of varying sizes on the entire model. Photographs were then taken of the model after each run.

Limited vapor-screen tests were conducted on the circular-arc delta-wing to study the cross section of the leeward-surface flow. In these tests, the free-stream total temperature was lowered to a value which allowed the air to condense (294 K ($\approx 530^{\circ}$ R)). A high-intensity screen of light, which was reflected and refracted by the gas and liquid molecules in the condensed air, was passed across the test section. Photographs were taken of this light screen during the test run.

Test Conditions

The tests were conducted at a free-stream Mach number of 6 with a free-stream total pressure and total temperature of 2.9 MN/m^2 and 492 K, respectively. The free-stream Reynolds number per meter was 2.33×10^7. The angle of attack was varied between 0° and 12° for the pressure tests and flow-visualization tests; the nominal angle of attack was varied between 0° and 8° for the heat-transfer tests.

Data Reduction

The thermocouple data were reduced to Stanton numbers by using methods similar to those described in reference 15 and by assuming an appropriate recovery factor (0.860 for laminar flow and 0.895 for turbulent boundary-layer conditions). The local Mach numbers used in determining the recovery temperatures were obtained from the ratio of the measured surface static pressure to free-stream total pressure. The pressure data were normalized by the free-stream static pressure.

Accuracy

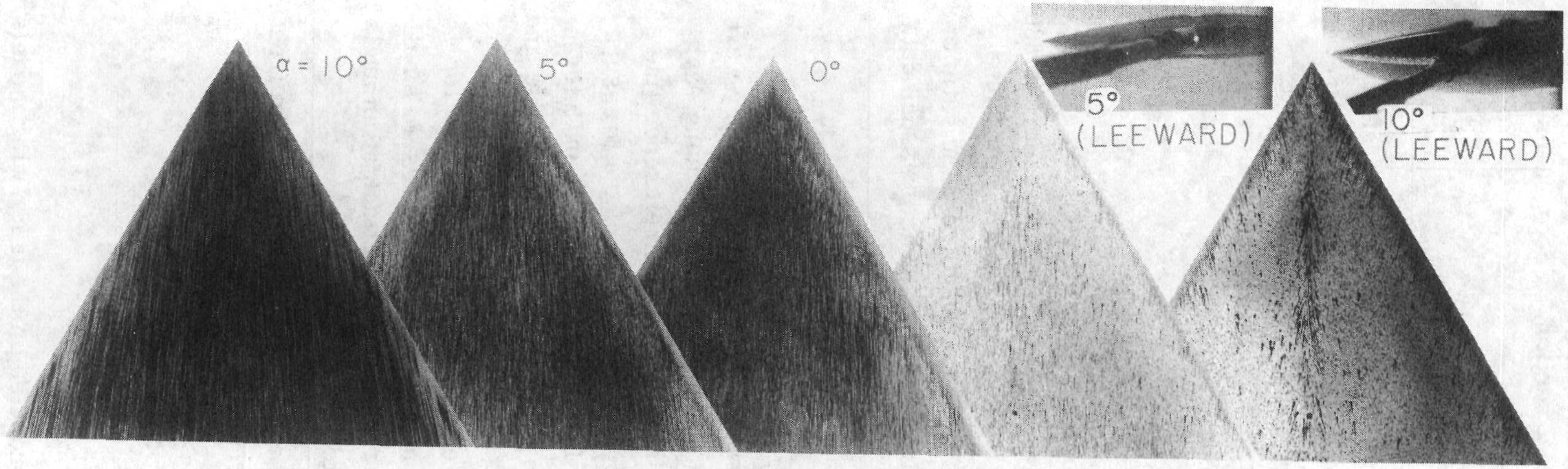
The true angles of attack for the pressure data that are compared with theory were measured on a comparator from photographs of the model taken before and during each test run and were accurate within $\pm 0.1^\circ$. A known zero-angle-of-attack reference line was marked on the tunnel floor. The angles of attack for the remainder of the pressure data were calculated theoretically by using the method-of-lines technique, taking into account the agreement between experiment and theory for the known angles of attack, and were estimated to be accurate within $\pm 0.2^\circ$. The quoted angles of attack for the heat-transfer and flow-visualization studies are nominal angles and were not corrected for wind loads on the models since the free-stream Stanton number is not sensitive to small errors in angle of attack. (That is, an unrealistic 2° error in angle of attack at an angle of attack of 5° results in only a 6-percent error in Stanton number.)

The pressure transducers used to measure the local static pressures and the free-stream total pressures were accurate to within 0.25 percent of the full-scale range.

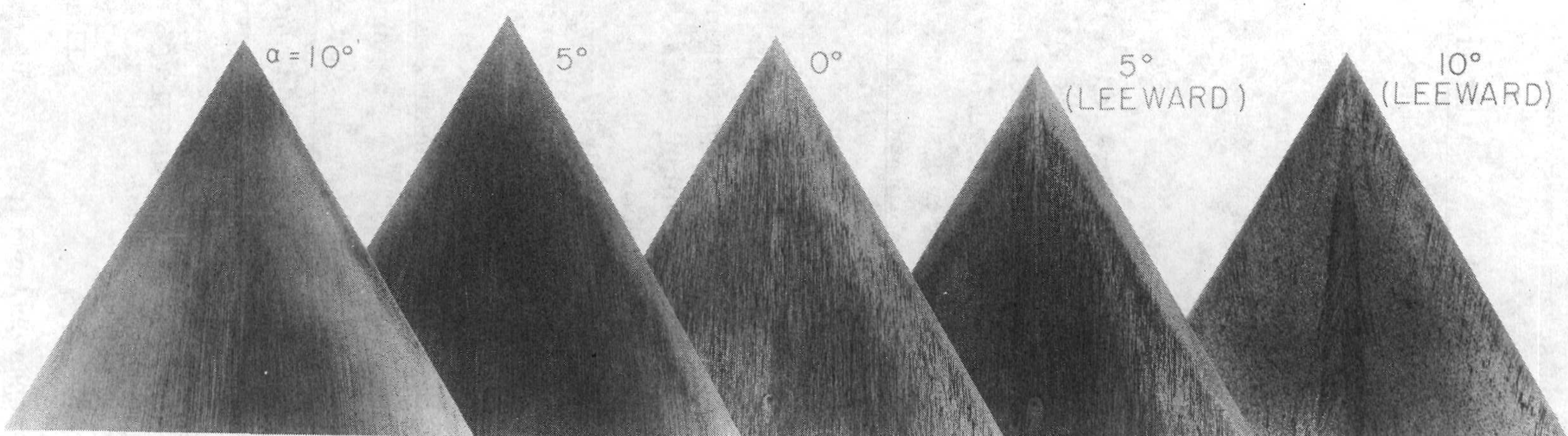
RESULTS AND DISCUSSION

Flow Field

Photographs obtained from the oil-flow studies conducted on the delta-wing models are presented in figure 2. For nominal angles of attack between 0° and 10°, the windward-



(a) Circular-arc delta wing.



(b) Rhombic delta wing.

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Figure 2.- Surface oil flow on 60° swept delta wings.

surface flow is nearly two dimensional with very little transverse outflow. A region of nonuniform surface shear is distinguishable on the delta wings at $\alpha = 0^{\circ}$ and 5° inboard of the leading-edge region. A comparison of the heat-transfer data (to be discussed later in this report) with the surface oil-flow data indicates that this region corresponds to the area where laminar flow ends and transitional flow begins.

On the leeward surfaces at nominal angles of attack of 5° and 10°, large regions of low surface shear prevail except for a high shear region at $\alpha = 10^{\circ}$ identified by a pronounced featherlike oil smear (indicative of vortex flows) down the leeward meridian. However, the leeward-surface flow is not believed to be separated since the oil-flow photographs do not show the abrupt change in surface shear characteristic of separated flows (ref. 16). Furthermore, vapor-screen photographs of the circular-arc delta wing show that the viscous boundaries lie close to the leeward surface. Note that the viscous layer on the leeward surface thickens on both sides of the meridian but thins in the high shear region of the meridian. This thinning is caused by vortices embedded within the viscous layer which transport mass away from the meridian (ref. 16).

Pressure Distributions

During the pressure tests the model-support system translated and deflected substantially more than anticipated as a result of large aerodynamic loads on the models. Therefore, both models with limited pressure instrumentation were rerun with improved angle-of-attack measurements to insure a high degree of accuracy for angle of attack (within $\pm 0.1^{\circ}$) and the results are presented in figures 3 and 4. Pressure distributions with a lesser degree of accuracy in angle of attack ($\pm 0.2^{\circ}$) are shown in figures 5 and 6. All of the windward-surface pressure data generated on the delta wings correlate for a given angle of attack with the conical-flow parameter ϕ.

A comparison between the present experimental data and predictions obtained by using the inviscid nonlinear method of lines (refs. 4 and 5) is shown in figures 3 and 4 for the windward surfaces of the delta wings. The method of lines predicts reasonably well both the magnitude and trend of the data shown with the better agreement occurring at the larger angles of attack where viscous effects are relatively small.

Heat Transfer

Heat-transfer distributions presented in terms of the free-stream Stanton number are shown in figures 7 and 8 for the circular-arc and the rhombic delta wings, respectively. The heat-transfer data on the wings correlate with the normalized distance from the model leading edge. Turbulent flow as indicated by the abrupt increase in Stanton number exists over most of the model surface. As expected, on the windward surface, transition is promoted with increasing angle of attack; on the leeward surface, transition

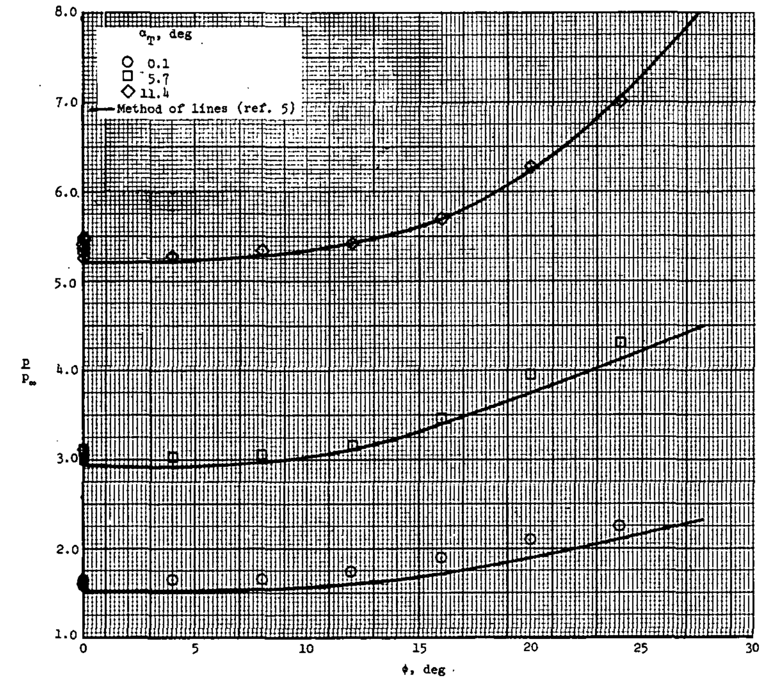


Figure 3.- Comparison of windward-surface pressures with method-of-lines predictions for circular-arc delta wing. $R_\infty = 2.33 \times 10^7$.

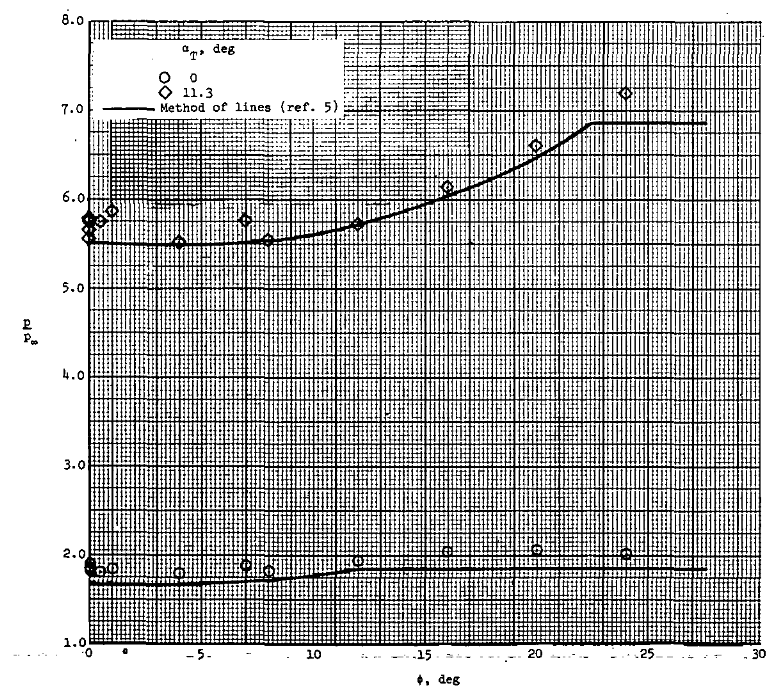


Figure 4.- Comparison of windward-surface pressures with method-of-lines predictions for rhombic delta wing. $R_\infty = 2.33 \times 10^7$.

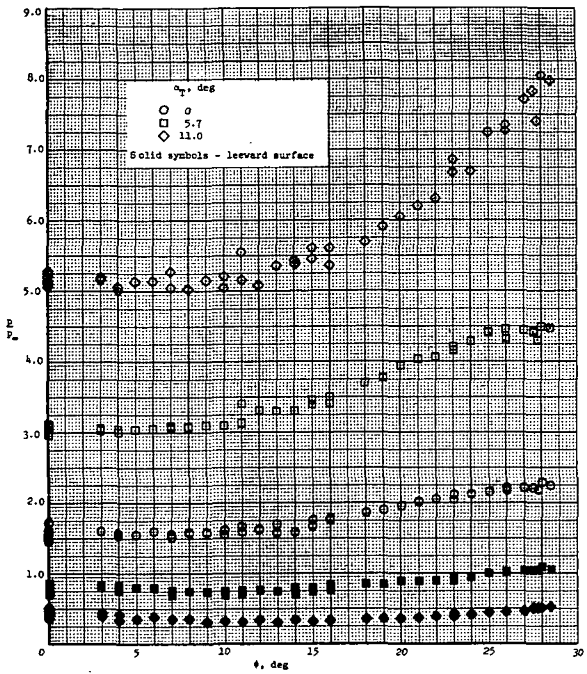


Figure 5.- Surface pressures on circular-arc delta wing. $R_\infty = 2.33 \times 10^7$.

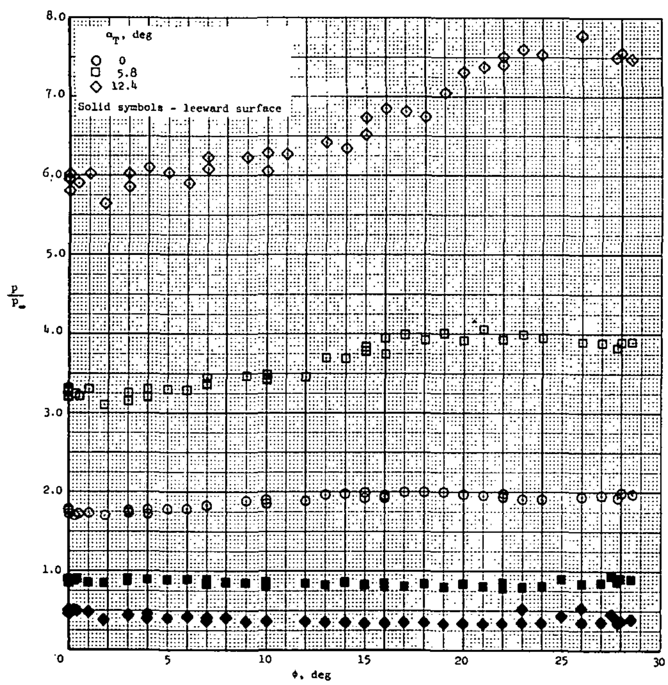
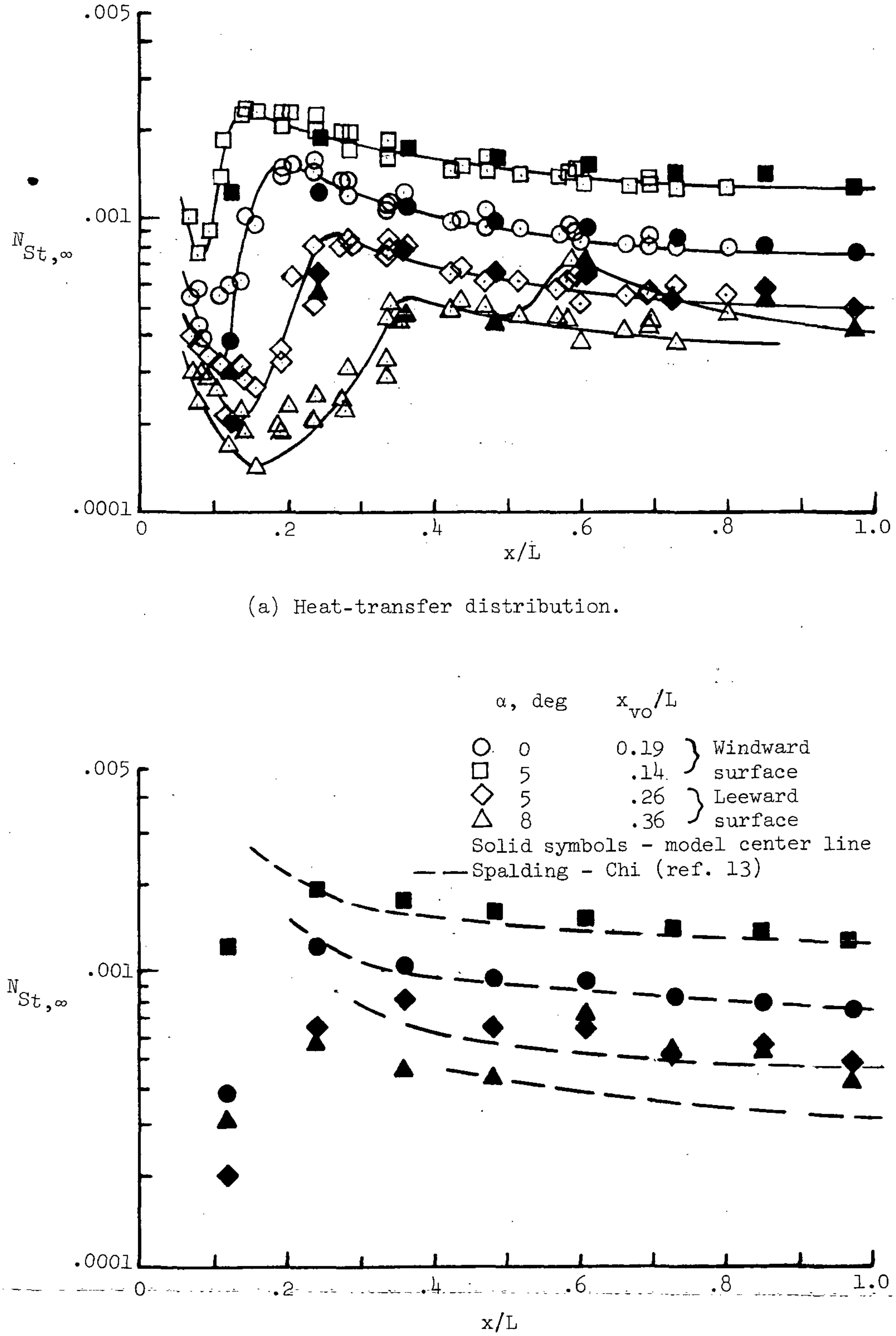


Figure 6.- Surface pressures on rhombic delta wing. $R_\infty = 2.33 \times 10^7$.



(a) Heat-transfer distribution.

(b) Center-line heating.

Figure 7.- Heat transfer on circular-arc delta wing. $R_\infty = 2.33 \times 10^7$.

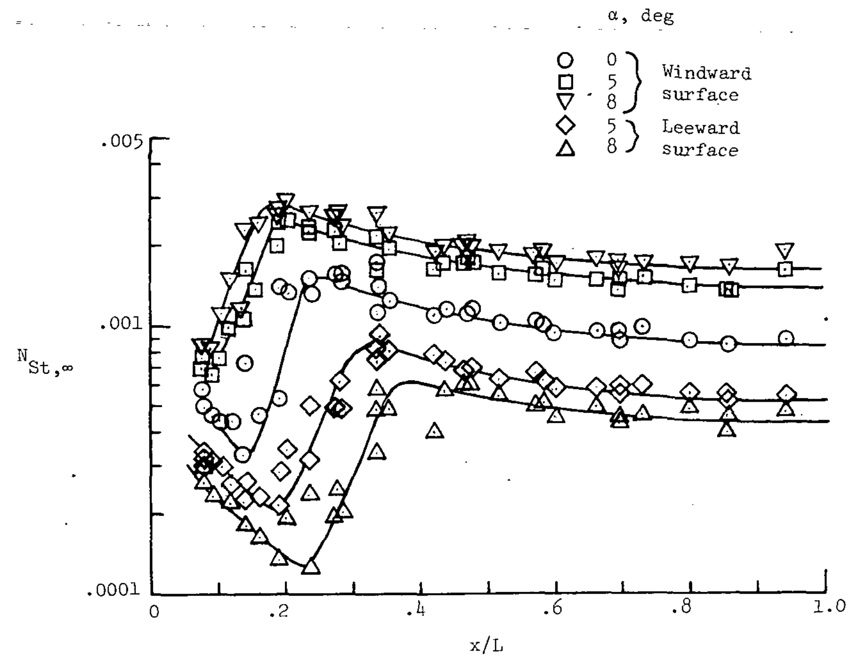


Figure 8.- Heat-transfer distribution on rhombic delta wing. $R_\infty = 2.33 \times 10^7$.

is delayed with increasing angle of attack. The method of Spalding and Chi (ref. 13) was applied to the data by assuming local conditions based on the nominal angle of attack and the virtual origin to be at the position of peak heating; this method predicted the turbulent levels of heating reasonably well. (See fig. 7(b).)

At $\alpha = 8^o$ the heating behavior down the leeward meridian of the circular-arc delta wing was characterized by two heat-transfer peaks. These localized heating peaks have been observed on other geometries and have been attributed to the interaction of the embedded vortices with the leeward-surface boundary layer and the inception of transitional boundary-layer flow. (See refs. 16 and 17.) In references 16 and 17, the first peak was indicated to result from the vortex thinning and the second peak from transition; the heating downstream of the first peak could be correlated by using the free-stream Reynolds number based on the distance from the first heating peak (vortex induced). However, none of the present data correlate with this Reynolds number parameter, which suggests that the first heating peak could be attributed to transition whereas the second peak is vortex induced. This hypothesis is in contrast to that discussed in the previously indicated references but the two hypotheses are not necessarily contradictory since vortices do not have to be confined to laminar flows. Further confirmation of the present hypothesis would be desirable.

CONCLUSIONS

An experimental investigation of the flow field over two 60° swept delta wings (circular-arc and rhombic cross-section delta wings) has been conducted at a Mach number of 6. The results of this study have indicated the following conclusions:

1. The windward-surface pressure distributions for both wings correlate with the conical flow angle measured from the model center line in the horizontal plane. The method of lines predicts both the magnitude and trend of the pressure data with the better agreement occurring at the higher angles of attack where the viscous effects are relatively small.

2. The heat-transfer distributions on the wings correlate with the normalized distance from the model leading edge. Turbulent flow exists over most of the model surface. The method of Spalding and Chi predicts the turbulent levels of heating reasonably well on the windward surfaces.

3. Oil-flow studies indicate that the windward-surface flow is nearly two dimensional for angles of attack between 0° and 10°. Vortices are present in the leeward-surface flow which distort the viscous regions and induce localized high heating in the region of the meridian.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., October 31, 1972.

REFERENCES

1. Jones D. J.: Numerical Solutions of the Flow Field for Conical Bodies in a Supersonic Stream. Aeronaut. Rep. LR-507 (NRC No. 10361), Nat. Res. Counc. Can. (Ottawa), July 1968.
2. Ndefo, D. Ejike: A Numerical Method for Calculating Steady Unsymmetrical Supersonic Flow Past Cones. Rep. No. AS-69-11 (AFOSR Grant 268-68), U.S. Air Force, May 1969. (Available from DDC as AD 691 270.)
3. South, Jerry C., Jr.; and Klunker, E. B.: Methods for Calculating Nonlinear Conical Flows. Analytic Methods in Aircraft Aerodynamics, NASA SP-228, 1970, pp. 131-155; Discussion, pp. 156-158.
4. Klunker, E. B.; South, Jerry C., Jr.; and Davis, Ruby M.: Calculation of Nonlinear Conical Flows by the Method of Lines. NASA TR R-374, 1971.
5. Klunker, E. B.; South, Jerry C., Jr.; and Davis, Ruby M.: Computer Program for Calculating Supersonic Flow on the Windward Side of Conical Delta Wings by the Method of Lines. NASA TM X-2438, 1972.
6. Bertram, Mitchel H.; and Everhart, Philip E.: An Experimental Study of the Pressure and Heat-Transfer Distribution on a 70° Sweep Slab Delta Wing in Hypersonic Flow. NASA TR R-158, 1963.
7. Stallings, Robert L., Jr.; Burbank, Paige B.; and Howell, Dorothy T.: Heat-Transfer and Pressure Measurements on Delta Wings at Mach Numbers of 3.51 and 4.65 and Angles of Attack From -45° to 45°. NASA TN D-2387, 1964.
8. Nagel, A. L.; Fitzsimmons, H. D.; and Doyle, L. B.: Analysis of Hypersonic Pressure and Heat Transfer Tests on Delta Wings With Laminar and Turbulent Boundary Layers. NASA CR-535, 1966.
9. Whitehead, Allen H., Jr.; and Dunavant, James C.: A Study of Pressure and Heat Transfer Over an 80° Sweep Slab Delta Wing in Hypersonic Flow. NASA TN D-2708, 1965.
10. Dunavant, James C.: Investigation of Heat Transfer and Pressures on Highly Swept Flat and Dihedraled Delta Wings at Mach Numbers of 6.8 and 9.6 and Angles of Attack to 90°. NASA TM X-688, 1962.
11. Paulsen, J. J.: Hypersonic Research Wing – A Study of the Pressure and Heat Transfer Distributions on Highly Swept Slab Delta Wings in Hypersonic Flow. ERR-AN-293, Gen. Dyn./Astronaut., Dec. 1963.

12. Mead, Harold R.; and Koch, Frank: Theoretical Prediction of Pressures in Hypersonic Flow With Special Reference to Configurations Having Attached Leading-Edge Shock – Pt. II. Experimental Pressure Measurements at Mach 5 and 8. ASD TR 61-60, Pt. II, U.S. Air Force, May 1962.

13. Neal, Luther, Jr.; and Bertram, Mitchel H.: Turbulent-Skin-Friction and Heat-Transfer Charts Adapted From the Spalding and Chi Method. NASA TN D-3969, 1967.

14. Goldberg, Theodore J.; and Hefner, Jerry N. (With appendix by James C. Emery): Starting Phenomena for Hypersonic Inlets With Thick Turbulent Boundary Layers at Mach 6. NASA TN D-6280, 1971.

15. Holloway, Paul F.; Sterrett, James R.; and Creekmore, Helen S.: An Investigation of Heat Transfer Within Regions of Separated Flow at a Mach Number of 6.0. NASA TN D-3074, 1965.

16. Whitehead, Allen H., Jr.; Hefner, Jerry N.; and Rao, D. M.: Lee-Surface Vortex Effects Over Configurations in Hypersonic Flow. AIAA Paper No. 72-77, Jan. 1972.

17. Rao, D. M.; and Whitehead, Allen H., Jr.: Lee-Side Vortices on Delta Wings at Hypersonic Speeds. AIAA J., vol. 10, no. 11, Nov. 1972, pp. 1458-1465.

TABLE I. - PRESSURE-ORIFICE LOCATIONS

(a) Circular-arc delta wing

Pressure orifice	x, cm	y, cm	R, cm	ϕ, deg	Pressure orifice	x, cm	y, cm	R, cm	ϕ, deg
1	2.86	0	2.86	0	27	38.15	6.73	38.74	10.0
2	4.45		4.45		28	32.41	6.30	33.02	11.0
3	7.30		7.30		29	43.48	9.24	44.45	12.0
4	10.16		10.16		30	26.60	6.14	27.31	13.0
5	13.02		13.02		31	9.86	2.46	10.16	14.0
6	15.88		15.88		32	37.58	9.37	38.74	14.0
7	18.73		18.73		33	15.33	4.11	15.88	15.0
8	21.59		21.59		34	42.73	12.25	44.45	16.0
9	4.45		24.45		35	36.84	11.97	38.74	18.0
10	30.16		30.16		36	31.22	10.75	33.02	19.0
11	33.02		33.02		37	41.77	15.20	44.45	20.0
12	35.88		35.88		38	25.49	9.78	27.31	21.0
13	38.74		38.74		39	35.91	14.51	38.74	22.0
14	41.59		41.59		40	14.61	6.20	15.88	23.0
15	44.45		44.45		41	30.24	12.90	33.02	23.0
16	38.71	1.35	38.74	2.0	42	40.61	18.08	44.45	24.0
17	32.97	1.73	33.02	3.0	43	24.75	11.54	27.31	25.0
18	21.54	1.51	21.59	4.0	44	19.41	9.46	21.59	26.0
19	44.34	3.10	44.45	4.0	45	34.82	16.98	38.74	26.0
20	27.20	2.38	27.31	5.0	46	29.42	14.99	33.02	27.0
21	38.52	4.05	38.74	6.0	47	39.34	20.70	44.45	27.8
22	15.76	1.94	15.88	7.0	48	34.20	18.18	38.74	28.0
23	32.77	4.02	33.02	7.0	49	39.06	21.21	44.45	28.5
24	44.02	6.18	44.45	8.0	50	31.89	8.55	33.02	15.0
25	26.97	4.27	27.31	9.0	51	20.75	5.95	21.59	16.0
26	21.26	3.75	21.59	10.0	52	24.22	12.61	27.31	27.5

TABLE I. - PRESSURE-ORIFICE LOCATIONS – Concluded

(b) Rhombic delta wing

Pressure orifice	x, cm	y, cm	R, cm	ϕ, deg	Pressure orifice	x, cm	y, cm	R, cm	ϕ, deg
1	2.87	0	2.87	0	25	32.77	4.02	33.02	7.0
2	4.44	.24	4.45	3.1	26	32.41	6.30		11.0
3	7.31	.24	7.32	1.9	27	31.89	8.55		15.0
4	13.03	.23	13.03	1.0	28	31.22	10.75		19.0
5	15.88	0	15.88	0	29	30.40	12.90		23.0
6	15.76	1.94		7.0	30	29.42	14.99		27.0
7	15.33	4.11		15.0	31	35.89	.24	35.89	.4
8	14.61	6.20		23.0	32	38.74	0	38.74	0
9	21.59	.24	21.59	.6	33	38.52	4.05		6.0
10	21.54	1.51		4.0	34	38.15	6.73		10.0
11	21.26	1.59		10.0	35	37.58	9.37		14.0
12	20.75	5.95		16.0	36	36.84	11.97		18.0
13	20.02	8.09		22.0	37	35.91	14.51		22.0
14	19.41	9.46		26.0	38	34.82	16.98		26.0
15	27.31	0	27.31	0	39	34.20	18.18		28.0
16	27.20	.24		5.0	40	44.45	.23	44.45	.3
17	26.97	4.27		8.0	41	44.34	3.10		4.0
18	26.60	6.14		13.0	42	44.02	6.18		8.0
19	26.11	7.98		17.0	43	43.48	9.24		12.0
20	25.49	9.78		21.0	44	42.73	12.25		16.0
21	24.75	11.54		25.0	45	41.77	15.20		20.0
22	24.22	12.62		27.5	46	40.61	18.08		24.0
23	30.18	.24	30.18	.5	47	39.34	20.70		27.8
24	32.97	1.73	33.02	3.0	48	39.06	21.21		28.5

TABLE II. - THERMOCOUPLE LOCATIONS

(a) Circular-arc delta wing

Thermocouple	x, cm	y, cm	R, cm	ϕ, deg	Thermocouple	x, cm	y, cm	R, cm	ϕ, deg
1	5.72	0	5.72	0	25	32.42	11.16	34.29	19
2	10.67	4.10	11.43	21	26	33.66	6.54		11
3	11.22	2.18		11	27	34.03	4.18		7
4	11.43	0		0	28	34.24	1.80		3
5	11.14	-2.57		-13	29	34.29	0		0
6	15.78	6.70	17.15	23	30	35.96	17.54	40.01	26
7	16.56	4.44		15	31	37.09	14.99		22
8	17.02	2.09		7	32	38.05	12.36		18
9	17.13	.60		2	33	38.82	9.68		14
10	17.15	0		0	34	39.40	6.95		10
11	16.56	-4.44		-15	35	39.79	4.18		6
12	21.20	8.56	22.86	22	36	39.98	1.40		2
13	21.97	6.30		16	37	40.01	0		0
14	22.51	3.97		10	38	39.79	-4.18		-6
15	22.80	1.60		4	39	40.37	21.46	45.72	28
16	22.86	0		0	40	41.77	18.60		24
17	22.51	-3.97		-10	41	42.96	15.64		20
18	25.90	12.08	28.58	25	42	43.95	12.60		16
19	27.33	8.35		17	43	44.72	9.51		12
20	28.22	4.47		9	44	45.28	6.36		8
21	28.47	2.49		5	45	45.61	3.19		4
22	28.57	.50		1	46	45.72	0		0
23	28.58	0		0	47	45.71	-.80		-1
24	30.55	15.57	34.29	27	48	45.61	-3.19		-4

TABLE II.- THERMOCOUPLE LOCATIONS – Concluded

(b) Rhombic delta wing

Thermocouple	x, cm	y, cm	R, cm	ϕ, deg	Thermocouple	x, cm	y, cm	R, cm	ϕ, deg
1	11.14	-2.57	11.43	-13	20	32.42	11.16	34.29	19
2	11.22	2.18		11	21	30.55	15.57	34.29	27
3	10.67	4.10		21	22	39.79	-4.18	40.01	-6
4	16.56	-4.44	17.15	-15	23	39.98	1.40		2
5	17.02	2.09		7	24	39.79	4.18		6
6	16.56	4.44		15	25	39.40	6.95		10
7	15.78	6.70		23	26	38.82	9.68		14
8	22.51	-3.97	22.86	-10	27	38.05	12.36		18
9	22.80	1.60		4	28	37.09	14.99		22
10	22.51	3.97		10	29	35.96	17.54		26
11	21.97	6.30		16	30	45.61	-3.19	45.72	-4
12	21.20	8.56		22	31	45.71	-.80		-1
13	28.47	2.49	28.58	5	32	45.61	3.19		4
14	28.22	4.47		9	33	45.28	6.36		8
15	27.33	8.35		17	34	44.72	9.50		12
16	25.90	12.80		25	35	43.95	12.60		16
17	34.24	1.80	34.29	3	36	42.96	15.64		20
18	34.03	4.18		7	37	41.77	18.60		24
19	33.66	6.52		11	38	40.37	21.46		28

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