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TECHNICAL NOTE
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VARIATION IN HEAT TRANSFER DURING TRANSIENT HEATING OF A HEMISPHERE AT A MACH NUMBER OF 2

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VARIATION IN HEAT TRANSFER DURING TRANSIENT HEATING OF A HEMISPHERE AT A MACH NUMBER OF 2

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SUMMARY

Convective heat-transfer tests were made on a 5-inch-diameter hemisphere to determine the variation of Stanton number with the ratio of wall temperature to total temperature. The tests were made at a nominal Mach number of 2 for stagnation temperatures of 760° R, 1,030° R, and 1,380° R. The model was constructed so that radiation effects and also streamwise conduction effects within the model skin were minimized. The results of the tests verified that these effects were small. Tests which were made with different masses of air inside the model to check for conduction effects to the internal air cavity showed these effects to be negligible. For laminar flow on the hemisphere, the Stanton number remained essentially constant as the ratio of wall temperature to total temperature increased. However, for fully established turbulent flow, the Stanton number at some stations decreased on the order of 50 percent as the ratio of wall temperature to total temperature increased. A theory which agreed fairly well with the trend of this decrease is shown for comparison.

INTRODUCTION

Some recent investigations concerning turbulent convective heat transfer carried out by the Langley Research Center have indicated an apparent decrease in the dimensionless heat-transfer coefficient (Stanton number) as the temperature ratio (wall to total temperature) increased. This decrease in Stanton number was much greater than that predicted by the turbulent theory of Van Driest. This decrease has for the most part been attributed to conduction to model supporting material or to radiation, or both. The present investigation has been made to determine whether this decrease of heat-transfer coefficient with temperature ratio would occur in the absence of conduction and radiation effects. In order to do this, an investigation has been made in which conduction to supporting material and radiation effects were made negligible. The investigation was made with a 5-inch-diameter thin-skin platinum-plated Inconel hemisphere in the preflight jet at

NASA Wallops Station at a nominal Mach number of 2.0 and at stagnation temperatures of approximately 760° R, 1,030° R, and 1,380° R.

SYMBOLS

c_p	specific heat of air at constant pressure, Btu/lb-°F
$c_{p,l}$	local value of specific heat of air at constant pressure, Btu/lb-°F
k	thermal conductivity of air, Btu-ft/ft^2-sec-°F
N_{Pr}	Prandtl number, $c_p\mu/k$
c_w	specific heat of Inconel wall, Btu/lb-°F
T_{aw}	adiabatic wall temperature, °R
p_i	internal pressure, lb/sq in. abs
T_l	local static temperature, °R
T_w	wall temperature, °R
T_t	stagnation temperature, °R
t	skin thickness, ft
η_r	temperature recovery factor
N_{St}	Stanton number
θ	angle between radial line on which a thermocouple is located and model axis, deg (see fig. 1)
ρ_w	weight density of Inconel wall, lb/cu ft
ρ_l	local value of air density, lb/cu ft
μ	viscosity of air, lb/ft-sec

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τ time, sec

V_l local value of air velocity, ft/sec

MODEL AND TESTS

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The model used in this investigation was a 5-inch-diameter hemisphere of 1/32-inch (nominal thickness) Inconel welded to a 1/4-inch-thick steel base. The model was plated internally and externally with platinum to a thickness of from 0.001 to 0.0015 inch. In order to regulate internal pressure during the tests, the model was hermetically sealed and equipped with a check valve located in the base. A row of chromel-alumel thermocouples was spotwelded to the internal surface of the skin along a great semicircle in a plane parallel to the direction of flow. Prior to the installation of the thermocouples, the skin thickness was measured at the thermocouple locations. Measured skin thicknesses are given in table I. Thermocouples were located along a radius of the base also to provide temperature measurements for use in estimating internal radiation. Thermocouple locations are shown on a sketch of the model in figure 1.

The tests were made in the preflight jet at NASA Wallops Station. Tests were made in the 12- by 12-inch jet at stagnation temperatures of approximately 760° R and 1,030° R, and in the ethylene-heated high-temperature jet at a stagnation temperature of approximately 1,380° R. Test Mach numbers were 2.01 and 2.03, respectively. Tests in the 12- by 12-inch jet were made at sea-level atmospheric pressure conditions at a free-stream Reynolds number based on a length of 1 foot of about 19×10^6 for a stagnation temperature of 760° R and about 13×10^6 for a stagnation temperature of 1,030° R. Tests in the high-temperature jet were at sea-level atmospheric pressure at a Reynolds number based on unit length of about 6×10^6. In both jets, the model was held out of the airstream until the jet free-stream flow had become steady. Then the model was injected by means of a hydraulically operated rotating stand to position it at the center of the jet. By this means, the model was subjected to transient flow conditions for only about 1 second. A description of the 12- by 12-inch jet is given in reference 1 and a description of the ethylene-heated, high-temperature jet, in reference 2. Figure 2 contains a photograph of the model in the test section of the high-temperature jet.

In order to obtain an estimate of the effects of conduction to the air in the interior of the model, tests were made at stagnation temperatures of approximately 1,030° R and 1,380° R with the model evacuated to an internal pressure of less than 1 inch of water, with model internal pressure of 1 atmosphere, and with an initial internal pressure of approximately 5 atmospheres. Table II lists the total temperatures and internal pressures for each test. During the tests with internal pressure of 1 atmosphere, the check valve was left open and the interior of the model was vented to the undisturbed atmosphere; thereby the internal pressure was kept constant during the tests. For the tests at initial internal pressure of 5 atmospheres, the check valve was necessarily closed and the internal pressure increased during the tests, because of the heating of the air, to about 8 to 9 atmospheres.

Skin temperature was measured during the tests by 30-gage chromel-alumel thermocouples and was recorded continuously on an oscillograph.

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DATA REDUCTION AND ACCURACY

When the terms for radiation, conduction along the model skin and into the thermocouple wires, and conduction to the air contained in the model are neglected, the expression for dimensionless heat-transfer coefficient (Stanton number) is

$$N_{St} = \frac{\rho_w c_w t \frac{dT_w}{d\tau}}{\left(T_{aw} - T_w\right) c_{p,l} \rho_l V_l}$$

The physical properties of Inconel used were 518 pounds per cubic foot for weight density and values of specific heat from reference 3. Plots of the skin temperature measured during the tests were graphically differentiated to obtain $dT_w/d\tau$. (A typical plot is shown in fig. 3.) The adiabatic wall temperature was calculated from

$$T_{aw} = \eta_r\left(T_t - T_l\right) + T_l$$

where $\eta_r = N_{Pr}^{1/2}$ for laminar flow and $\eta_r = N_{Pr}^{1/3}$ for turbulent flow. Values of T_l were calculated from measured stagnation temperature by using the Newtonian pressure distribution. Prandtl numbers were evaluated from the physical properties of air at skin temperature for the tests in the 12- by 12-inch jet and from the physical properties of the exhaust gas (ref. 2) at skin temperature for the tests in the high-temperature jet. For purposes of comparison, adiabatic wall temperatures were also obtained from the data by plotting the slope of the temperature-time curve $dT_w/d\tau$ against the temperature and considering the temperature at which the slope became zero to be the adiabatic wall temperature.

These adiabatic wall temperatures obtained from a fairing of the data and those obtained from theoretical calculations are compared in table III. In general, the agreement was good. The heat-transfer coefficients herein were calculated by using the theoretical adiabatic wall temperatures.

It will be noted that the foregoing analysis neglects the effects of radiation, conduction along the model skin, and conduction to the interior of the model. The effects of radiation and conduction along the model skin have been calculated and were found to be negligible. Figure 4 shows some typical streamwise temperature distributions along the model for various times. For the tests during which the model was evacuated, the interior of the model was for all practical purposes a vacuum and there was no interior conduction. It was impossible to estimate accurately the magnitude of interior conduction during the tests in which the model contained air. However, the analysis of the data showed no essential difference in measured Stanton number for the model when the interior was evacuated and when air was present. Consequently, effects of interior conduction were assumed to be small and were neglected in the data reduction.

The accuracy of the skin-temperature measurements resulting from limitations of instrumentation and record reading is within 6° R, 12° R, and 18° R (± 2 percent of full-scale range) for the tests at total temperatures of 760° R, $1,030^{\circ}$ R, and $1,380^{\circ}$ R.

RESULTS AND DISCUSSION

Effects of Varying Temperature Ratio

The effect of variation in temperature ratio on the Stanton number is shown in figure 5 for laminar flow, in figure 6 for transitional flow, and in figure 7 for turbulent flow, wherein the data from the tests during which the model was evacuated are presented. The variety in the types of flow encountered during these tests was not the result of any attempt to influence the nature of the boundary layer. It is apparent from the magnitude of the measured Stanton numbers that laminar flow existed over the entire model at the beginning of the first test. As a result of roughening of the model surface by scale present in the gas stream, transition started to take place over the downstream portion of the model during the first test and the transition point moved progressively forward during subsequent tests.

Inspection of figure 5 shows no appreciable variation of Stanton number over a range of temperature ratios from 0.55 to 0.90 for laminar

flow. The gradual increase in heat transfer for a stagnation temperature of 760° R at $\theta = 7.5^{\circ}$ and $\theta = 15^{\circ}$ and for a stagnation temperature of 1,030° R at $\theta = 45^{\circ}$ is due to a tendency toward transitional flow rather than to an effect of varying wall temperature. Any possible effect on Stanton number of varying temperature ratio in transitional flow (fig. 6) is hidden by the wide variation of Stanton numbers associated with this type of boundary layer. At some locations, a marked decrease in heat transfer is found for turbulent flow (fig. 7) as the temperature ratio increases from about 0.52 to 0.92. The magnitude of the decrease is from about 20 percent to about 50 percent and a comparison of figure 7 ($\theta = 15^{\circ}$ and 22.5°) with figure 5 ($\theta = 15^{\circ}$ and 22.5°) L
shows that at some locations the turbulent Stanton number near a tempera- 4
ture ratio of 0.92 is of the same order of magnitude as the laminar 6
Stanton number. 3

No account has been taken previously of the possible effects of axial skin temperature and pressure gradients which existed on the model during the tests. Reference 4, which is a theoretical investigation of these effects, has indicated that they would not be large for the conditions of the tests herein. That is, the temperature and pressure gradients did not cause the large decrease in Stanton number with temperature ratio as noted for some stations in figure 7.

The effect on Stanton number distribution of varying temperature ratio is shown in figure 8. The slight variation in Stanton number distribution for laminar flow is the result of experimental accuracy rather than an effect of varying temperature ratio. However, there was a wide variation in Stanton number distribution for turbulent flow; this variation was much greater than that expected to be caused by the movement of transition. As shown on figure 8, the decrease in Stanton number with increasing skin temperature was larger between $\theta = 15^{\circ}$ and $\theta = 45^{\circ}$ than on the upstream and downstream parts of the model.

Effects of Conduction to the Air in the Interior of the Model

Some effects on Stanton number of conduction to the interior of the model are shown in figure 9 for laminar flow and in figure 10 for turbulent flow where data obtained from the evacuated model are compared with those obtained from the model with an internal pressure of 1 atmosphere and an internal pressure varying from about 5 to about 8 atmospheres during the test. No appreciable effect of conduction to the interior of the model is shown. It is obvious from the increase in pressure that the air in the model was being heated during the tests starting with an internal pressure of 5 atmospheres. Since skin temperature varied from thermocouple to thermocouple, it was impossible to estimate accurately the amount of heat being lost to the interior at any one point. However, the heat capacity of the internal air in the model compared with the heat capacity of the model was small as shown by the following calculation:

$$\frac{\text{Heat capacity of air}}{\text{Heat capacity of model}} = \frac{c_v W_a}{c_w W_i} = \frac{(0.171)(0.007235)}{(0.11)(0.368)} = 0.0305$$

where

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c_v specific heat of air at constant volume, Btu/lb-$^{\circ}$F

c_w specific heat of Inconel, Btu/lb-$^{\circ}$F

W_a weight of air in model, lb

W_i weight of Inconel in model, lb

It is apparent from this low value of 0.0305 and from figures 9 and 10 that the heat losses to the internal air were negligible.

Also, heating rates of the air in the model have been calculated by using the measured variation in model internal pressure and were found to be less than 3.5 percent of the lowest heating rates on the model skin at corresponding times.

Comparison With Theory

The data obtained in these tests are compared with theoretical calculations in figure 11. Since no appreciable effects of conduction to the interior of the model were found, all the experimental data were included in the figure. The method of reference 5 was used for laminar flow and the method of reference 6 for turbulent flow. Local flow conditions for use in the theories were calculated by assuming a Newtonian pressure distribution. No effects of skin-temperature gradient were considered in the calculations. The laminar theory of reference 5 generally underestimated the experimental data. The turbulent theory of reference 6 underestimates experimental data on the upstream portion of the model, gives fair agreement between $\theta = 15^{\circ}$ and $\theta = 60^{\circ}$, and overestimates experiment near the downstream end of the model. The proper trend of the variation of Stanton number with temperature ratio is indicated by both laminar (ref. 5) and turbulent (ref. 6) theory, that is, no variation for laminar flow and a decrease in Stanton number with increasing temperature ratio for turbulent flow. It is noteworthy that the theory of reference 5 uses values of the gas properties evaluated at local static temperature whereas reference 6 uses gas properties evaluated at wall temperature. A laminar theory presented in reference 6 in which gas properties are evaluated at wall temperature indicates a variation of about 7 percent in laminar Stanton numbers over the range of temperatures investigated in these tests.

CONCLUDING REMARKS

Heat-transfer tests were made with a 5-inch-diameter hemisphere at L
a nominal Mach number of 2 for stagnation temperatures of about 760° R, 4
$1{,}030^{\circ}$ R, and $1{,}380^{\circ}$ R. The model was constructed so that radiation 6
effects and also streamwise conduction effects within the model skin 3
were minimized. The temperature distributions obtained verified that these effects were small. Tests which were made with different masses of air inside the model to check for conduction effects showed these effects to be negligible. For laminar flow on the hemisphere, the Stanton number remained essentially constant, but, for fully established turbulent flow on the hemisphere, the Stanton number at some stations decreased from 20 to 50 percent with increasing ratio of wall temperature to total temperature. Calculations by an existing theory which agreed fairly well with this decrease are shown for comparison.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., March 22, 1960.

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TABLE I.- MEASURED SKIN THICKNESS

Thermocouple	Thickness, in.
1	0.031
2	.031
3	.031
4	.031
5	.031
6	.031
7	.031
8	.031
9	.030
10	.030
11	.030

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TABLE II.- TOTAL TEMPERATURES AND INTERNAL

PRESSURES FOR THE TESTS

Test	T_t, °R	Internal pressure
1	1,030	Evacuated
2	1,030	Evacuated
3	1,030	1 atmosphere
4	1,030	5 atmospheres
5	760	Evacuated
6	1,380	Evacuated
7	1,380	1 atmosphere
8	1,380	5 atmospheres

TABLE III.- ADIABATIC WALL TEMPERATURES FOR BOTH DATA AND THEORY

Thermocouple		Adiabatic wall temperatures, °R, for -															
		Test 1		Test 2		Test 3		Test 4		Test 5		Test 6		Test 7		Test 8	
Number	Location,* deg	Data	Theory	Data	Theory	Data	Theory	Data	Theory	Data	Theory	Data	Theory	Data	Theory	Data	Theory
1	45	1,035	999	1,016	1,022	1,028	1,015	1,027	1,026	758	746	-----	1,370	-----	1,340	-----	1,350
2	37.5	1,006	1,006	1,015	1,029	1,027	1,022	1,042	1,033	760	751	1,368	1,380	1,350	1,350	1,366	1,359
3	30	1,017	1,012	1,018	1,035	1,020	1,028	1,035	1,039	765	750	-----	1,388	-----	1,357	-----	1,367
4	22.5	1,020	1,017	1,036	1,040	1,033	1,033	1,045	1,044	770	759	1,380	1,395	1,355	1,364	1,360	1,374
5	7.5	997	1,022	1,027	1,045	1,022	1,038	1,038	1,049	767	764	1,385	1,402	1,330	1,371	1,350	1,381
6	15	1,026	1,020	1,027	1,043	1,025	1,036	1,038	1,047	767	762	1,370	1,399	1,330	1,368	1,350	1,378
7	22.5	1,018	1,017	1,016	1,040	1,020	1,033	1,036	1,044	760	759	1,350	1,395	1,328	1,364	1,340	1,374
8	30	1,011	1,012	1,030	1,035	-----	1,028	-----	1,039	---	756	-----	1,388	-----	1,357	-----	1,367
9	60	1,000	983	995	1,004	1,015	998	1,020	1,008	755	734	1,340	1,347	1,330	1,318	1,340	1,327
10	67.5	1,010	973	1,007	995	-----	989	-----	999	---	727	-----	1,335	-----	1,306	-----	1,315
11	82.5	1,010	959	980	981	997	974	994	984	735	716	1,310	1,315	1,280	1,286	-----	1,296

*Angle between free-stream direction and a perpendicular to the surface at a thermocouple.

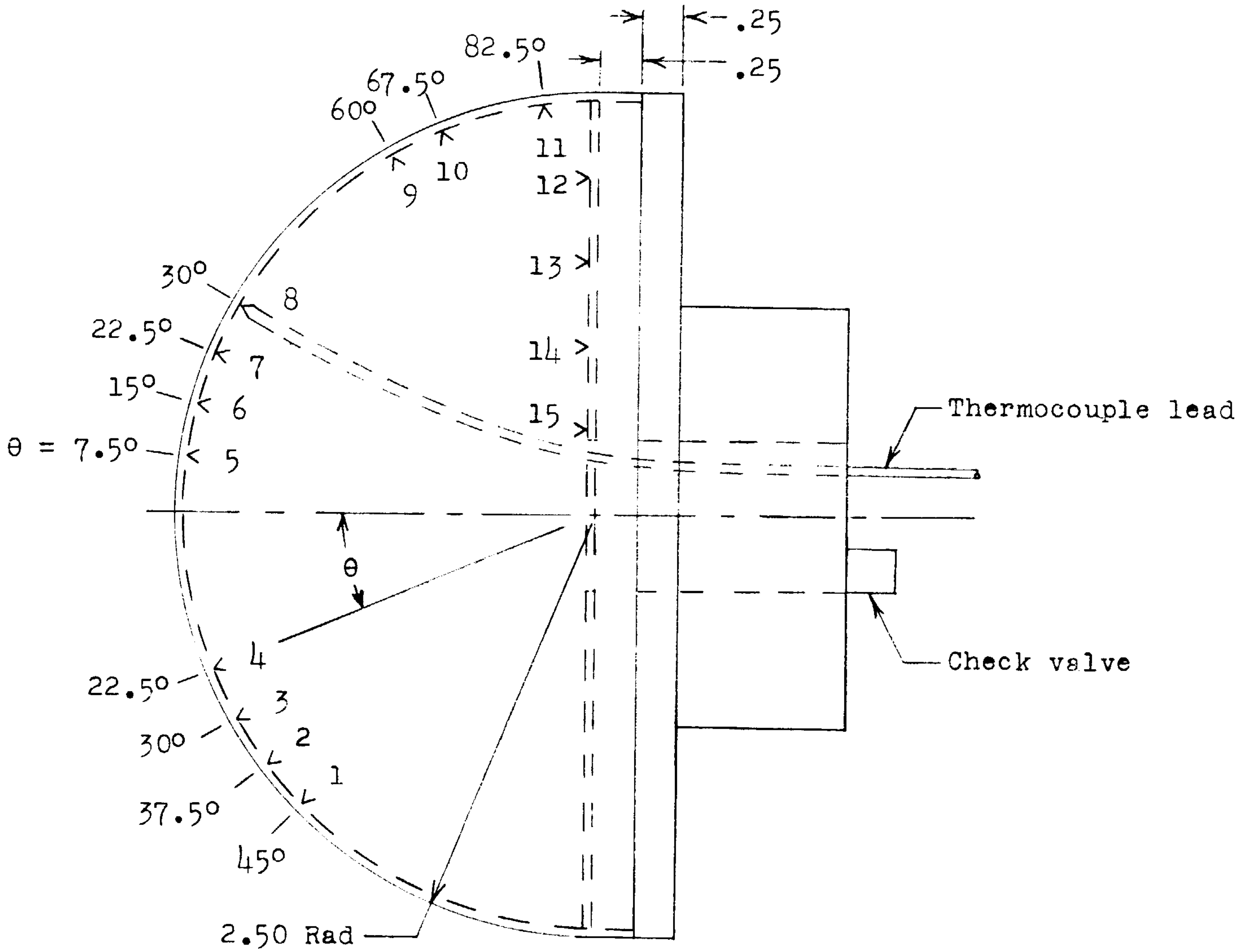

Figure 1.- Sketch of model showing thermocouple locations. All dimensions are in inches.

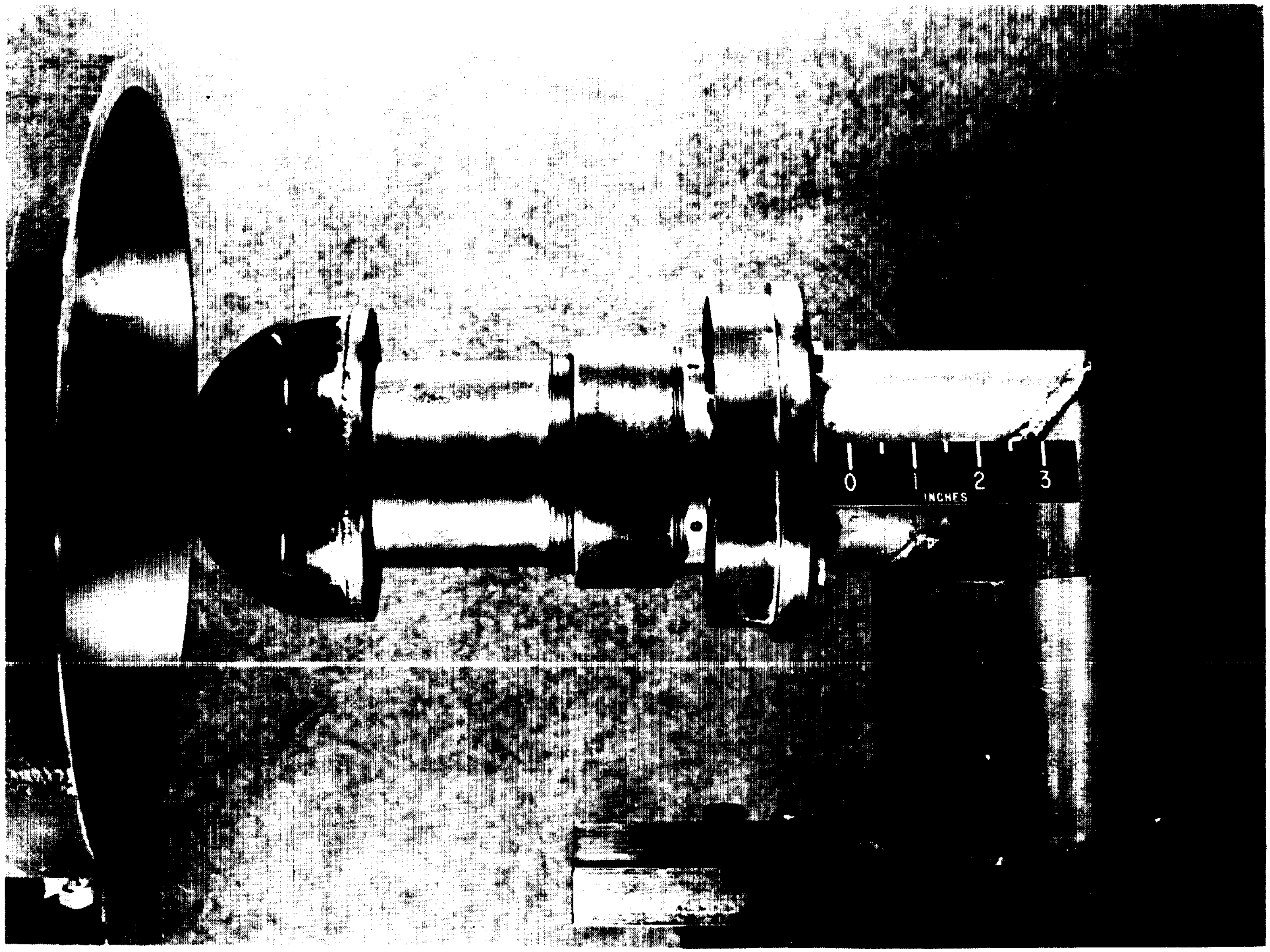

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Figure 2.- Hemisphere model in the test section of the ethylene-heated blowdown jet.

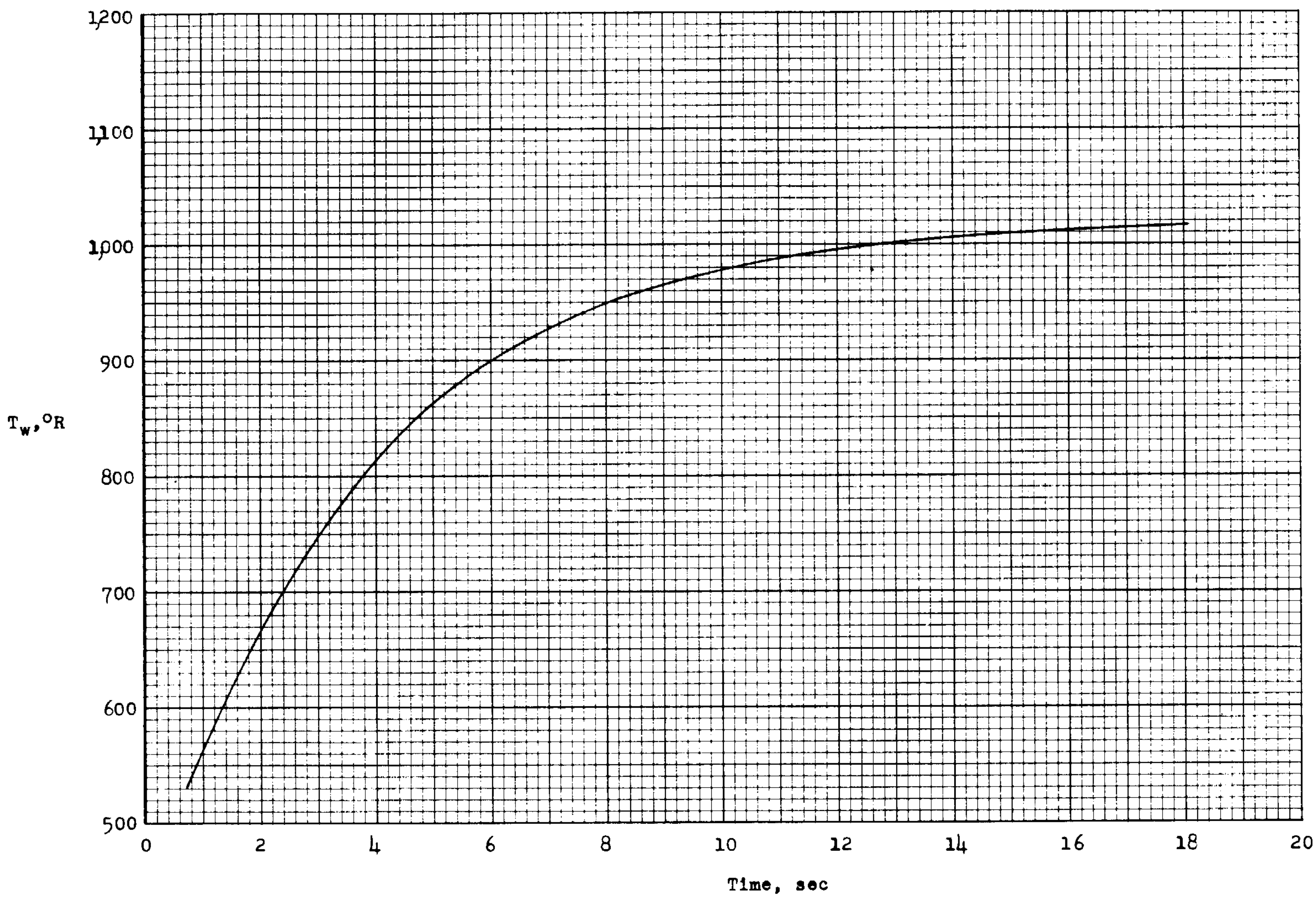

Figure 3.- Typical wall temperature time history. $\theta = 7.5^{\circ}$; $T_t = 1{,}030^{\circ}$ R; $p_i = 0$ pound per square inch absolute.

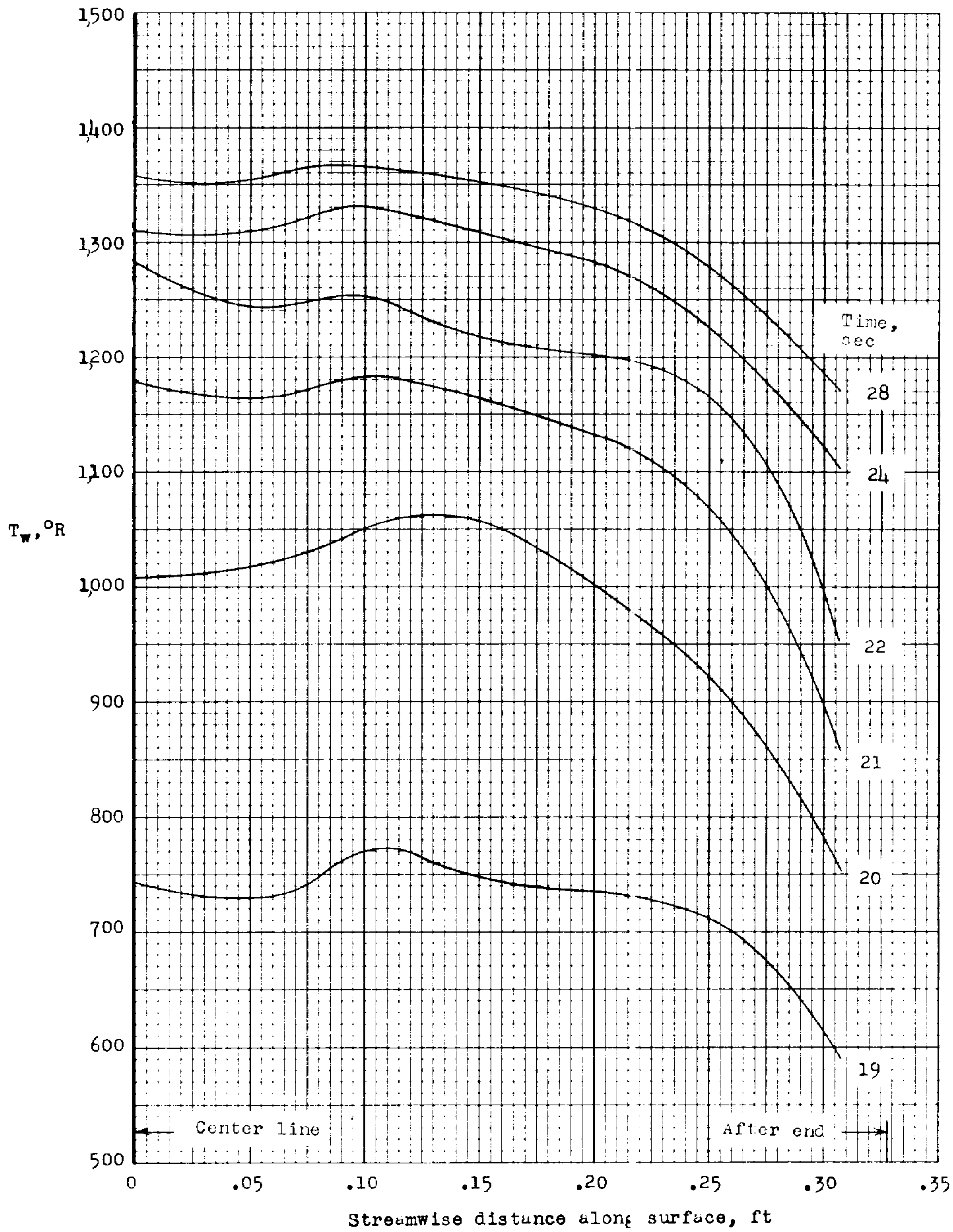

Figure 4.- Typical streamwise wall temperature distributions along the model for various times. Test 6; T_t = 1,380° R; p_i = 0 pound per square inch absolute.

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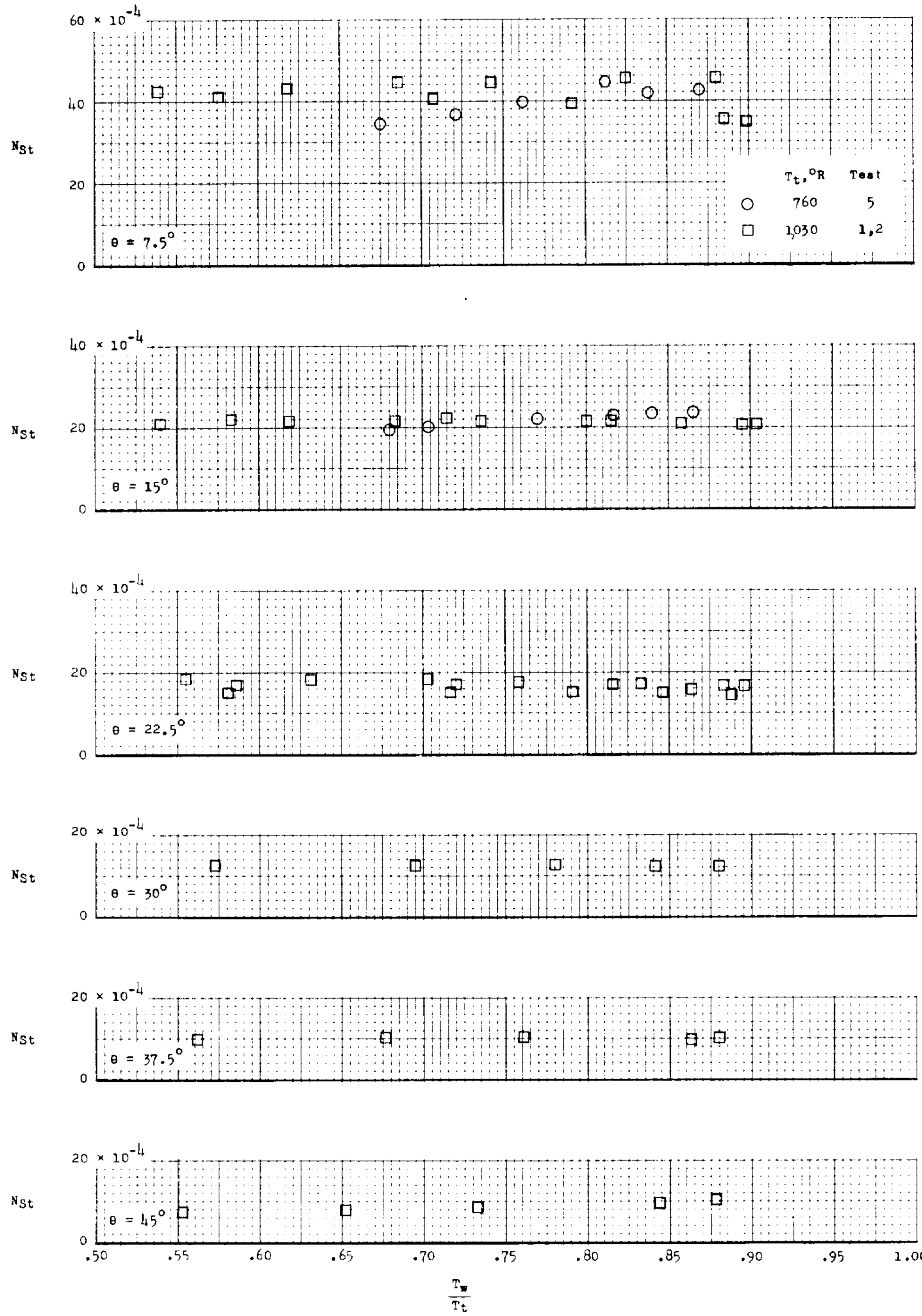

Figure 5.- Variation of Stanton number with temperature ratio for laminar flow.

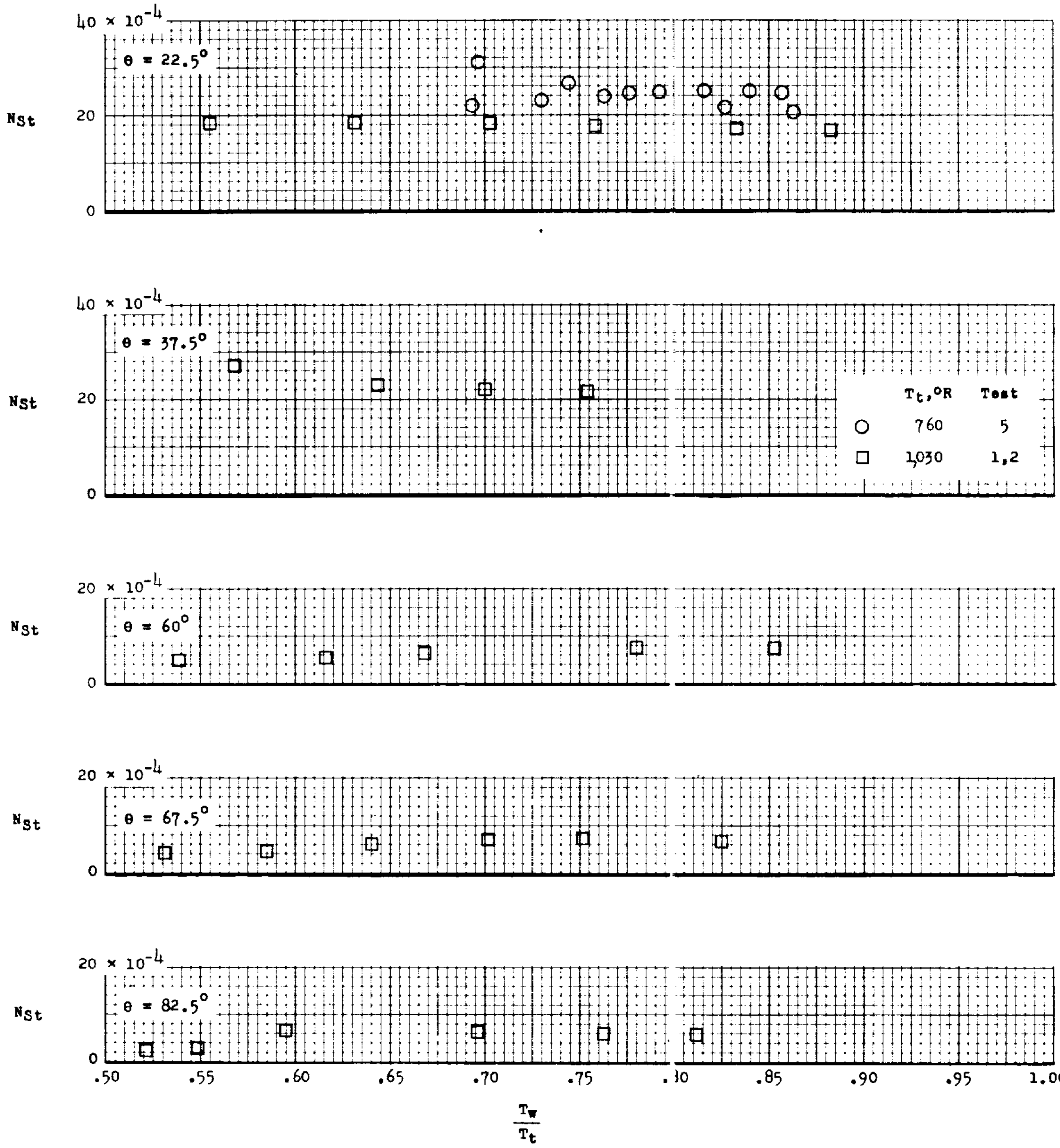

Figure 6.- Variation of Stanton number with temperature ratio for transitional flow.

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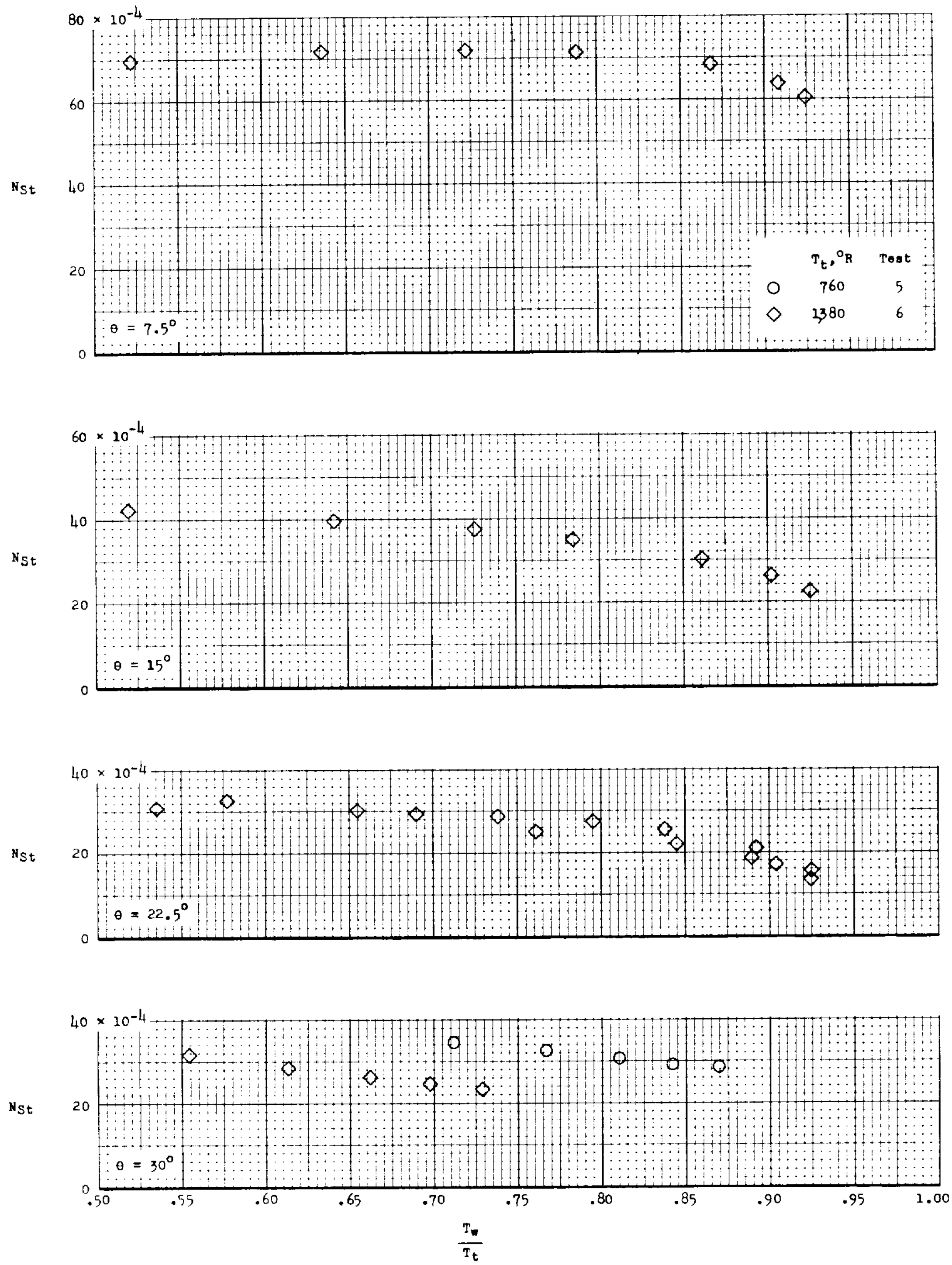

Figure 7.- Variation of Stanton number with temperature ratio for turbulent flow.

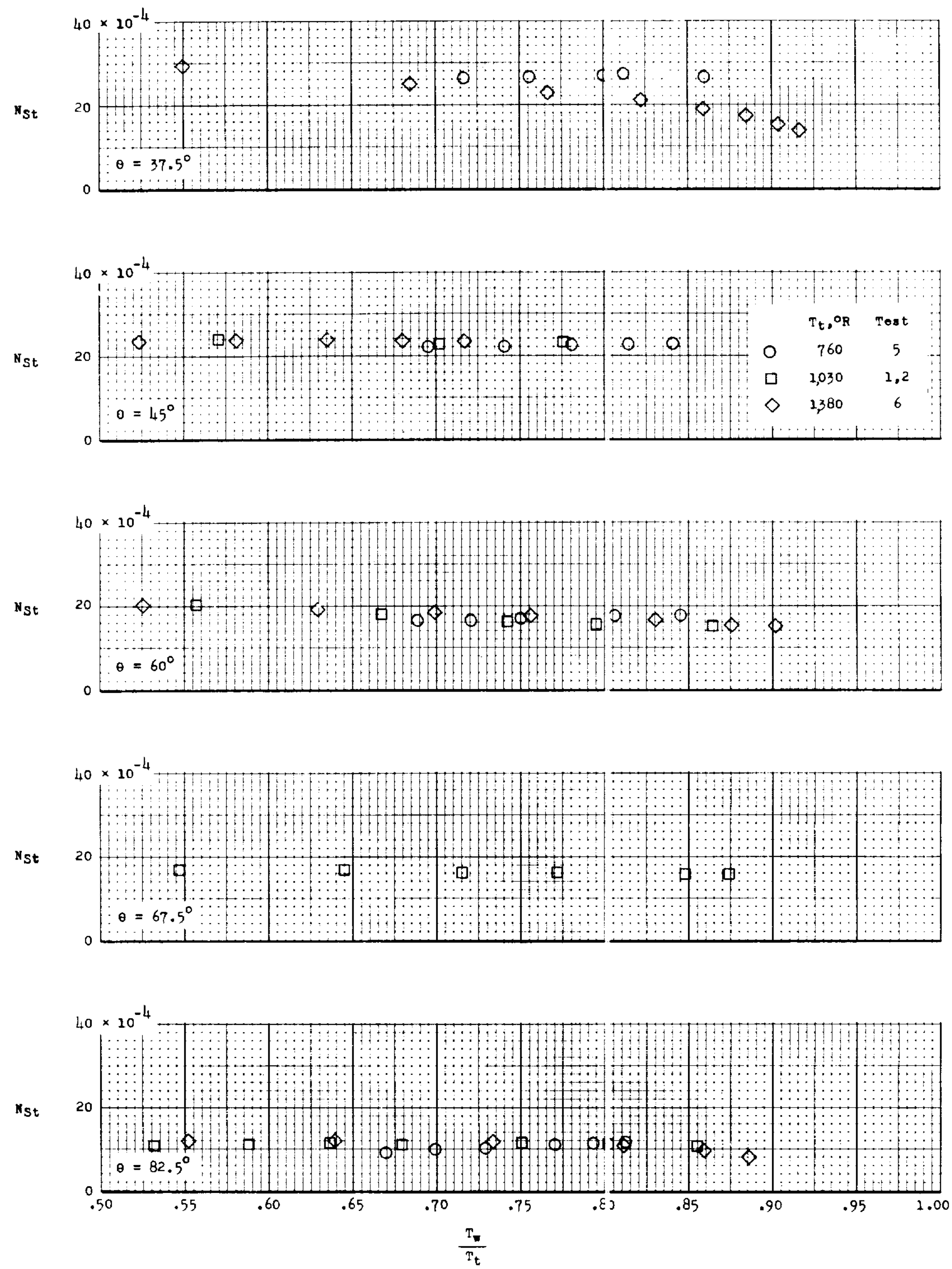

Figure 7.- Concluded.

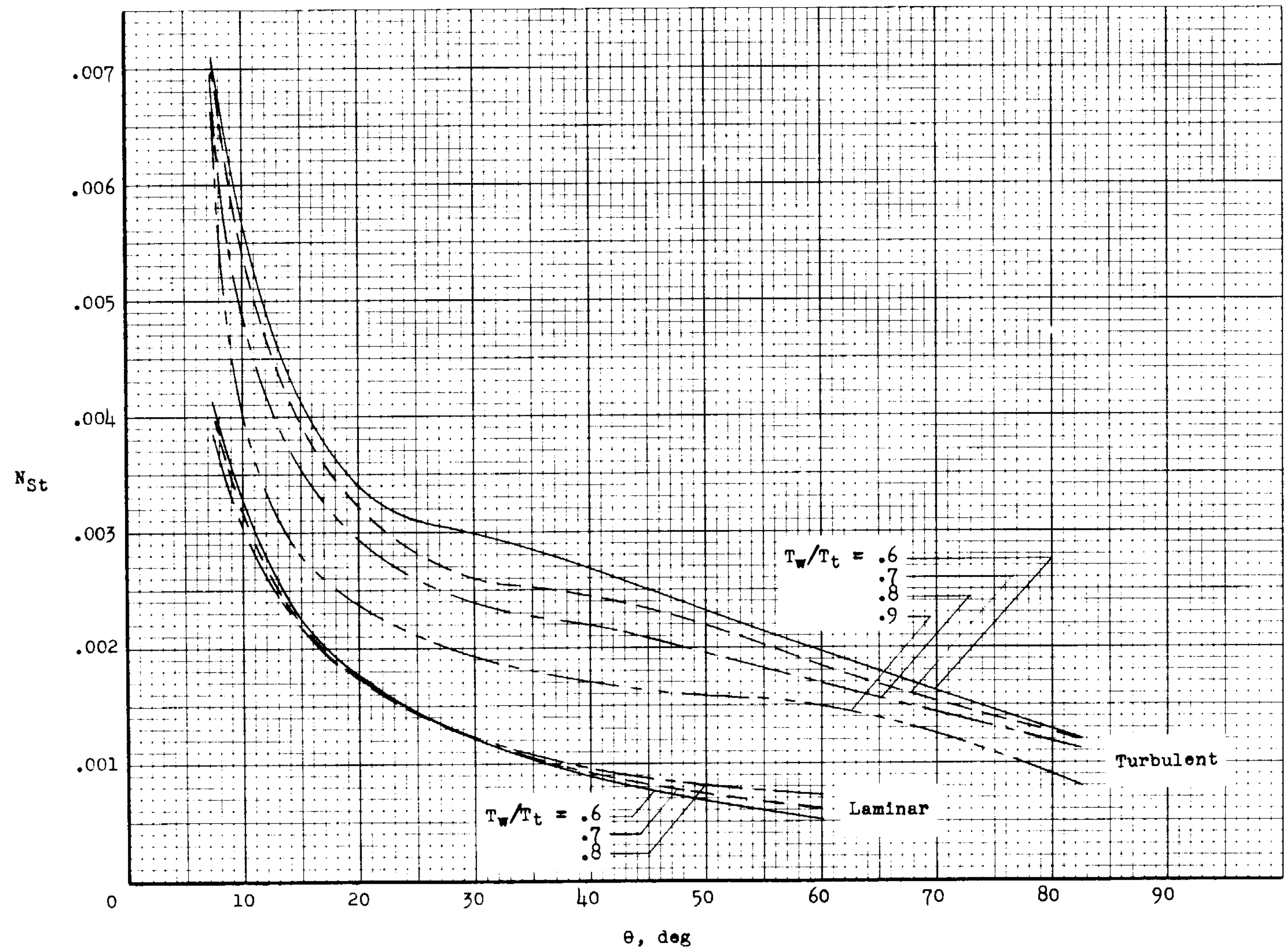

Figure 8.- Effect of the ratio of wall temperature to total temperature on the Stanton number distribution.

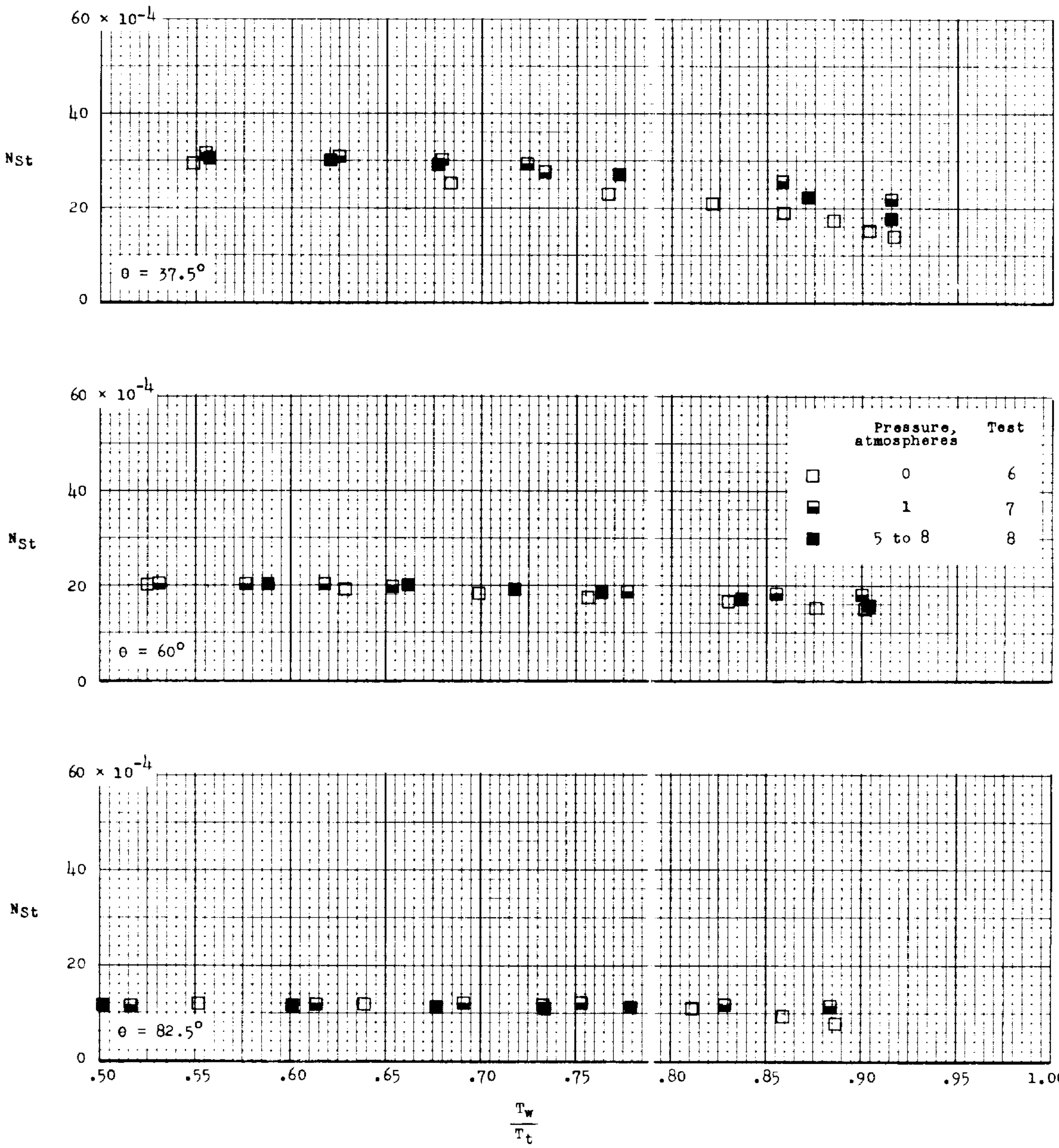

Figure 10.- Concluded.

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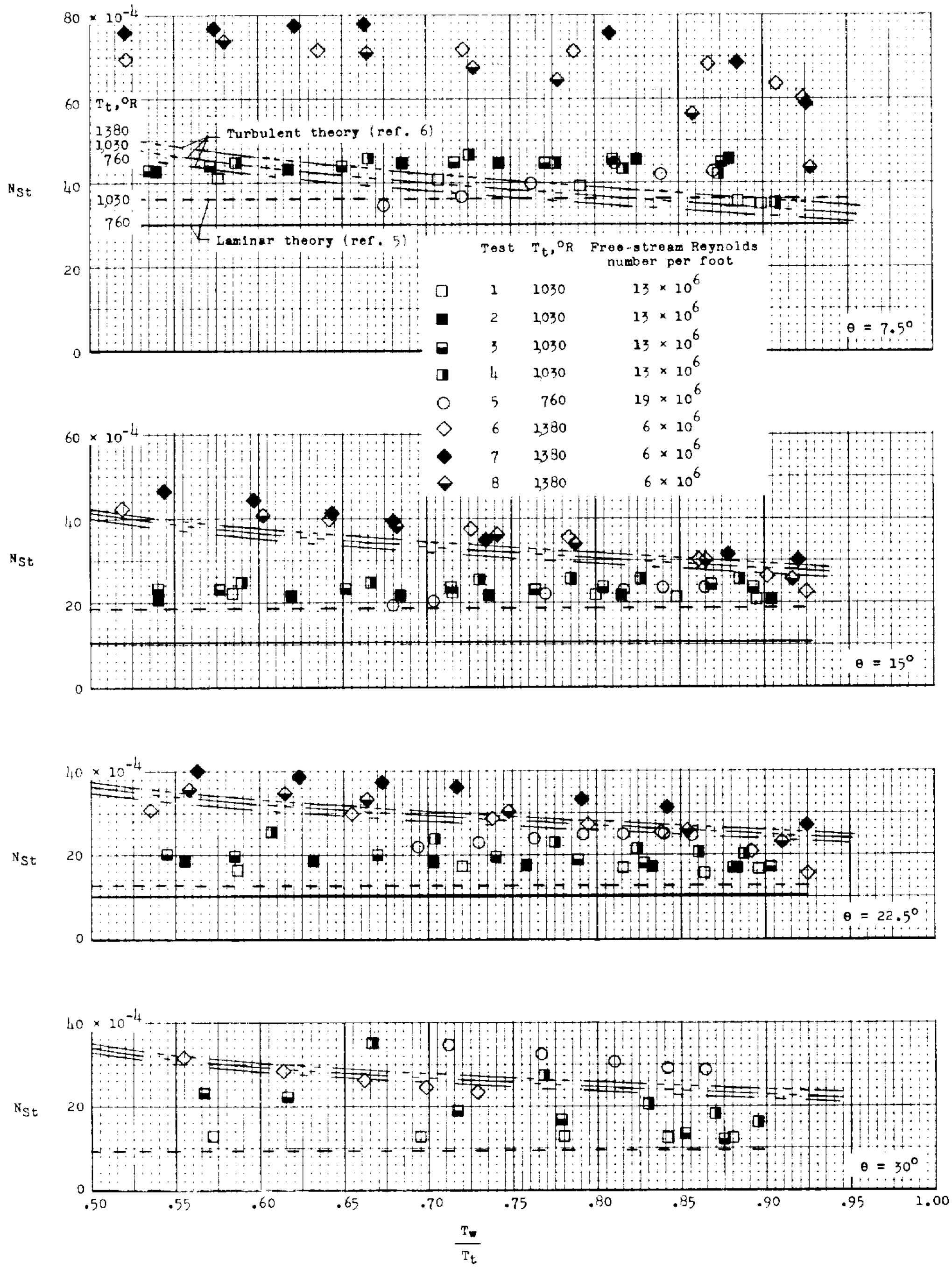

Figure 11.- Comparison of experimental and theoretical Stanton numbers.

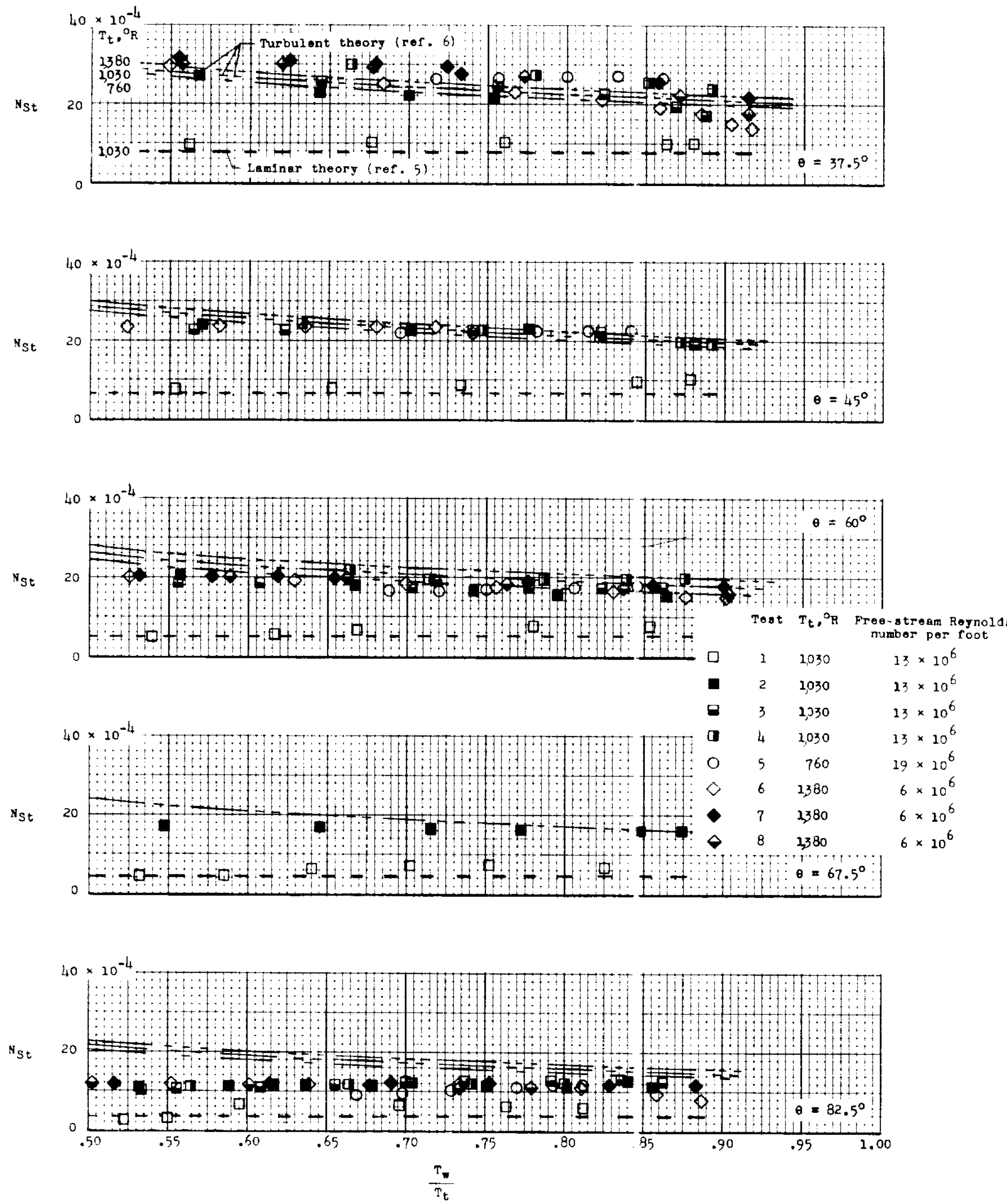

Figure 11.- Concluded.