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RESEARCH MEMORANDUM

APPARATUS FOR OBTAINING A SUPERSONIC FLOW OF VERY SHORT DURATION AND SOME DRAG MEASUREMENTS OBTAINED WITH ITS USE

By John E. Yeates, Jr., F. J. Bailey, Jr., and T. J. Voglewede

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SUMMARY

Apparatus consisting of a supersonic nozzle, vacuum tank, and sting dynamometer has been developed to provide supersonic flows of very short duration for comparison of the drag of aerodynamic shapes.

Measurements of the drag of bodies of revolution, with fineness ratios 6 and 12, four triangular-wing models, and an RM-10 body with stabilizing fins have been made in the Mach number range from 1.4 to 1.5 with this apparatus. To get a preliminary check on the accuracy of drag data obtained in this way, the measurements have been compared with drag data obtained from free-fall, free-flight, and wind-tunnel tests of similar bodies.

The comparison indicates that the apparatus gives drag values in good agreement with free-fall, free-flight, and wind-tunnel tests. The apparatus is, therefore, well-suited for use in making comparative drag studies. The agreement with other test methods also indicates that, with some further development, the apparatus may be useful for direct quantitative drag determinations.

INTRODUCTION

During the course of experiments being carried out by means of the freely-falling-body technique to assist in the development of bodies and wing-body combinations having low drag in the transonic and supersonic speed range, a need became evident for an inexpensive and rapid method of providing preliminary experimental indications of favorable design trends. With such a method available, the effect of design parameters could be explored quickly and cheaply to guide the selection of configurations to be subjected to confirmatory tests by free-fall and other methods.

The auxiliary apparatus described in this paper was constructed to meet this need and has been developed to a point where satisfactory measurements of relative drag can be obtained. In comparing the results obtained with this apparatus with those of more exact methods, it was found that the apparatus seemed to give more accurate results than was originally thought necessary for its intended use. In view of this situation, the basic idea of the apparatus might be developed for specific problems and uses other than that considered here. In this paper, the apparatus is described and some comparisons are made between the results obtained with this apparatus and the measurements of free-fall, rocket, and supersonic-tunnel techniques.

APPARATUS AND METHOD OF TEST

A two-dimensional nozzle with a 1-foot-square test section is vertically attached to a tank (figs. 1(a) and 1(b)). Because of the relatively small size of the tank, a quick acting valve must be employed to obtain supersonic velocity in the test section. This valve is in the form of a piece of acetate sheeting 0.015 inch thick and 18 inches in diameter, which is placed between the exit of the nozzle and the mouth of the tank. The tank having a volume of 80 cubic feet is pumped down to 25-inches-mercury vacuum by a 5-horsepower pump. The process takes about 2 minutes. The acetate diaphragm is then punctured by a solenoid-operated punch from inside the tank. The sudden rush of air into the partial vacuum sets up supersonic flow in the nozzle.

In figure 2 is shown a record of static pressure against time measured in the test section during a run. The orifice for this pressure measurement is located on a flat-wall side at the center line of the test section. The test-section length was designed to be 14 inches. The nominal test-section Mach number is calculated from the inlet total pressure (atmospheric pressure) and the measured static pressure, neglecting condensation. For the triangular-wing models and RM-10 body, the Mach number in the test section is calculated from the measured static and test-section total pressures. Actual pressure measurements have been taken across the width and along the length of the test section and show the Mach number to be constant within $\pm$ 0.01. The pressure measurements also show that there is a loss in total pressure which is presumed to be caused by a condensation shock near the throat of the nozzle. The instrument used to measure the static pressure was an NACA pressure recorder using a diaphragm-type pressure cell. The natural frequency of this cell is about 300 cycles per second. For the record of figure 2, 1 inch of $\frac{1}{4}$-inch inside-diameter tubing was used to connect the pressure cell to the measuring orifice. In figure 2 the first part (1) is the starting procedure. The unsteady-flow process that brings about this trace of the static pressure is similar to that studied theoretically in reference 1. In the second part (2), the air reaches a steady static pressure, corresponding to a Mach number of 1.5, and remains at this pressure for about 0.03 second, the time depending upon the initial pressure in the tank. The time duration of steady supersonic flow has been substantiated by measuring the test-section static pressure using a Statham pressure pickup, range ± 10 pounds per square inch. This instrument has a natural frequency of about 1000 cycles per second and the lag of the pressure pickup and recording galvanometer is believed to be about 0.003 second. This Statham gage shows that steady supersonic flow is obtained for about 0.03 second (counting pickup and recording-system lag). The last part (3) is the blow-down and is that part of the run in which the velocity decreases from Mach number 1.5 to zero. The total duration of the pressure-time curve is about 0.10 second.

With this particular arrangement of the nozzle and tank a considerable overshoot is found in the return to atmospheric pressure. Since the measurement of drag during the supersonic portion is all that is of interest, this resonance phenomenon is of no particular concern. During the early development stages a pressure tank instead of vacuum was tried with the nozzle. Although this arrangement showed no overshoot in the return to atmospheric pressure, it was shelved due to model-mounting difficulties.

Drag measurements are obtained using a Statham strain-gage dynamometer, range ± 32 ounces, type G-1. This dynamometer is basically a resistance strain bridge network with an external oil dash pot used to dampen the response of the dynamometer element. It does not cause any appreciable lag in the recording system. The bridge unbalance due to drag forces applied to the dynamometer through the model support sting is recorded by a galvanometer. A static calibration, which requires about 10 minutes, is made before each series of runs. This calibration consists simply of hanging various weights on the dynamometer and taking records.

The dynamometer is attached to a horizontal beam which extends over the mouth of the nozzle. The sting is suspended from the dynamometer by a knife-edge arrangement. This type of sting attachment is necessary to minimize damage to the dynamometer when negative forces are applied. The sting rod had to be stiff enough to keep the models from hitting the sides of the nozzle when the blowback occurred and small enough to get a reasonable sting tare. For these reasons a 5-foot drill rod 0.082 inch in diameter (about 1 percent of the body frontal area) was selected (see fig. 3). The models are attached to this sting rod through the nose. This type of mounting limits the measurements to that of drag at zero lift, and the models to be tested must also be stable. In figure 4 is shown a record of the drag of a body of fineness ratio 6 measured during a run. The sting tare is measured separately and has to be subtracted from the total drag. It usually amounts to about 25 percent of the total drag measured. From a record of the sting drag shown in figure 5, the momentum thickness of the boundary layer based on this measured sting drag corrected for the pressure drag on the end of the sting has been determined. Atmospheric pressure was assumed at the front and stream static pressure at the rear of the sting. The momentum thickness of the boundary layer was found to be 0.034 inch at a point where the nose of the model begins.

The momentum thickness of the boundary layer has also been calculated theoretically. The boundary layer on the sting was considered to be laminar to the throat of the nozzle and turbulent from the throat to the end of the sting because of a condensation shock which occurs in the low-supersonic region of the nozzle. By using the methods of references 2 and 3 the momentum thickness of the boundary layer was found to be 0.039 inch. This value is in good agreement with the value of 0.034 inch which was determined from the measured sting drag and is strong evidence that the assumed character of the flow along the sting is substantially correct. The difference between experimental and calculated thickness may possibly be explained by the fact that two-dimensional, incompressible-flow relations were used in the theoretical calculations. Unfortunately, more exact methods are not presently available.

Calculations, assuming laminar flow over the entire length of the sting, give a momentum thickness of the order of 0.0053 inch. This value is so much lower than that determined from the measured drag of the sting that it further substantiates the assumption of turbulent flow at the nose of the model.

The base pressure for the RM-10 body was obtained by using a pressure probe at the rear of the model, supported by a suitable rig in the extreme downstream end of the test section. This pressure probe was an open-end tube having its end placed about 3/8 inch from the base of the model and was connected to a nesting-diaphragm type pressure cell and recorder. The ratio of the tube diameter to the diameter of the base of the model was 0.313.

The models tested were two bodies of revolution, fineness ratios 6 and 12, four triangular-wing models of varying sweep angle, and an RM-10 body with stabilizing fins. The bodies of fineness ratios 6 and 12 were 5 and 10 inches long, respectively, and were both 5/6 inch in diameter at the maximum position. A photograph of these models, which are similar to those of references 4 and 5, is shown in figure 6. A line drawing of a triangular-wing model with diamond profile is shown in figure 7 and the dimensions of the models tested are tabulated. These triangular-wing models are the same ones that were tested in the Langley 9-inch supersonic tunnel and reported in reference 6. A drawing of an RM-10 body is shown in figure 8. This model is similar to the one of reference 7. All the models tested were small enough with respect to the test section that they were not affected by the bow shock wave which was reflected to the center of the nozzle about 13.3 inches behind the nose of the models.

RESULTS AND DISCUSSION

The setup and operation of this apparatus is sufficiently different from ordinary supersonic wind tunnels that considerable investigation could be made into the effects of these differences on the measurement of drag. For its intended purpose, however, absolute values are not required and confidence in its use is based on showing that a reasonable magnitude for drag and for drag differences is obtained. In evaluation of the measurements, the following points should be kept in mind:

(a) The time during which steady supersonic flow is obtained is about 0.03 second, sufficient for 45 feet of air to pass through the test section. Because the maximum model length is of the order of 1 foot, there should be ample time for the flow about the model to stabilize and for the boundary layer on the model to build up in 45-body-lengths travel (see reference 8). The drag records for models and for the sting support alone show that a steady value of drag is obtained during the available time.

(b) The method of mounting the model by a tension support through the nose allows correct simulation of details at the rear of a body. At the same time, an appreciable boundary layer is built up on the 4 feet of sting exposed to the air flow. It is assumed that this boundary layer is turbulent because of the pressure rise through the condensation shock shortly after the throat of the nozzle. The displacement area of the boundary layer is calculated to be about 3 percent of the frontal area of the bodies tested and would cause more modifications of the flow about the nose of blunt shapes than for pointed ones. With a turbulent boundary layer over the whole body, the large changes accompanying laminar separation at low Reynolds number would not be expected.

(c) The condensation that takes place in the nozzle reduces the Mach number by a maximum of 0.1 within the range of humidity encountered. The limited measurements in the test section gave no evidence of any flow irregularity, probably because the condensation shock is expected to take place shortly after the throat of the nozzle at a low-supersonic speed and should be normal to the flow.

The results of tests made on models of fineness ratios 6 and 12 are presented in figure 9. Measured values of the drag coefficient C_D (based on frontal area) are shown by the experimental points at a Mach number of approximately 1.5. These data indicate that the body of fineness ratio 12 has one-half the drag of the one of fineness ratio 6. There is limited scatter of repeat runs on different days despite the possibility of humidity effects. The range of relative humidity was from 40 to 75 percent for this series of tests.

The curves of C_D against M in figure 9 are from free-fall tests of similar bodies (references 4 and 9). The free-fall bodies differed from the models tested only in that they were stabilized by surfaces supported on a boom with a diameter one-fifth the maximum diameter of the body. The drag of the tail surfaces themselves was measured and had been subtracted from the over-all drag to obtain the curves in figure 9. From more recent information (reference 10), the drag-coefficient curve for the body of a fineness ratio 6 can be expected to reach a maximum at $M = 1.15$ and then decrease slightly at higher Mach numbers. It can be seen in figure 9 that the drag-coefficient curve for the body of fineness ratio 12 has already reached its maximum value and is gradually decreasing as higher Mach numbers are reached. Based on the preceding remarks, an extrapolation from $M = 1.15$ through the experimental points would seem reasonable. Therefore, absolute values of C_D and the difference in drag coefficient of the two models are considered to be in reasonably good agreement with the free-fall measurements.

A comparison is made in figure 10 of drag data obtained on four triangular-wing models in the Langley 9-inch supersonic tunnel and with the apparatus of this paper. To compare these data obtained at different Mach numbers on the same chart (reference 11), a reasonable value of skin-friction coefficient ($C_f = 0.006$) is subtracted from the total drag coefficient C_{D_T} and the remainder is multiplied by the appropriate β $\left(\beta = \sqrt{M^2 - 1}\right)$. The value βC_D is then plotted against $\tan \epsilon/\tan m$ in figure 10 for the Langley 9-inch-supersonic-tunnel data at Mach numbers of 1.62, 1.92, and 2.40 and for the apparatus of this paper at $M = 1.42$. An average curve is then faired through the 9-inch-supersonic-tunnel data and through $\frac{\tan \epsilon}{\tan m} = 0$. As can be seen by the plot, the drag values obtained with this apparatus are slightly low.

As a further comparison, C_{D_T} is plotted against Mach number in figure 11 for the four triangular-wing models. The experimental values are plotted as points, whereas the curves were obtained by taking values of βC_D from the faired curve of figure 10. The Langley 9-inch-supersonic-tunnel data are in close agreement with the curves of figure 11 but again the drag values obtained with the apparatus of this paper are slightly low.

Tests have been made on an RM-10 body of fineness ratio 12.2 (see fig. 8) with stabilizing tail fins. Measured values of the total drag coefficient C_{D_T} and the base drag coefficient C_{D_B} are plotted in figure 12 at $M = 1.4$ (Reynolds number = 4.7×10^6) for the apparatus of this paper and $M = 1.59$ (Reynolds number = 3.7×10^6) for the Langley 4-foot supersonic tunnel (unpublished results). The curves of C_{D_T} and C_{D_B} against M in figure 12 are from rocket-propelled tests of a similar body (reference 7). The Reynolds number for the full-scale model varies from 35×10^6 to 210×10^6 and that of the half-scale model varies from 25×10^6 to 100×10^6.

The results obtained from the Langley 4-foot supersonic tunnel in figure 12 are for an RM-10 body with normal and fully turbulent boundary layer over the body. From detailed measurements made of the component parts of the total drag, it appears that the main difference between the total-drag values for the normal and turbulent cases is that the skin friction for the fully turbulent case is more than four times that of the normal case which is presumed to have a fully laminar boundary layer.

For the full-scale rocket-propelled RM-10 body, the boundary layer over most of the body would tend to be turbulent. At a flight Reynolds number of 75×10^6, the drag coefficient would be less than for the case of the model with a fully turbulent boundary layer in the 4-foot tunnel.

The drag measurement from the 4-foot supersonic tunnel for the turbulent-boundary-layer case was actually taken under conditions quite similar to those obtained with the apparatus of this paper. The nominal Reynolds number is practically the same for the two tests. In the 4-foot-supersonic-tunnel tests, the model was mounted on a wire stretched through the test section and the drag force was measured with a balance built into the model. The only significant differences in the two tests are that the wire support was somewhat smaller in proportion to the model (65 percent of the diameter) for the 4-foot-supersonic-tunnel test and the 4-foot supersonic tunnel had no disturbance in the tunnel corresponding to the condensation shock of this apparatus to cause transition of the boundary layer on the sting an appreciable distance ahead of the model. The difference in drag between the two tests is in the direction expected, if a smaller turbulent boundary layer were present at the nose of the model, in the 4-foot-tunnel tests, and the value of drag obtained with the apparatus of this paper is within the range shown to be possible by changing the boundary-layer characteristics.

On the basis of comparisons made in figures 9 to 12 between tests made using the apparatus of this paper and other facilities, it seems that the data obtained with the apparatus of this paper are reliable enough for this equipment to be used in screening new ideas.

CONCLUDING REMARKS

Data obtained from model tests when compared with data obtained by free-fall, rocket-propelled, and supersonic-tunnel test methods show that the drag measured with this test apparatus is reasonable. They also show that the effect of changes in body shape can be satisfactorily established.

This relatively simple and quick means of comparing the merits of aerodynamic shapes in supersonic flows should prove useful for further research, particularly in industry and universities.

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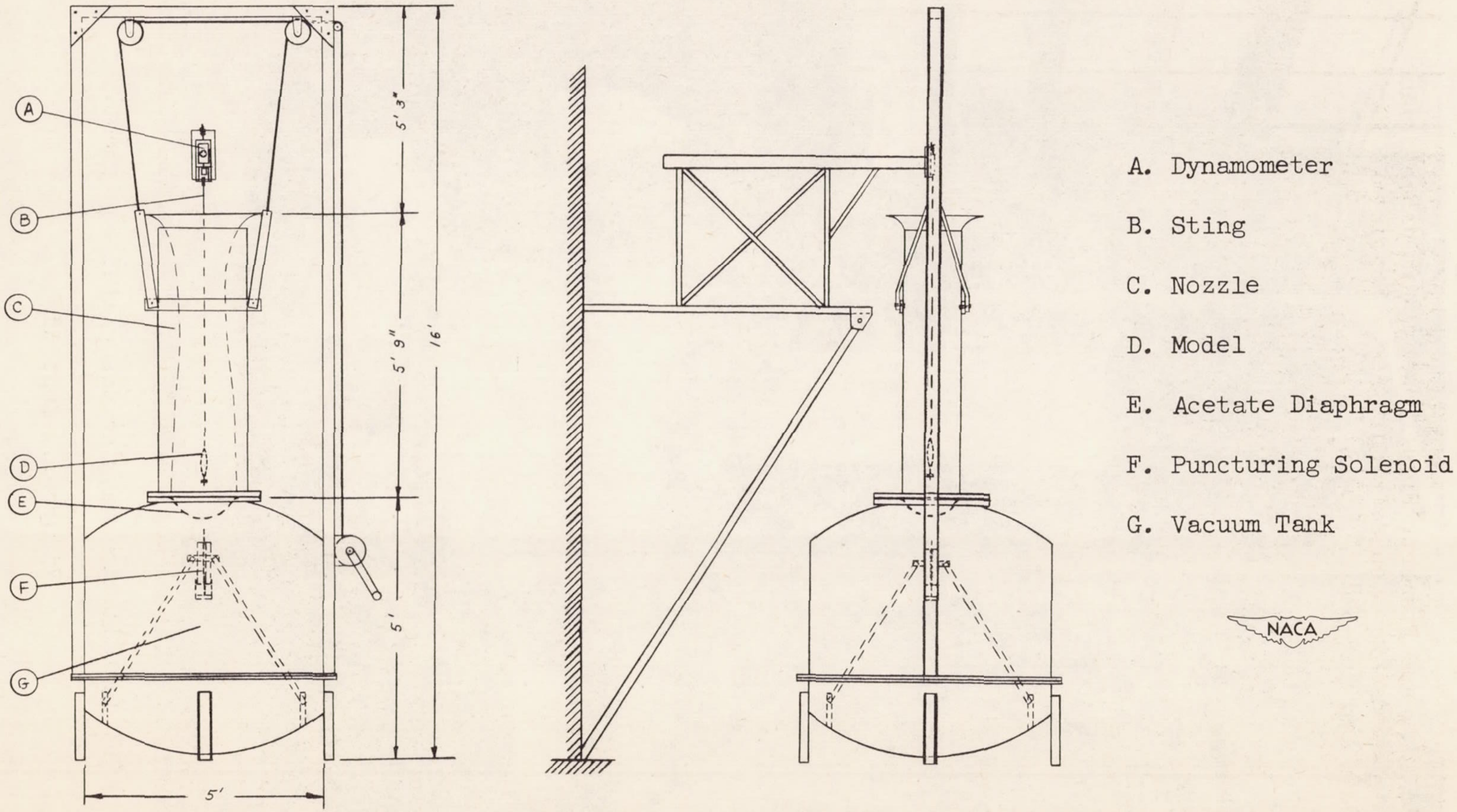

(a) Schematic diagram.

Figure 1.- Vacuum-actuated supersonic nozzle.

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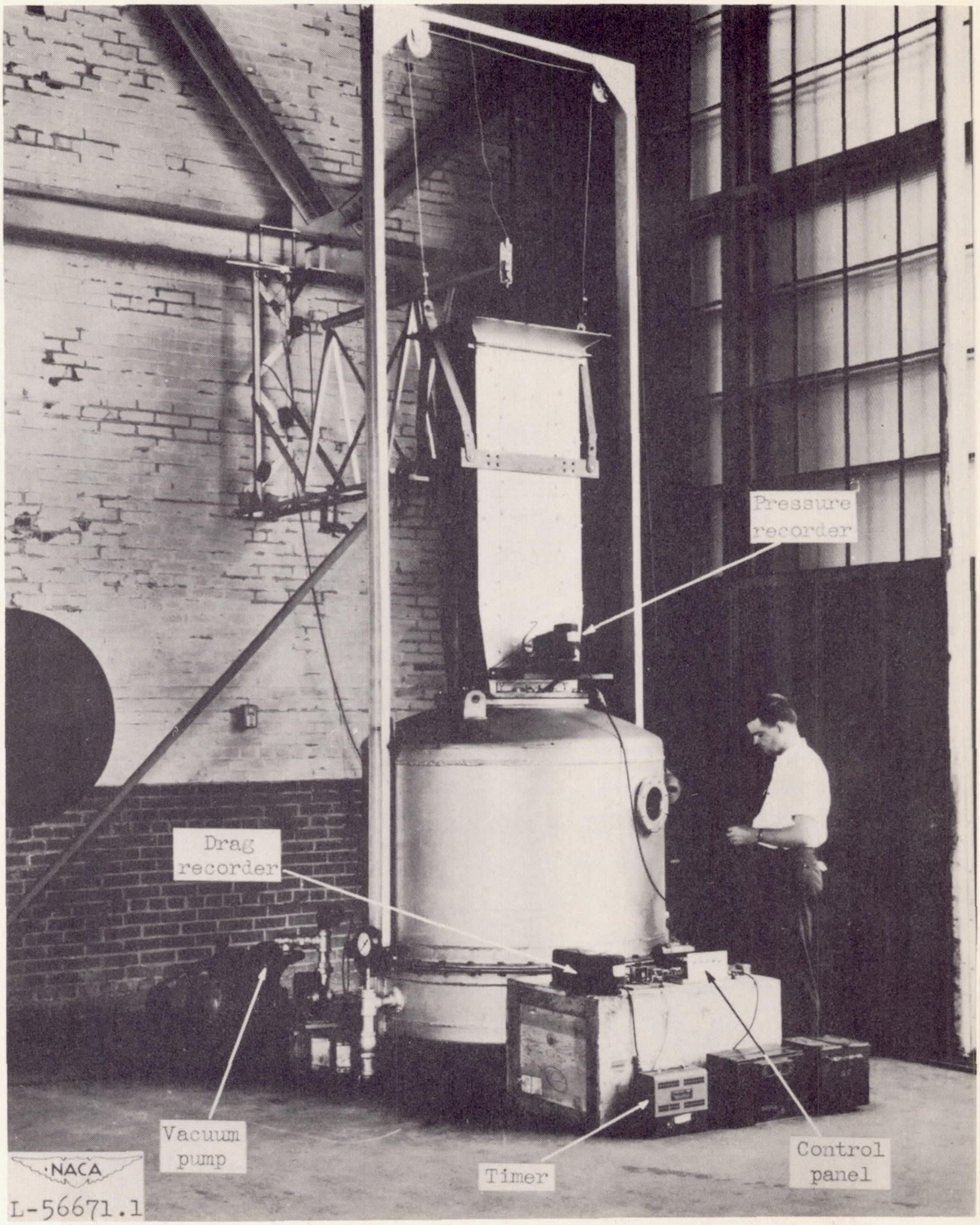

(b) Installation showing associated equipment.

Figure 1.- Concluded.

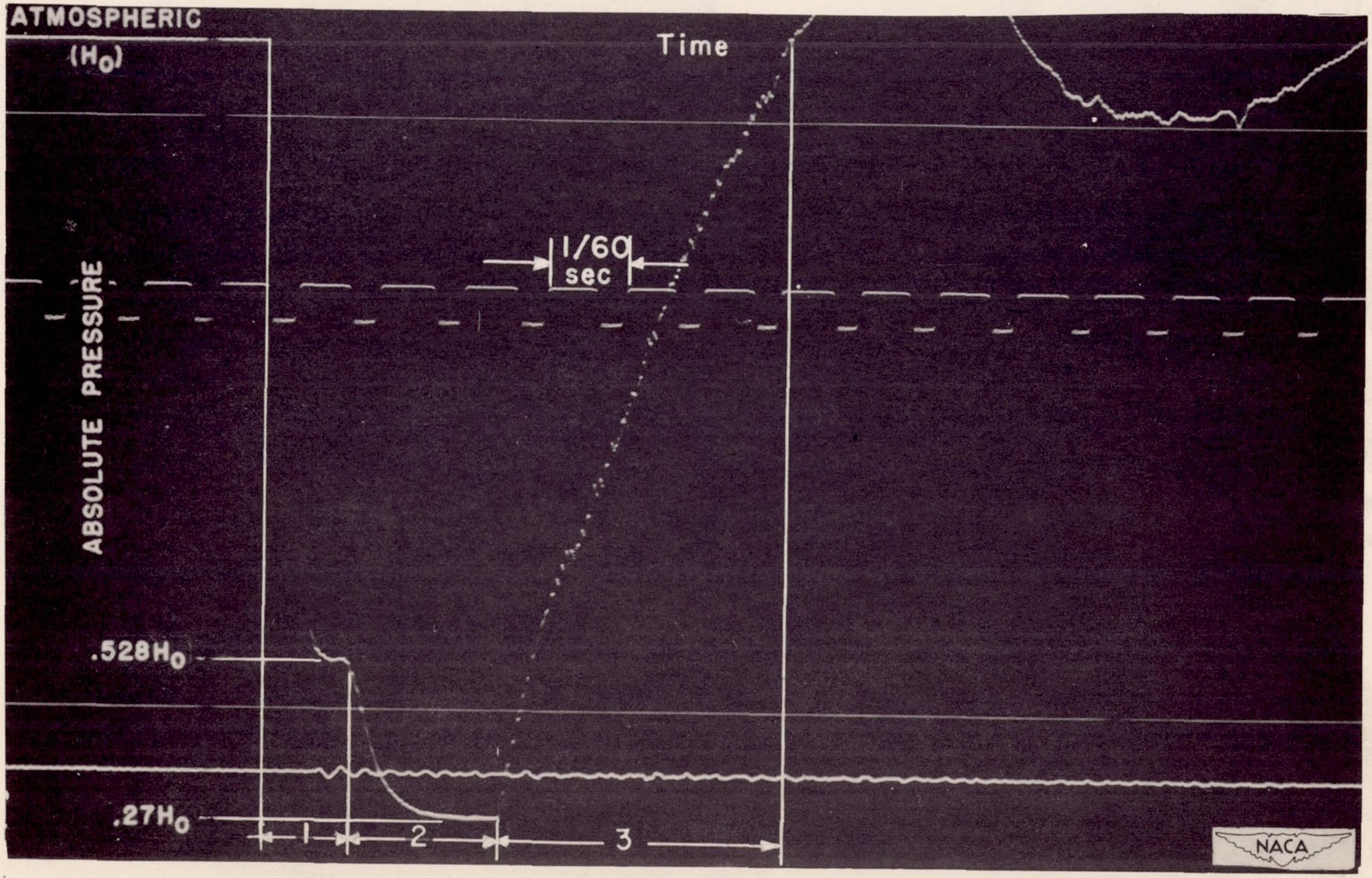

Figure 2.- Time history of static pressure in test section of nozzle during a run.

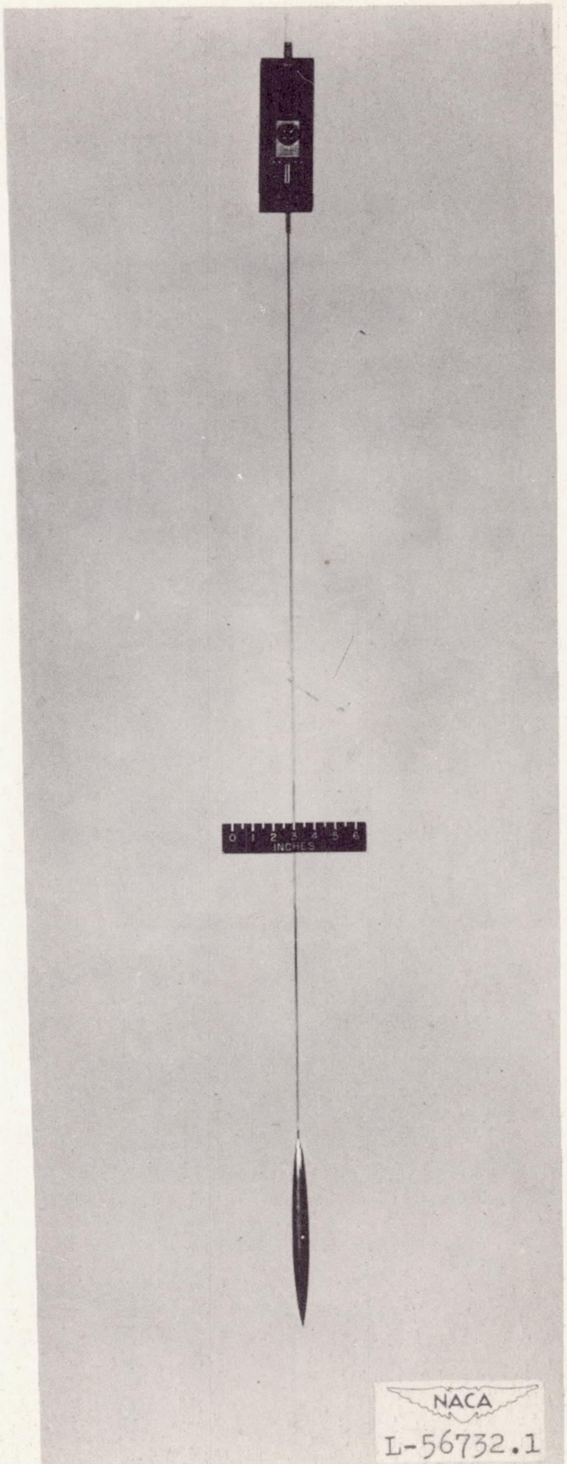

Figure 3.- Sting showing attachment to dynamometer and model.

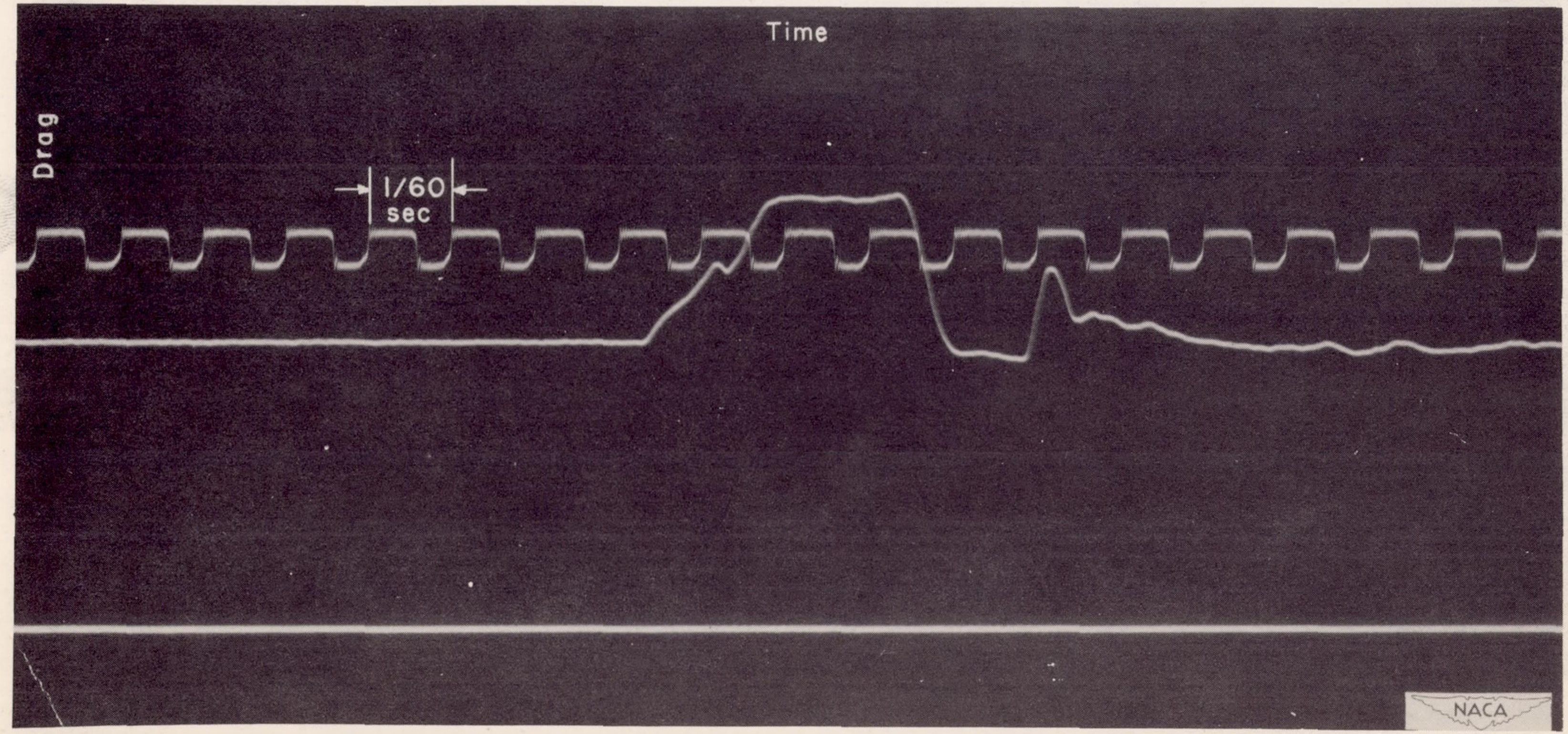

Figure 4.- A record of the drag of a body of revolution measured during a run.

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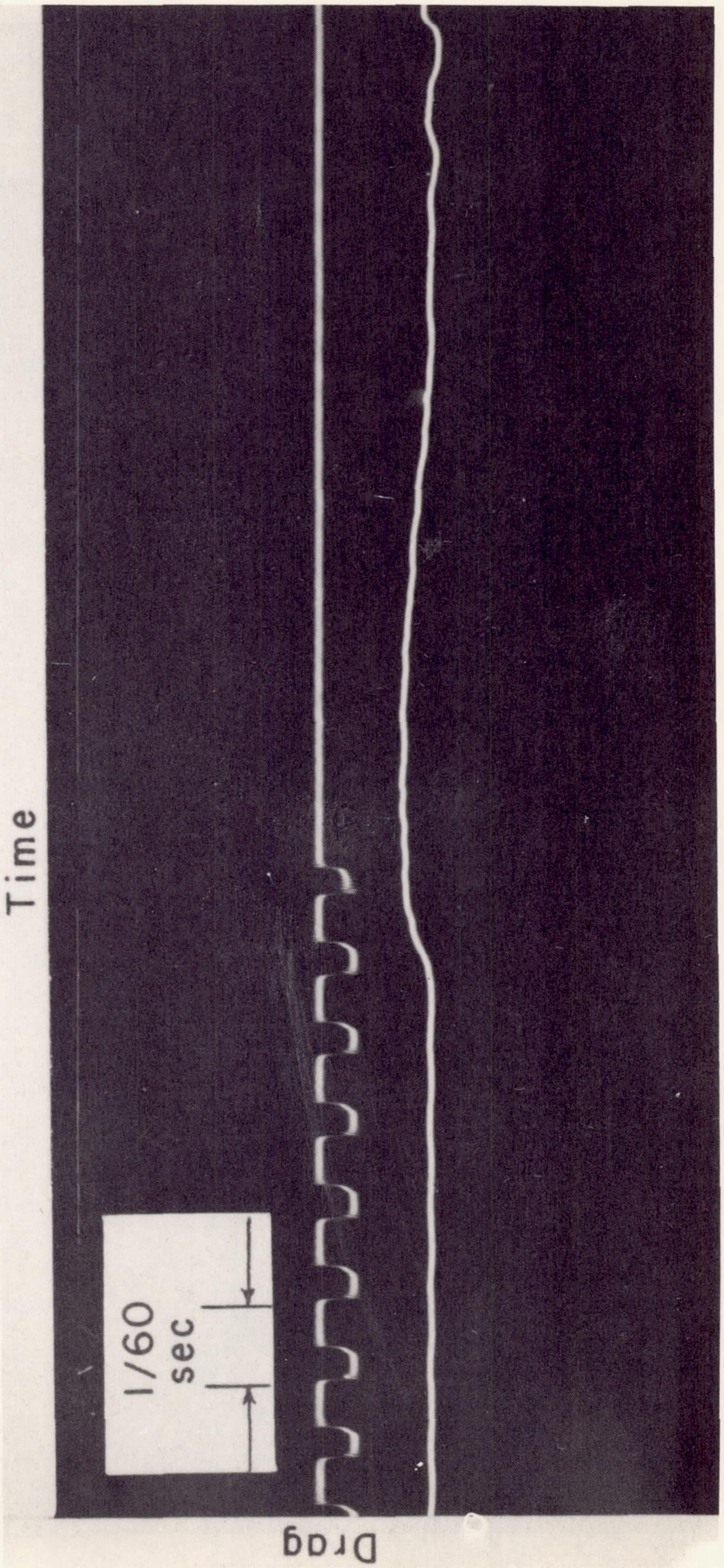


Figure 5.- A record of the sting drag measured during a run.

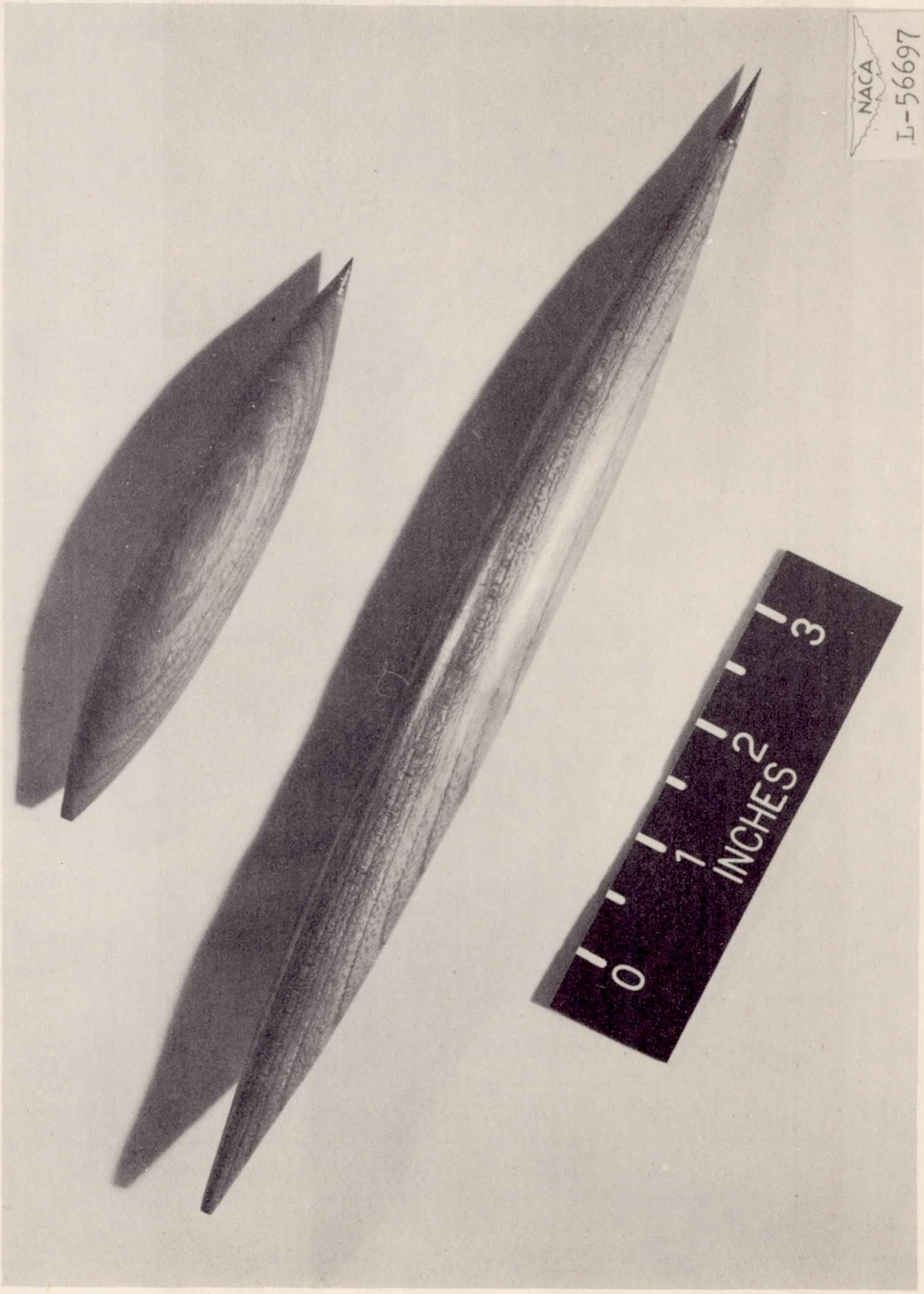
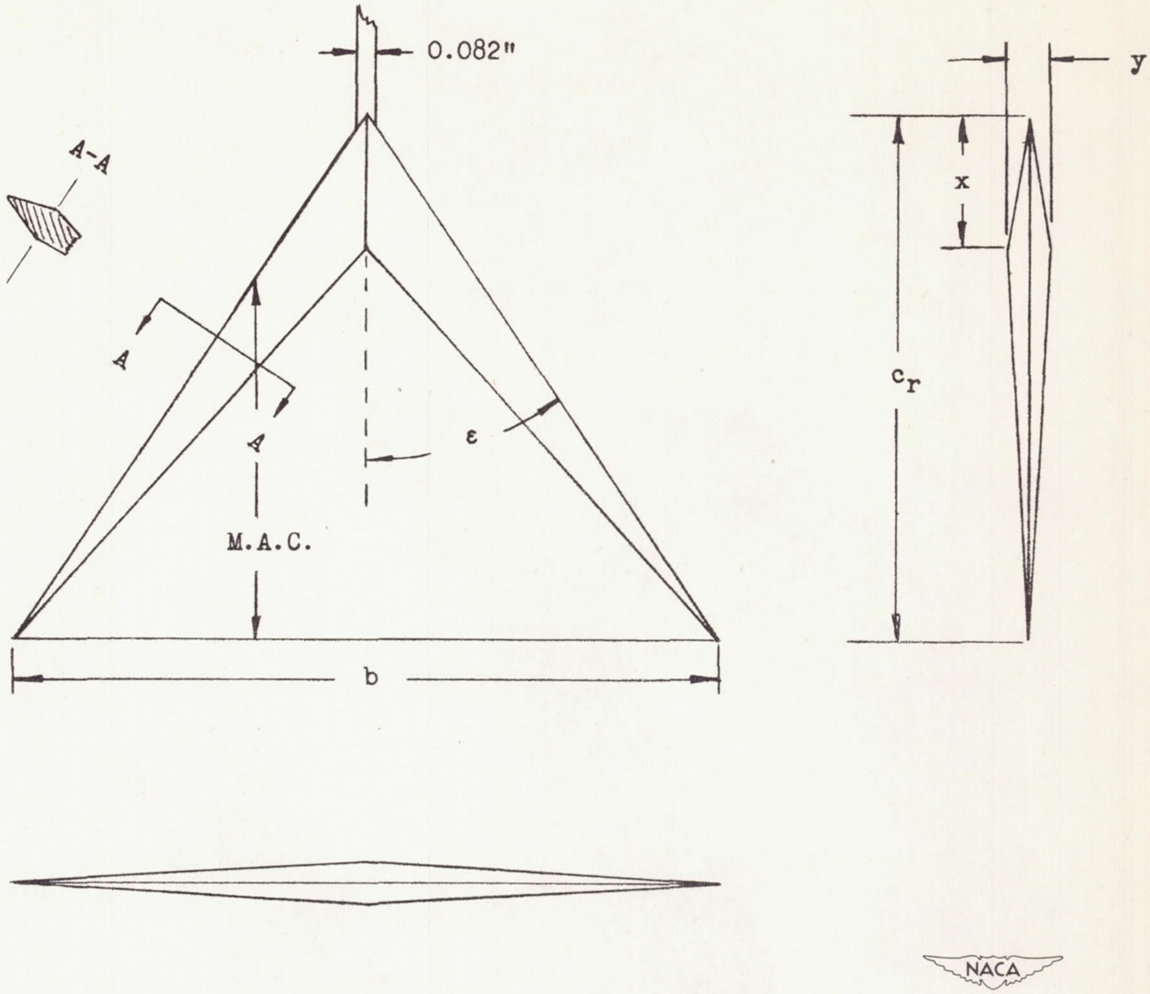


Figure 6.- Bodies of fineness ratio 6 and 12.



Wing	b (ft)	c_r (ft)	ε (deg)	x (percent chord)	y (percent chord)	M.A.C. (ft)	Area (sq ft)	Aspect ratio
4	0.402	0.431	25.01	0.18	0.08	0.287	0.0867	1.869
6	.413	.360	29.84	.18	.08	.240	.0743	2.301
8	.433	.307	35.21	.18	.08	.205	.0665	2.812
10	.444	.265	39.92	.18	.08	.177	.0588	3.350

Figure 7.- Dimensions of 8-percent-thick triangular wing models.

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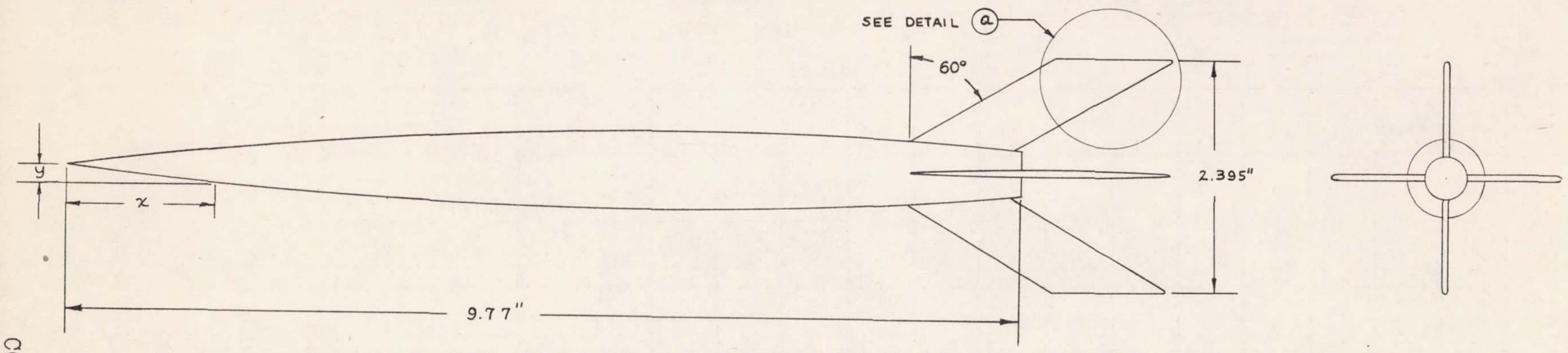

BODY COORDINATES											
x	y	x	y	x	y	x	y	x	y	x	y
0	0	.8	.100	2.6	.272	4.6	.378	6.2	.399	8.2	.346
.1	.013	1.0	.122	3.0	.300	5.0	.389	6.6	.396	8.6	.325
.2	.026	1.4	.165	3.4	.325	5.4	.396	7.0	.389	9.0	.300
.4	.052	1.8	.204	3.8	.346	5.8	.399	7.4	.378	9.4	.272
.6	.076	2.2	.240	4.2	.364	6.0	.400	7.8	.364	9.77	.242

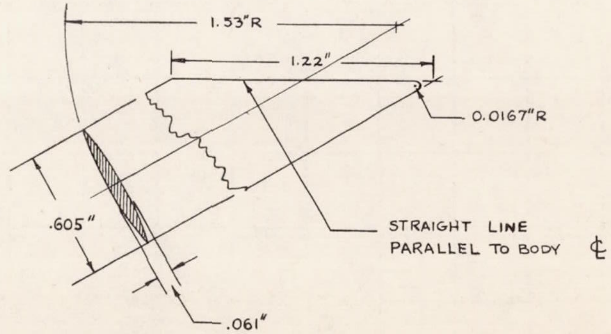

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Figure 8.- RM-10 research model.

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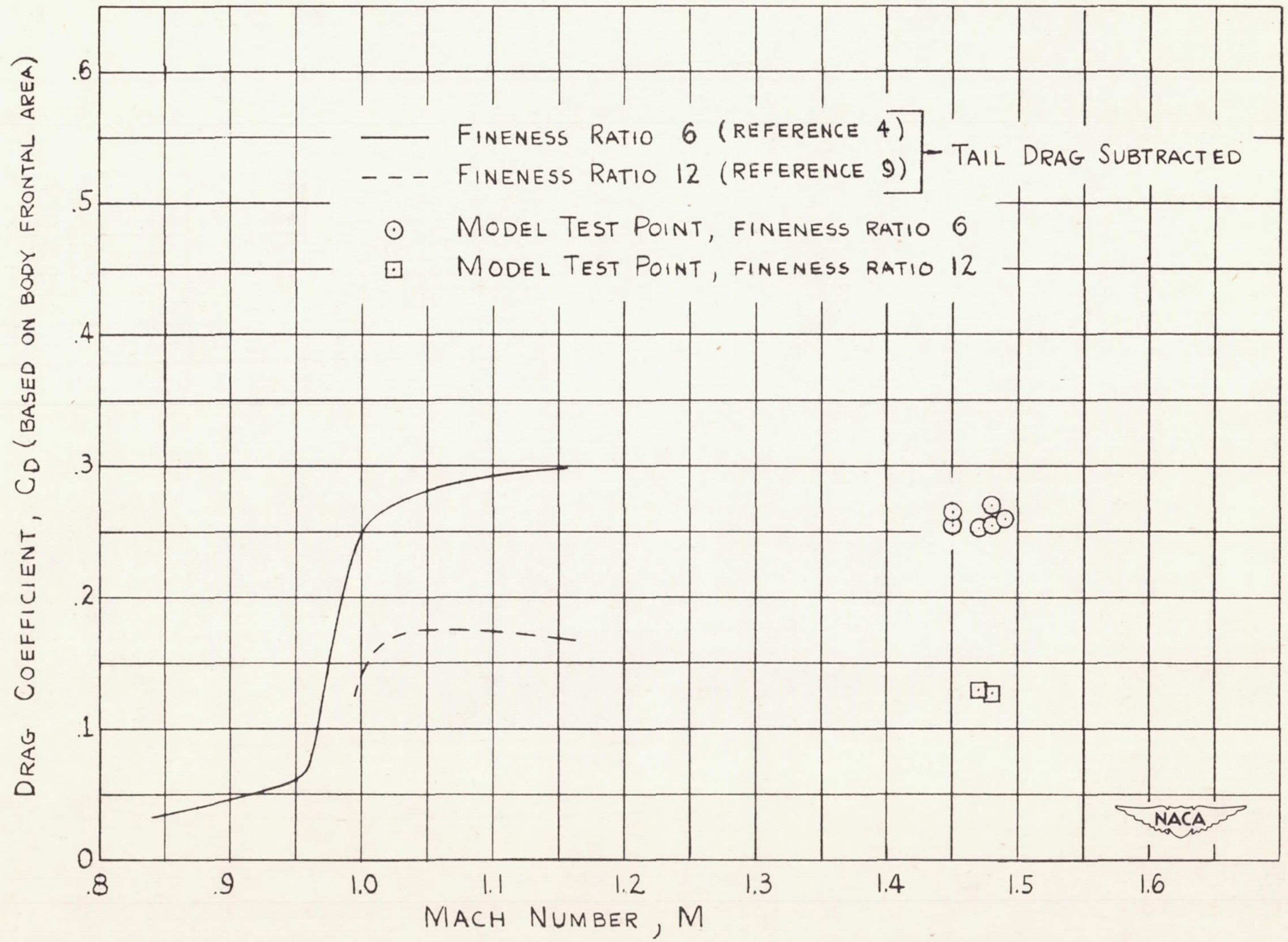

Figure 9.- A comparison of model tests and free-fall tests with tail drag subtracted.

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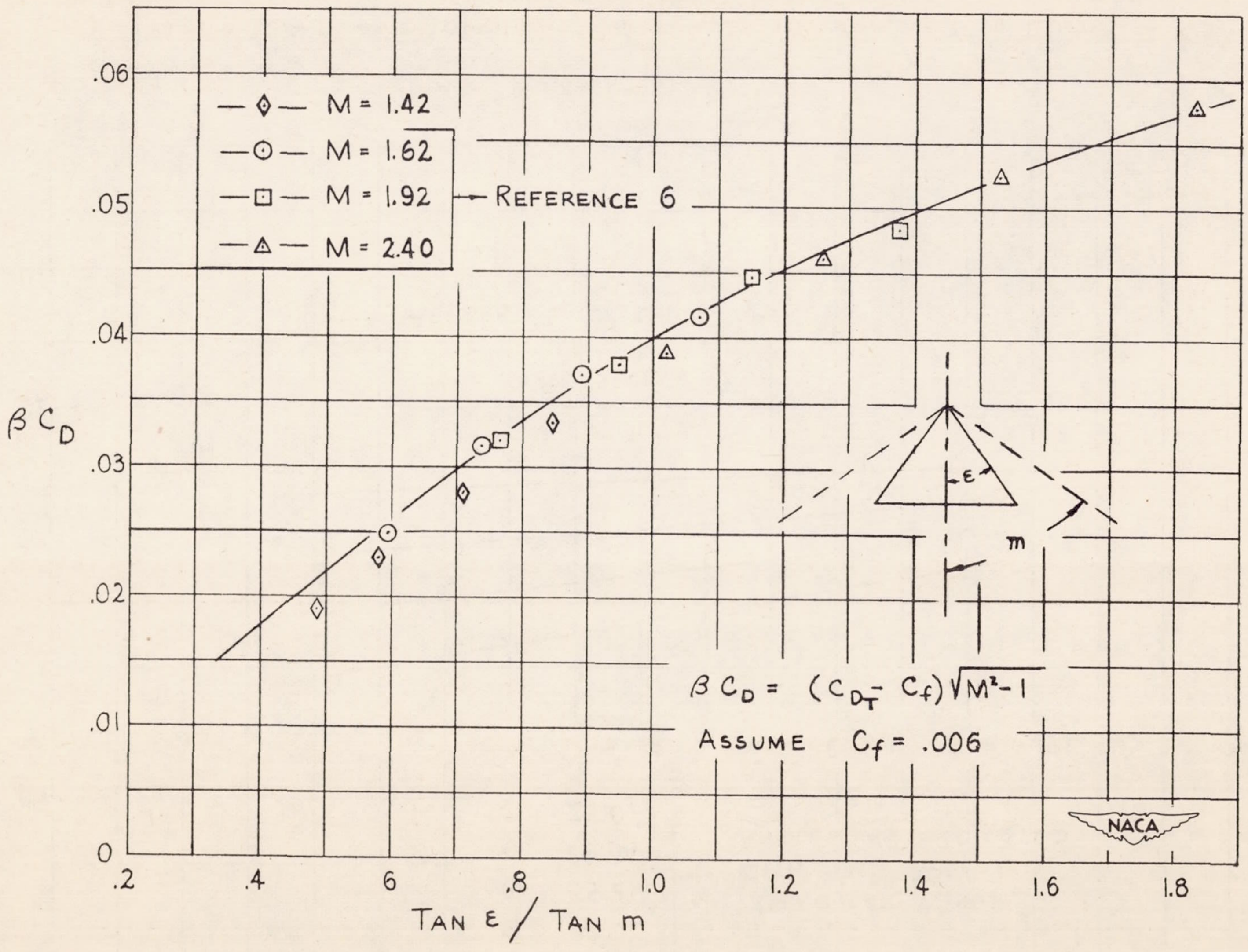

Figure 10.- General comparison of the drag of four 8-percent-thick triangular wings.

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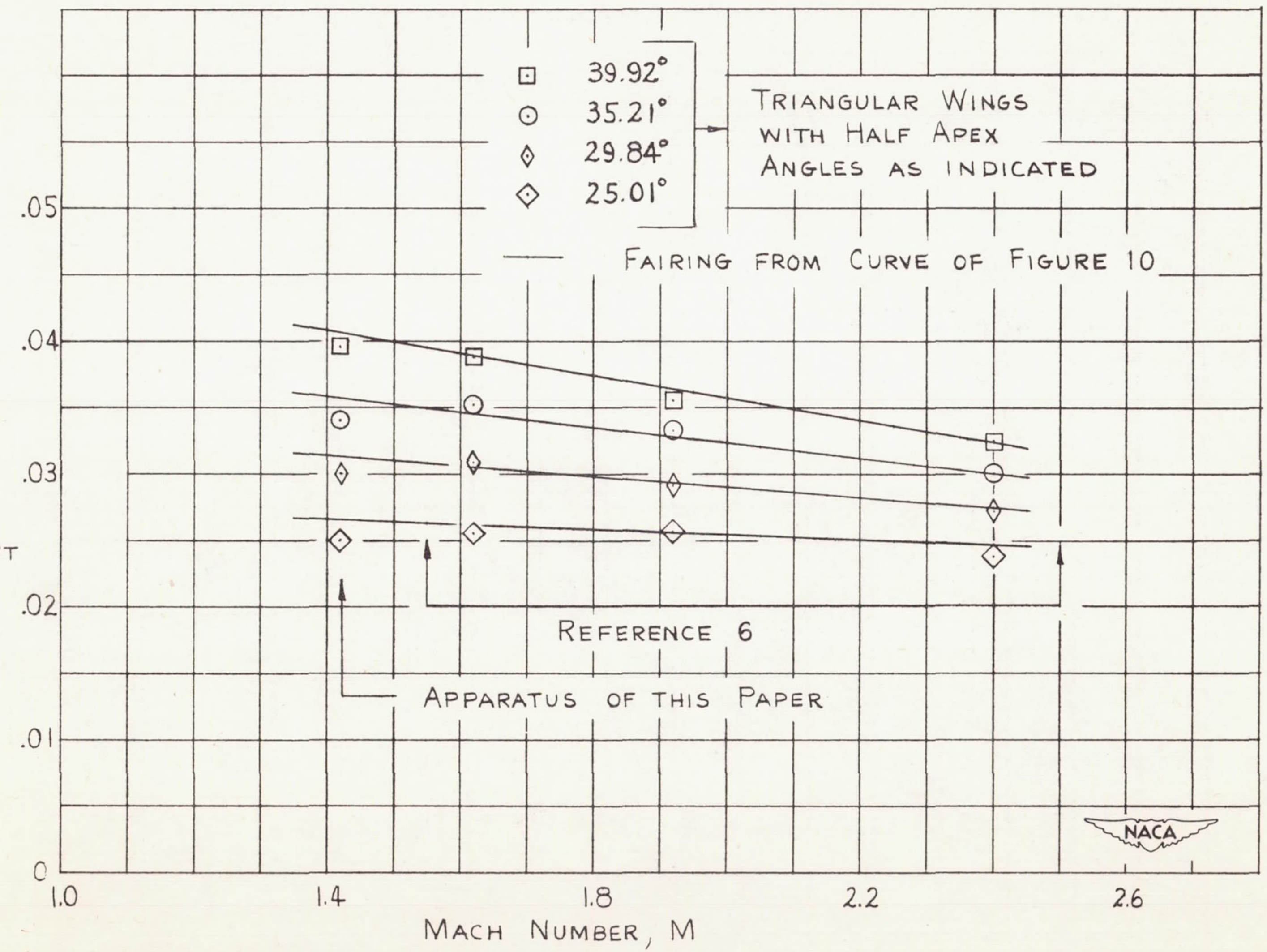

Figure 11.- An extrapolation of the results of reference 6 to lower Mach numbers.

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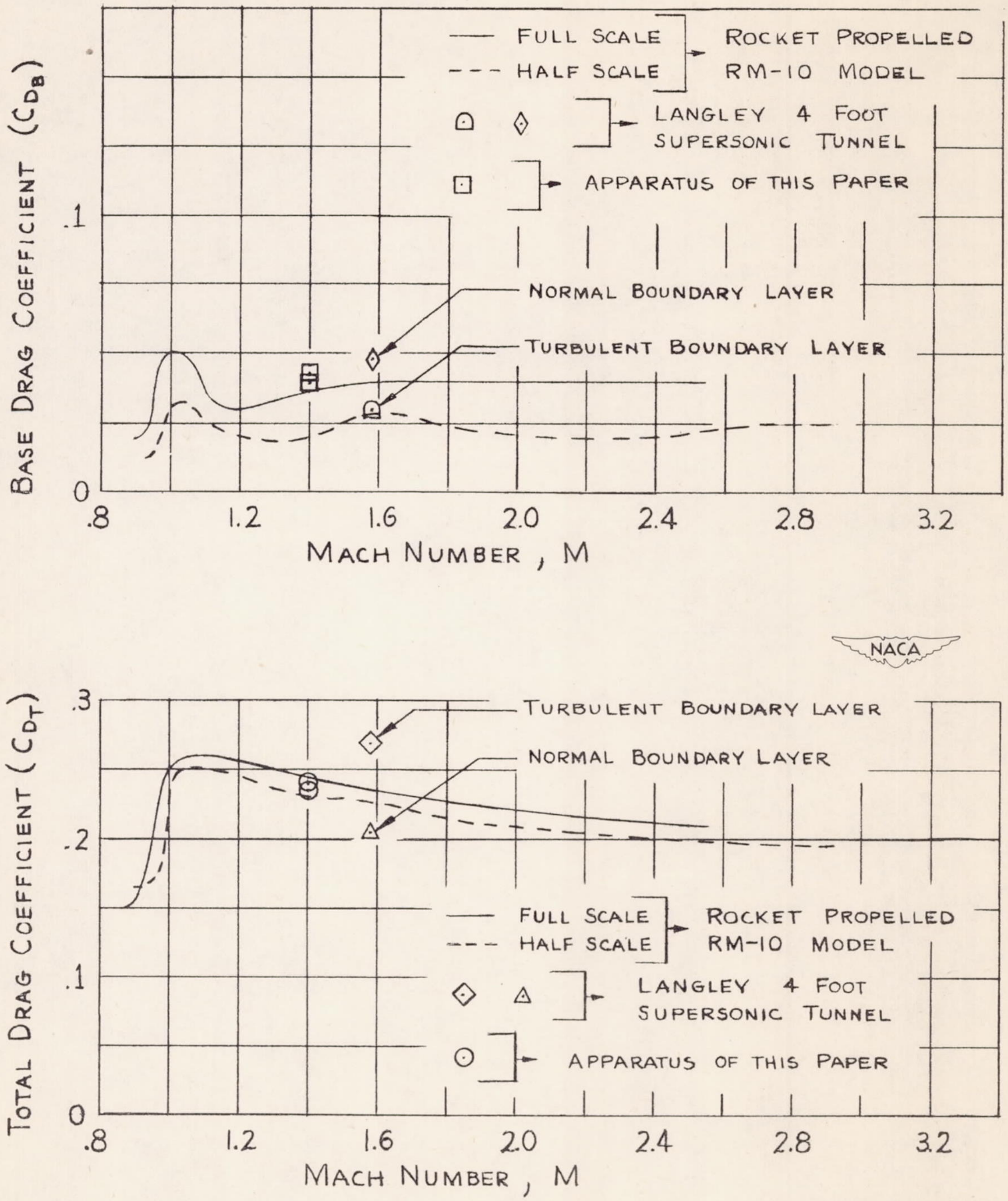

Figure 12.- A comparison of the drag results for an RM-10 model obtained using the apparatus of this paper and other facilities.

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