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MEASUREMENTS OF THE MACH NUMBER AND PRESSURE ON THE AFTERBODY OF A BLUNT-NOSED MODEL IN FREE FLIGHT AT MACH NUMBERS FROM 6.5 TO 14

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TECHNICAL MEMORANDUM X-360

MEASUREMENTS OF THE MACH NUMBER AND PRESSURE ON THE AFTERBODY OF A BLUNT-NOSED MODEL IN FREE FLIGHT AT MACH NUMBERS FROM 6.5 TO 14*

By Simon C. Sommer

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SUMMARY

A blunt-nosed model was tested in free flight in the Ames Supersonic Free-Flight Wind Tunnel at Mach numbers from 6.5 to 14 to determine the local flow properties at one position on the afterbody. It was found that the local Mach number was essentially invariant with free-stream Mach number, whereas the pressure ratio (local to free-stream static) was found to be extremely sensitive to Mach number. These data in combination with data from the Naval Ordnance Laboratory Pressurized Ballistic Range show that the pressure ratio varies from about 0.25 to 4 as Mach number is increased from about 3.5 to 14.

Total drag and static stability were also measured.

INTRODUCTION

For vehicles flying at extremely high speeds, the pressure level on the sides or afterbody can greatly influence the design of the vehicle. If high pressure levels are present, which indicate high heat transfer to the surface of the body, adequate provision for heat protection must be made in the vehicle design. Calculations by Ferri and Pallone (ref. 1) indicate that the pressures on the afterbody of a blunt vehicle with a strong shock wave can be considerably higher than free-stream static pressure. The reason for this, according to reference 1, is the large entropy increase caused by the strong bow shock wave. These effects are expected to become more important with increasing Mach number.

The local flow properties on the afterbody of a blunt-nosed vehicle at supersonic speeds were measured in the NOL Pressurized Ballistic Range

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at Mach numbers from about 3 to 6.[1] Similar models were tested in the Ames Supersonic Free-Flight Wind Tunnel at Mach numbers up to 14, in order to determine the effect of a large increase in Mach number on the local flow properties. These latter tests were limited to the determination of the local static pressure and Mach number at one location on the afterbody of the model. The results of these tests are the subject of this paper.

SYMBOLS

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C_D	total-drag coefficient, dimensionless
C_{m_α}	pitching-moment-curve slope, per radian
p_l	local static pressure, lb/sq in.
p_∞	free-stream static pressure, lb/sq in.
M_l	local Mach number
M_∞	free-stream Mach number
x	axial distance along body from stagnation point, in.
l	total length of body, 1.075 in.

DESCRIPTION OF TESTS

The tests were conducted in the Ames Supersonic Free-Flight Wind Tunnel (ref.2). This facility is a variable-pressure, supersonic, blowdown wind tunnel with nine shadowgraph stations 3 feet apart. Models are fired upstream through either still air or through a Mach number 2 or Mach number 3 air stream. For the tests described herein, models were fired from a shock-heated helium gun (described in ref.3) into both still air and a Mach number 3 air stream at approximately the same muzzle velocity. The nominal free-stream Mach numbers for the two conditions were 6.6 and 14.2 and the nominal free-stream Reynolds numbers were 4.5 and 5.5 million, based on maximum diameter.

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[1]These data are unpublished and were made available by the Naval Ordnance Laboratory, White Oaks, Silver Spring, Maryland.

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MODELS

The model tested is shown in figure 1. It consisted of a spherical cap, followed by a very short cylindrical section and a double-angle boattail. The spherical cap on the rear of the model (shown by the dashed line) was removed, and the threaded spike was added to hold the model in a sabot. Two rings were machined onto the afterbody of the model by a rolling process. The height of these rings was between 0.0015 and 0.0020 inch. The purpose of the rings was to produce Mach lines in the flow, from which the Mach angle could be read and the local Mach number and pressure deduced. The distances to the two rings measured from the stagnation point, and made dimensionless with respect to the total length of the body, x/l, were 0.51 and 0.84.

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DATA REDUCTION

Local Mach numbers were determined from measurements of the Mach angles on the afterbody of the model. These measurements and the free-stream Mach number were then used to calculate the local pressure on the assumption that the total pressure at the measuring station was the total pressure behind a normal shock wave. This technique of determining local pressure and Mach number was also used in the NOL tests.

Drag and static stability of the models were also determined. Drag coefficients were calculated from the measured model deceleration, and the pitching-moment-curve slope about the model's center of gravity was determined from the measured pitching frequencies. These data reduction procedures are discussed in reference 2.

RESULTS AND DISCUSSION

Local Flow Measurements

At the lower Mach number tested ($M_\infty \approx 6.5$), the Mach lines were clearly visible in almost all shadowgraphs, whereas only two shadowgraphs had clearly defined Mach lines at $M_\infty = 14$. A shadowgraph of a model in flight at a Mach number of 6.8 is shown in figure 2. The rear Mach line ($x/l = 0.84$) is very close to the position where the flow over the afterbody separates, and the Mach angle may have been influenced by the flow separation. At the higher Mach number tested, the rear Mach line is not visible. For these reasons, local flow measurements reported herein are for the forward ring position, $x/l = 0.51$, only.

Local Mach number.- Local Mach numbers were measured at nominal free-stream Mach numbers of 6.5 and 14, and are compared with the NOL data at the same value of x/l in figure 3. Although there is some scatter in the data below a Mach number of 6, the local Mach number appears to be insensitive to changes in free-stream Mach number. This limited set of data could lead one to the conclusion, then, that the local Mach number on the afterbody of a high-speed vehicle similar to the present test model could be determined from tests of the configuration at low supersonic speeds. This conclusion should be considered tentative, since only one set of data is available over a wide Mach number range and for only one position on the afterbody.

Local pressure ratio.- Pressures, in the form of the ratio of local to free-stream static pressure, were calculated from the local Mach number measurements and are compared with the corresponding NOL data in figure 4. Note that the pressure ratio remains below 1 until the free-stream Mach number is about 6.5. At a Mach number of 14, the local pressure is about 4 times the free-stream static pressure. (The solid curve on the figure is a faired curve through the data.) It is interesting to note that the large changes in static pressure on the afterbody at Mach numbers above 6 could not be anticipated from the results of the NOL tests.

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The data in the figure were not compared with Newtonian theory because Newtonian theory does not expressly give a value for the local flow properties in the region of the body not facing the flow. The theory intimates, however, that either the pressure coefficient or the local static pressure will be zero. In any case, the Newtonian prediction for pressure ratio would be between 1 and 0, a considerable underestimate. Calculations of the local pressure ratio were also made from blast wave theory of reference 4. The calculated values overestimated pressure ratio by about a factor of 7 in the Mach number range from 6 to 14 and are not shown in the figure. Neither of these approximate calculation methods predicts any dependence of pressure ratio on afterbody shape.

Drag and Static Stability

Drag coefficients.- Total-drag coefficients were measured and are compared with NOL data and modified Newtonian theory in figure 5. The agreement of all the data is considered good. Newtonian prediction is also in good agreement with the measurements, about 5 percent high.

Static stability.- Static stability, in the form of C_{m_α}, was also measured and is compared with modified Newtonian theory in figure 6. The agreement is again good, with the Newtonian prediction about 5 percent low. No data were available for comparison from NOL tests.

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CONCLUDING REMARKS

A highly blunt model was tested at Mach numbers from 6.5 to 14 in the Ames Supersonic Free-Flight Wind Tunnel to determine the local flow properties at one position on the afterbody.

By comparison of data from the present investigation with data at lower Mach numbers from the NOL Pressurized Ballistic Range, the local Mach number was found to be insensitive to free-stream Mach number, whereas the pressure ratio (local to free-stream static pressure) was found to be extremely sensitive to free-stream Mach number. At a Mach number of 14, the pressure ratio was about 4, whereas at Mach numbers below 6, the pressure ratio was below 1.

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Total-drag coefficients and static stability were also measured. Prediction of these quantities by modified Newtonian theory was in good agreement with the measured data.

Ames Research Center
National Aeronautics and Space Administration
Moffett Field, Calif., Nov. 25, 1959

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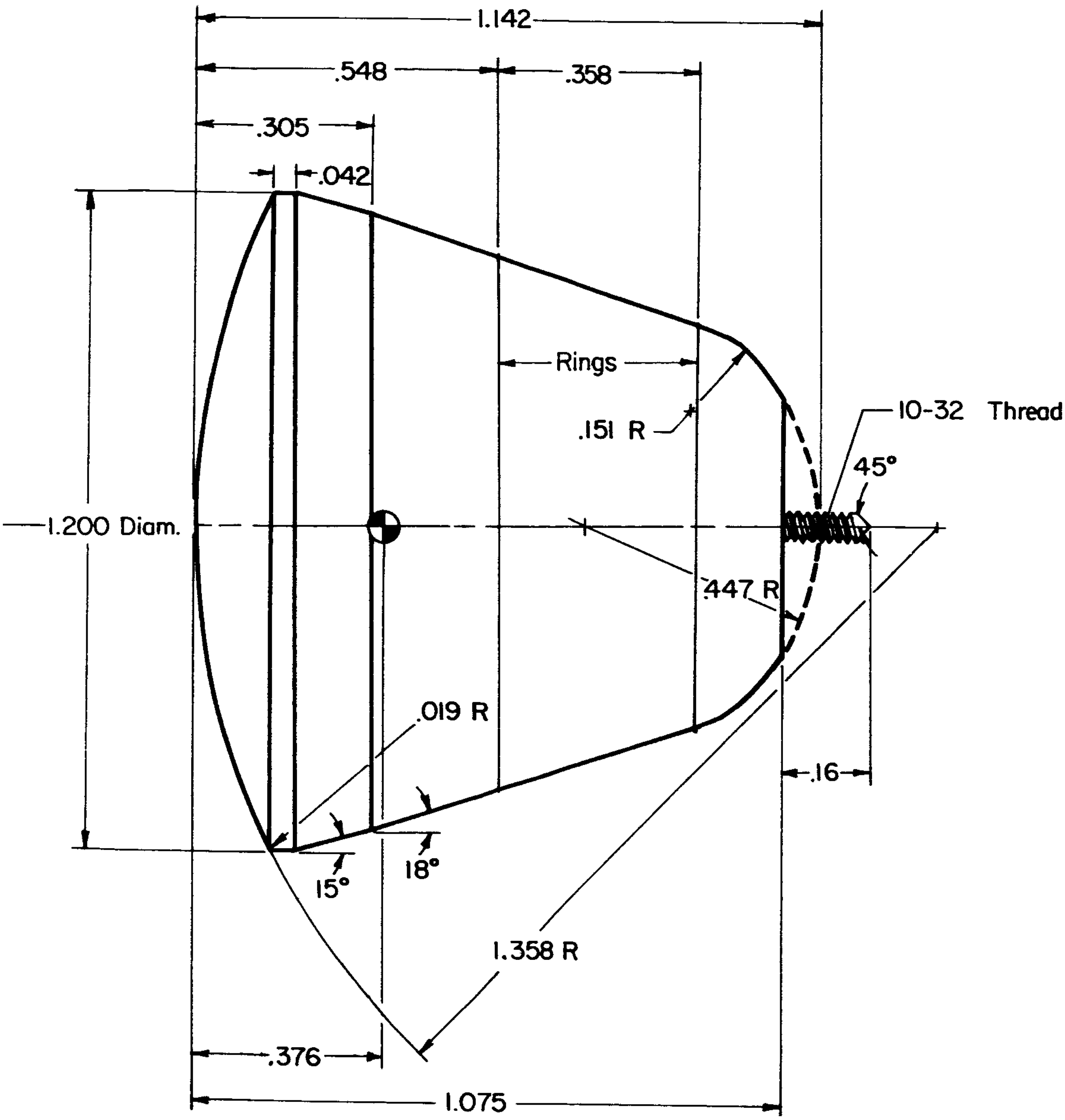

Figure 1.- Test model; all dimensions in inches unless otherwise noted.

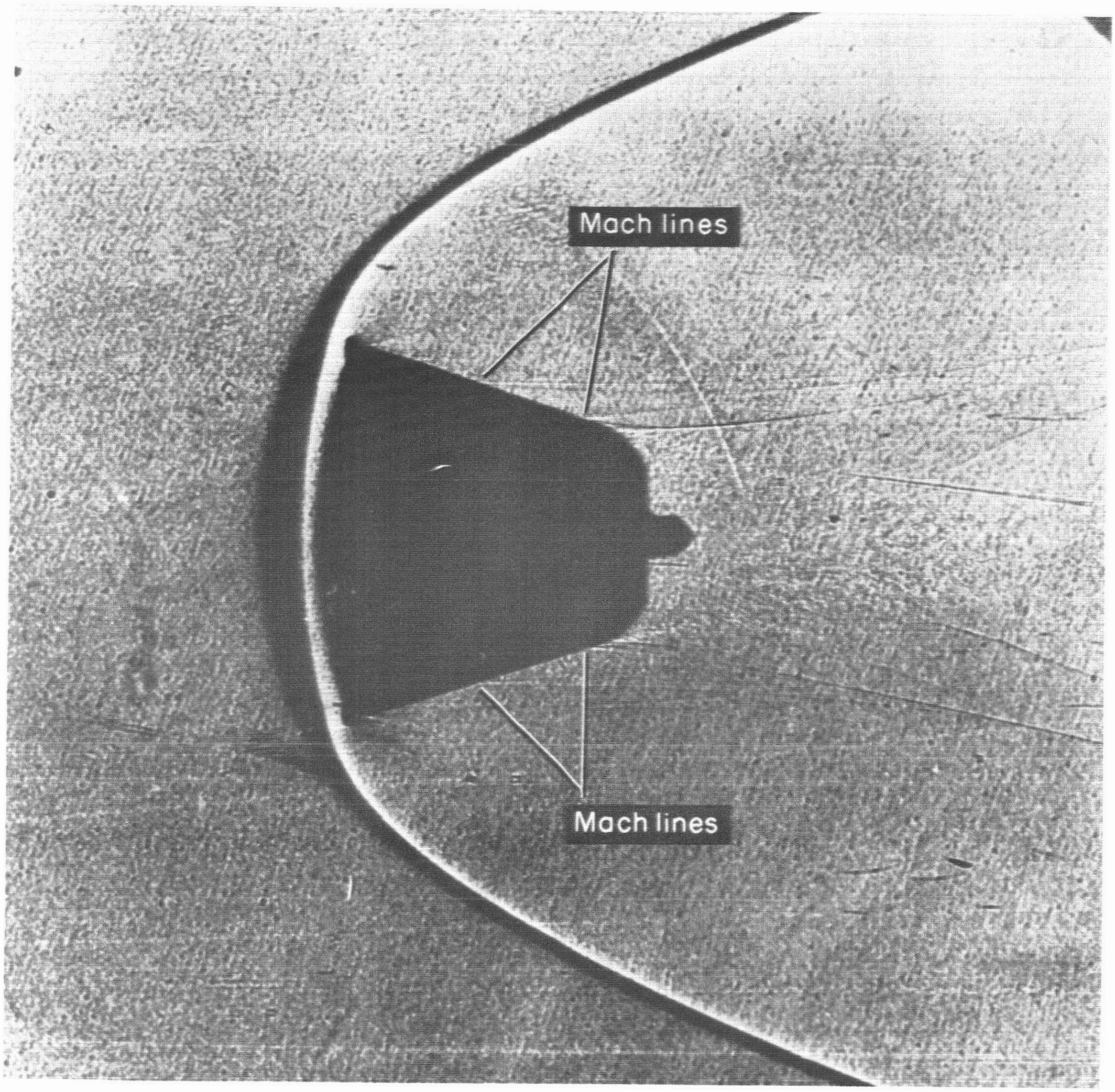

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Figure 2.- Shadowgraph of model in flight at $M_\infty = 6.8$.

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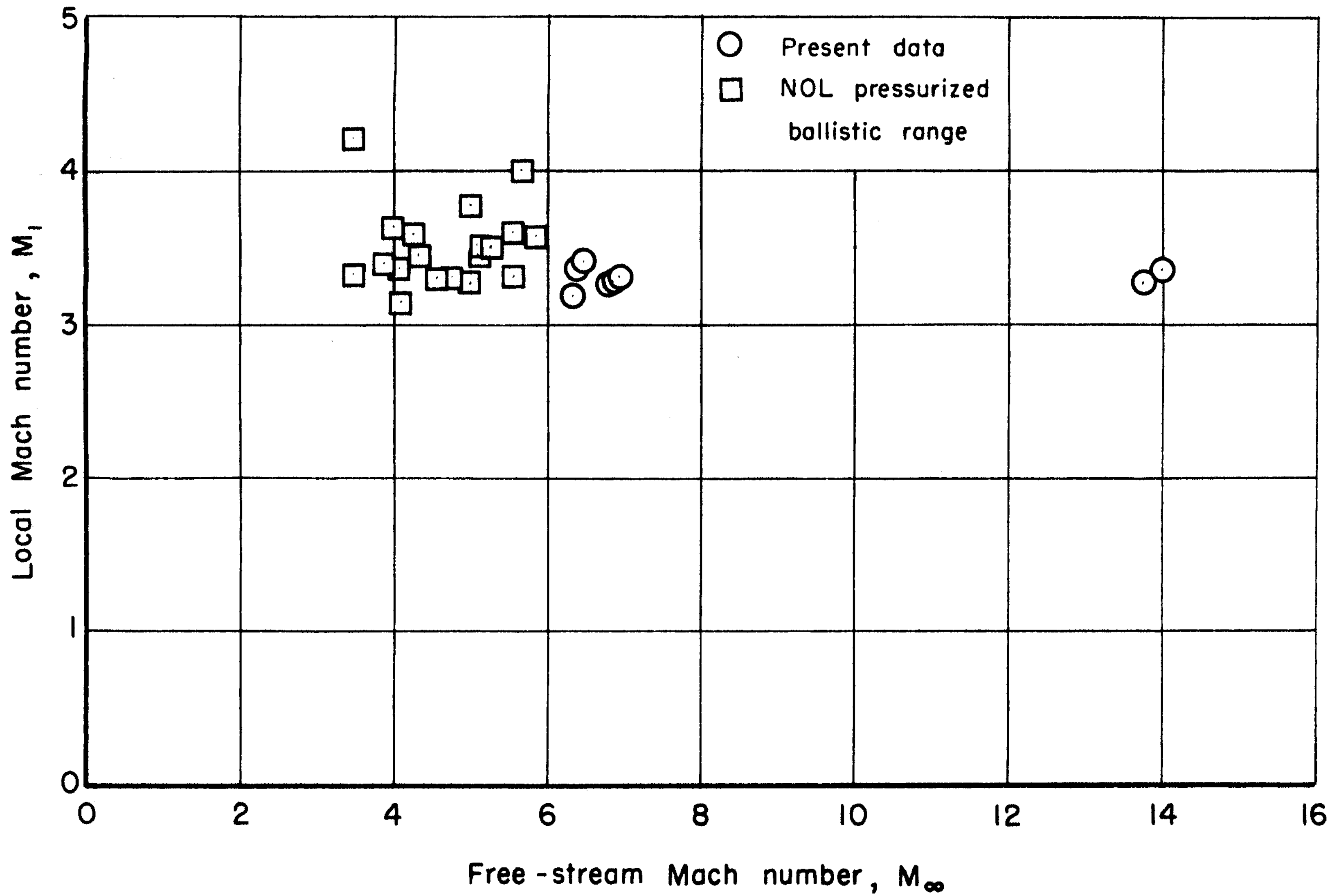

Figure 3.- Variation of local Mach number, M_l, with free-stream Mach number, M_∞, at $x/l = 0.51$.

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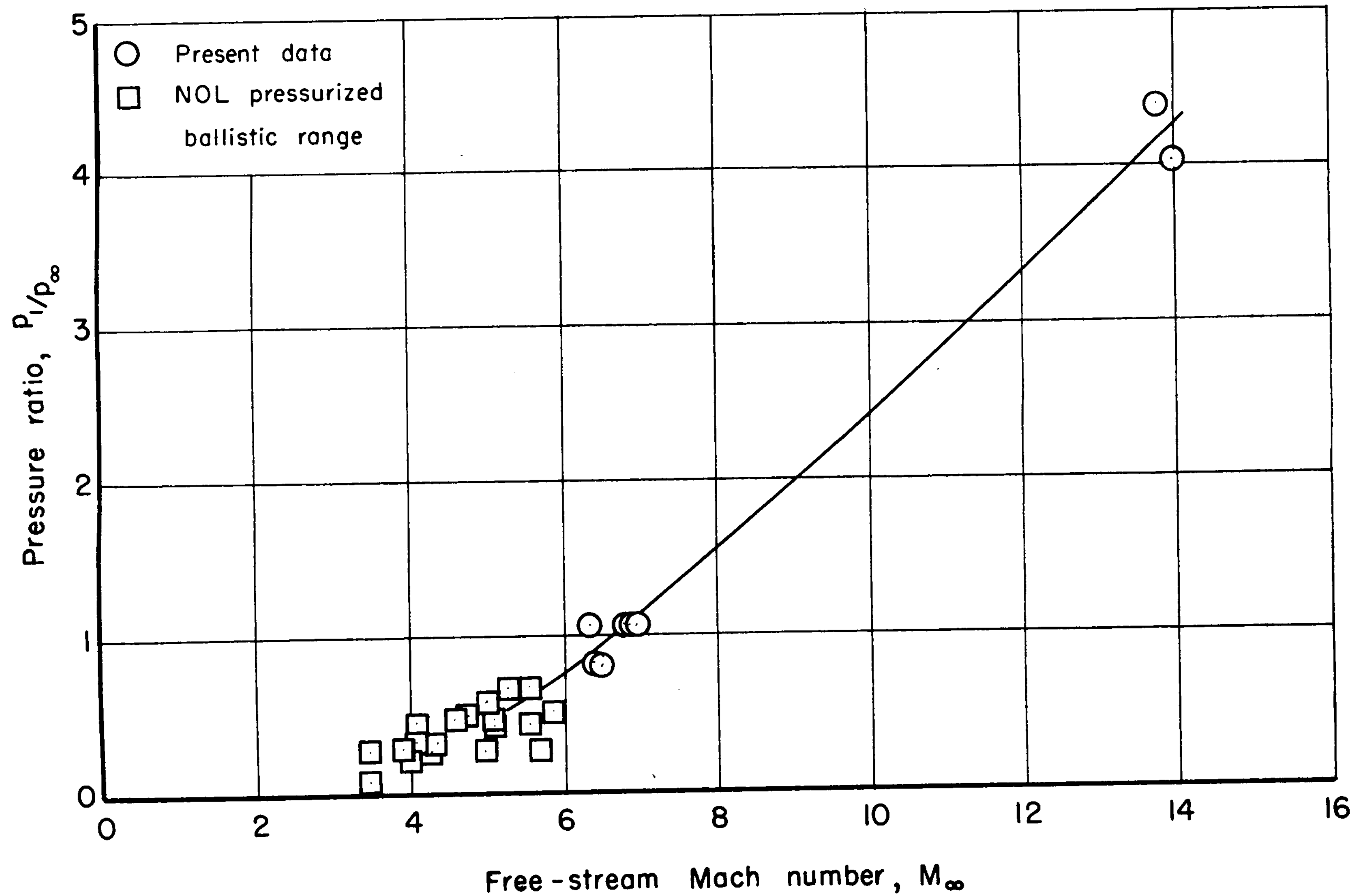

Figure 4.- Variation of local pressure ratio, with free-stream Mach number at $x/l = 0.51$.

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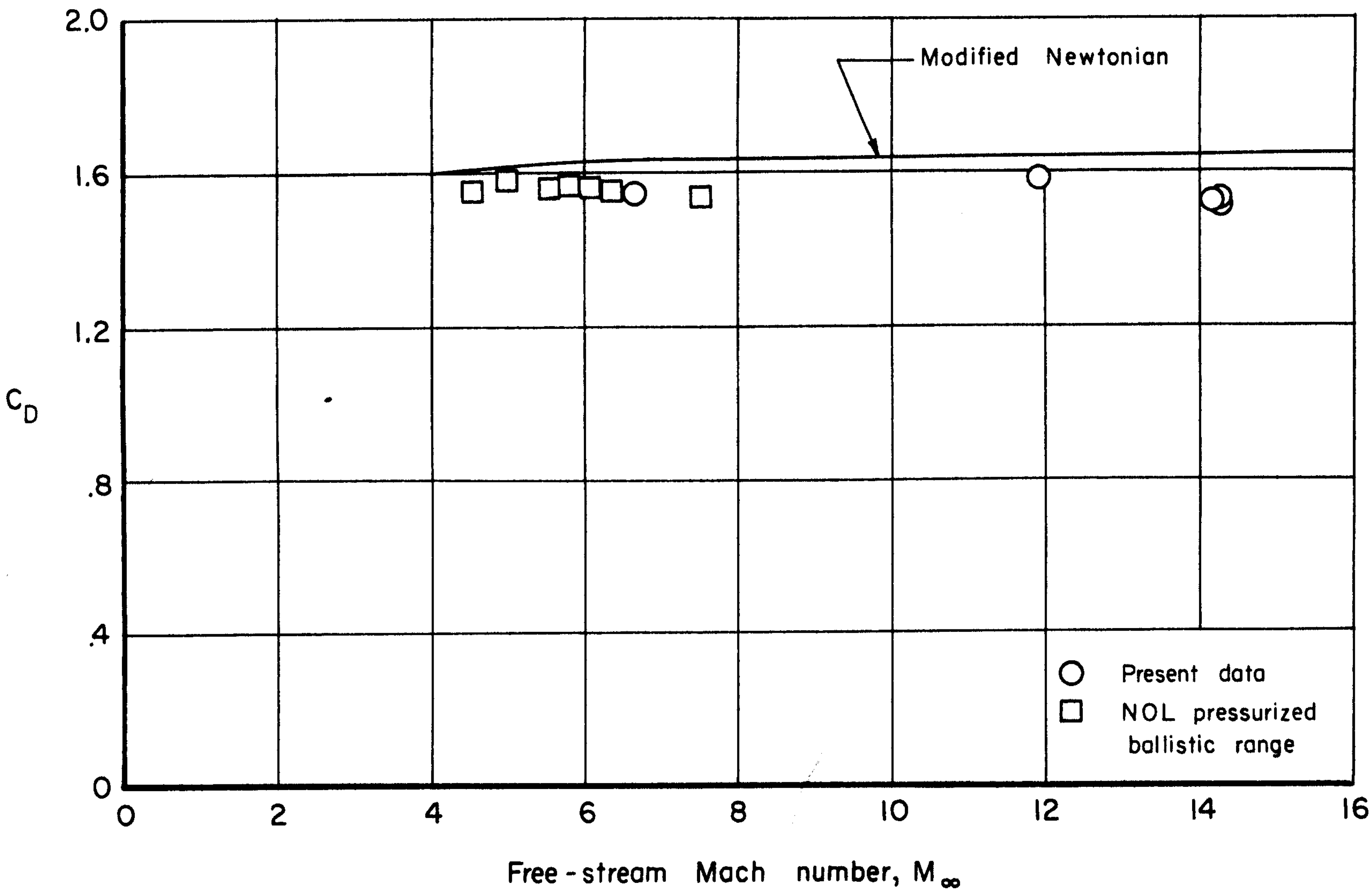

Figure 5.- Variation of total-drag coefficient with free-stream Mach number.

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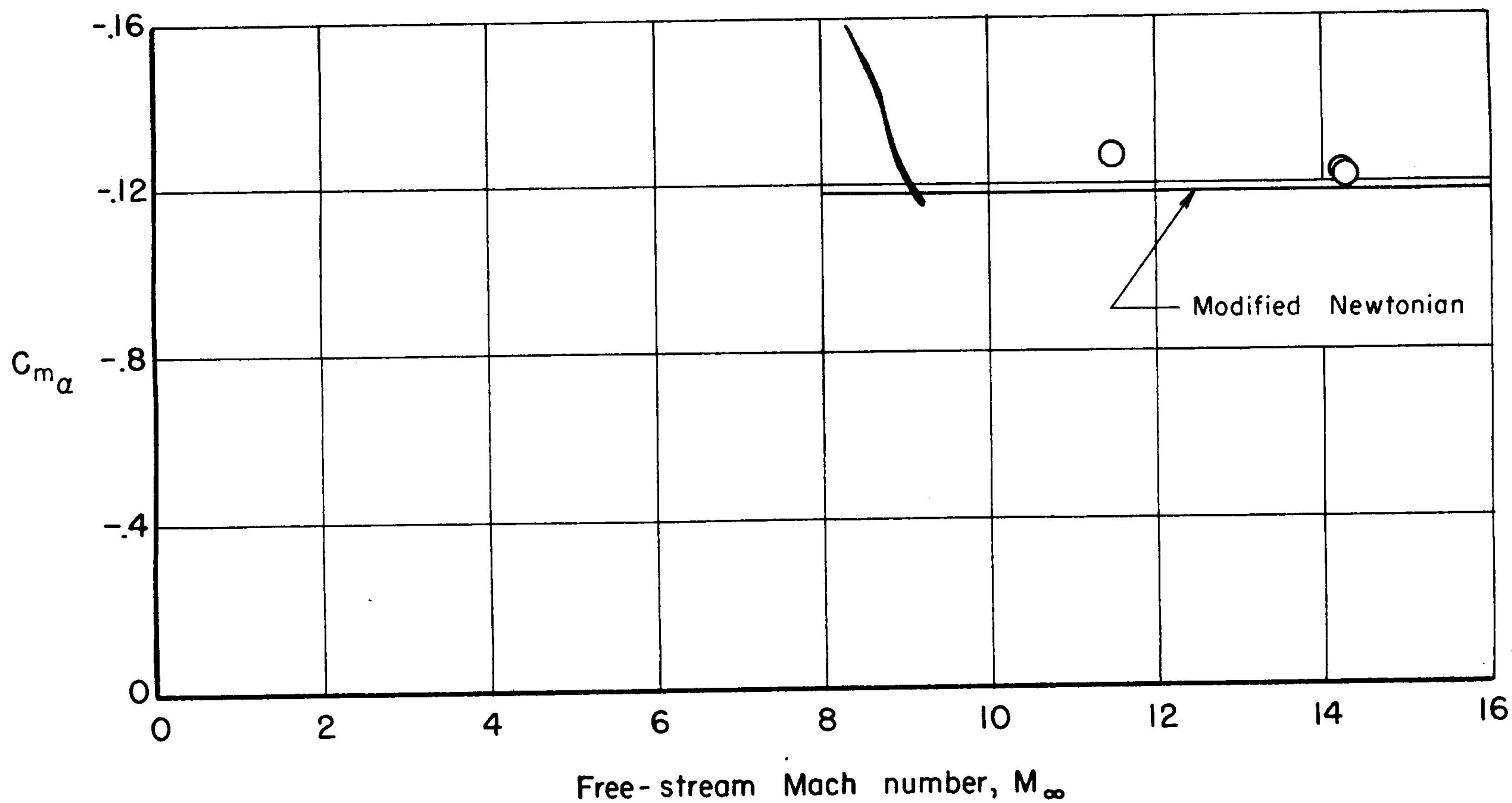

Figure 6.- Variation of pitching-moment-curve slope with free-stream Mach number.

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