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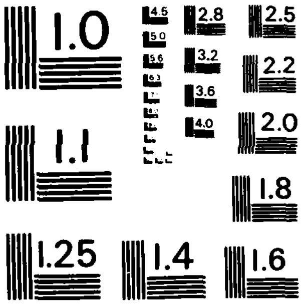

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SUPPLEMENT

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80-MICRON PARTICLE ACCELERATION IN AN EXPANDING GAS

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Aerodynamics & Airframe Branch
Aeromechanics Division

June 1983

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20. ABSTRACT (*Continue on reverse side if necessary and identify by block number*)

In ~~Technical~~ Report AFWAL-TR-81-3105 a new method had been introduced to calculate the flow with heat loss and friction in expansion nozzles. This method has now been improved, leading to somewhat more favorable results for the performance prediction of these nozzles. Derivations for the improved method and new calculation results are given in this Supplement.

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FOREWORD

This Supplement, prepared under Work Unit 24041059, Project 2404, in the Aerodynamics and Airframe Branch, Aerodynamics Division, Flight Dynamics Laboratory in May 1982, covers improvements in the method of calculation and some new results for the Technical Report AFWAL-TR-3105, published by this Laboratory in November 1981.

LIST OF SYMBOLS

A	cross sectional area
D	nozzle diameter
L	nozzle segment length (also nozzle length in the calculation results)
M	flow Mach number
p	static pressure
R	gas constant
t	segment exit to inlet area ratio
T	static temperature
v	gas velocity
c_f	pipe friction coefficient
γ	ratio of spec. heats
ρ	density
τ	fluid force parameter $\tau = (t+1)/(2t)$, see Ref. 1

INDICES:

o	refers to stagnation conditions
1	refers to nozzle segment inlet
2	refers to nozzle segment exit
g	refers to gas
p	refers to particle

I. INTRODUCTION

This Supplement to Technical Report AFWAL-TR-81-3105 (Ref. 1) provides some improvement to the method used therein for calculating the flow in an expansion nozzle under the influence of wall friction and heat losses. As mentioned in Reference 1 the method had been previously developed as a general tool for calculating adiabatic flows in ducts with friction. By using a segmental approach the method was also considered suitable for calculating non-adiabatic flows as they occurred in the expansion nozzle of Reference 1. In this case the nozzle was divided into many small segments. The flow in an individual segment was then treated as being adiabatic and only between segments the flow conditions were corrected for the occurring heat loss. This process has now been found to be a poor approximation of the real flow. New investigations revealed that, for a better approximation, the segments themselves must be treated as non-adiabatic, i.e. the total temperature change over the segment must be part of the input to calculate the flow in the segment. Since the total temperature change is a function of the unknown flow conditions in the segment, an iteration procedure appears to be necessary. However, calculations have shown that for small enough steps for which the heat transfer conditions do not change abruptly from segment to segment, it is sufficient to simply assign the total temperature change of the preceeding segment to the following one for calculating the non-adiabatic flow conditions. Heat loss and total temperature change in the segment are then recalculated before proceeding to the next segment.

The improved method has been checked for accuracy by comparing its results with those obtained for the same case with Shapiro's method of calculating flows in ducts described in Reference 2. For each method the steps in the calculation procedure were chosen small enough to practically eliminate the influence of the step size on the results. Under these conditions nearly perfect agreement between the two methods was obtained. The comparison of these two methods together with discussions on the practicability of each method has been made a subject of a separate report intended for publication as an AIAA Technical Note (Ref. 3).

The following sections contain the derivations for the improved calculation method and some new results for the various nozzle cases of Reference 1.

II. THE IMPROVED CALCULATION METHOD

Only Equation 14 in Reference 1 is affected by the new calculation method. To explain the change, the derivations given in Reference 1 are partly repeated here beginning with Equation 10, with the changes introduced which lead to an improved Equation 14. To better understand the changes, the derivations have been made more detailed.

$$\frac{\gamma M_1^2}{t} + \tau = \frac{p_2}{p_1}\left[\gamma M_2^2\left(\frac{c_f L}{2 D} + 1\right) + \tau\right] \qquad (1)$$

This equation, derived from the flow momentum conditions, contains two unknowns: the exit Mach number M_2 and the pressure ratio p_2/p_1 across the expanding flow. Conservation of mass and energy as expressed by the following two relations

$$v_1 A_1 \rho_1 = v_2 A_2 \rho_2 \qquad (2)$$

$$\frac{T_o}{T} = 1 + \frac{\gamma - 1}{2} M^2 \qquad (3)$$

allows one to write an additional condition for the two unknowns. The energy equation under the assumption of ideal gas conditions*), describes

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*) The ideal gas assumption is very realistic for helium used in the present calculations. For gases with state-dependent gas properties too, the ideal gas assumption constitutes no severe restriction. In the segmental approach the assumption applies only to a small segment of the expansion nozzle and the gas properties can be changed from segment to segment. For a rougher approximation constant average gas properties may be assumed for the whole expansion process as done for the case of air in Table 1.

the balance between heat energy and kinetic energy in a flowing gas in reference to the heat energy of the gas at rest. In the present flow problem the change in total temperature of the gas from inlet to exit of the segment is known from the inlet conditions and certain assumptions made about the heat loss from the segment. For finding the two unknowns only energy conservation at the segment exit, as expressed by Equation 3, enters the derivations as will be seen below.

Using the equation of state

$$\rho = \frac{p}{R\ T} \tag{4}$$

Equation 2 can be written

$$\frac{v_2\ A_2\ p_2}{R_2\ T_2} = \frac{v_1\ A_1\ p_1}{R_1\ T_1} \tag{5}$$

or

$$\frac{v_2\ A_2\ p_2\ \sqrt{\gamma_2}}{\sqrt{R_2\ T_2\ \gamma_2}\ \sqrt{R_2\ T_2}} = \frac{v_1\ A_1\ p_1\ \sqrt{\gamma_1}}{\sqrt{R_1\ T_1\ \gamma_1}\ \sqrt{R_1\ T_1}} \tag{6}$$

since $$M = \frac{v}{\sqrt{\gamma\ R\ T}} \tag{7}$$

and for a perfect gas $\gamma_1 = \gamma_2$ and $R_1 = R_2$

Equation 6 becomes

$$\frac{M_2\ A_2\ p_2}{\sqrt{T_2}} = \frac{M_1\ A_1\ p_1}{\sqrt{T_1}} \tag{8}$$

or

$$M_2 = M_1\ \frac{A_1}{A_2}\ \frac{p_1}{p_2}\ \sqrt{\frac{T_2}{T_1}} \tag{9}$$

The temperature ratio in Equation 9 can be expressed by

$$\frac{T_2}{T_1} = \frac{(T_1)_o}{T_1}\ \frac{T_2}{(T_2)_o}\ \frac{(T_2)_o}{(T_1)_o} \tag{10}$$

At station 1 as well as at station 2 the static temperature is related to the total temperature by the energy equation (Equation 3).

Let
$$\frac{A_2}{A_1} = t$$

then Equation 3, 9, and 10 combined give

$$M_2\left(1 + \frac{\gamma-1}{2} M_2^2\right)^{1/2} = \frac{M_1}{t} \frac{p_1}{p_2} \sqrt{\frac{(T_1)_o}{T_1} \quad \frac{(T_2)_o}{(T_1)_o}} \qquad (11)$$

This equation is the "improved equation 14". The change from the original equation consists of the addition of the ratio of the segment exit-to-inlet total temperature. This ratio must be derived from the heat transfer occurring on the segment walls. This heat transfer is a function of the still to be determined flow conditions in the segment. To avoid an iteration process in calculating heat transfer and flow conditions, it is sufficient to use the total temperature ratio of the preceeding segment to calculate the flow conditions in the new segment by means of Eq. 11. For the first segment it is sufficient to use unity for the total temperature ratio, i.e. adiabatic conditions as done in Ref. 1.

III. RESULTS

The subsonic nozzle flow remains practically unaffected by the improved method due to the small change in the total gas temperature occurring in this region. For easy reference, the old Figure 6 showing the subsonic conditions is included here as Figure 1. The notations on this figure also provide the unchanged input data for the calculations. The data list has been extended to indicate that the pipe friction coefficient is 0.02 for the subsonic and 0.015 for the supersonic nozzle portion.

Figures 7a - c of AFWAL-TR-81-3105 have been recalculated with the improved method. The results are shown here as Figures 2a - c. Some of the less essential curves have been omitted from the figures to improve their clarity.

To simplify the calculations, the new Figures 2b and 2c are calculated without the short 0.34° nozzle portion next to the throat, which was included in the old calculations (Figures 7b and 7c). This 0.34° portion is the result of a design consideration and not essential for the particle acceleration. The nozzle portion near the throat requires a special design for handling the enormous heat load in this region. For actual experiments, extensions with various wall angles would be attached to this specially designed throat portion.

The improved method affects mostly the supersonic Mach numbers, increasing, for instance, in the case of the 0.34° nozzle, the maximum Mach number from 1.4 to 2.05. It also strongly increased the drop in the static gas pressure along the nozzle. Since the maximum gas velocity was only increased from 3280 m/sec to 3730 m/sec, the large increase in Mach number must be associated with a substantial decrease in the local velocity of sound or under the present circumstances, with a substantial decrease in the static gas temperatures. Since the heat losses were reduced, i.e. the drop in total gas temperature was decreased, the lowering of the static temperatures is essentially the consequence of the faster static pressure drop indicated above. Since, as also indicated, the gas velocities were not greatly changed, the particle velocities were even less affected, changing in the present example from 1950 m/sec to 2100 m/sec for the maximum value obtained.

In the original calculations, a nozzle wall angle of 0.5° was found to be optimum for the acceleration of the particles. The new calculations shift the optimum to smaller angles. As an example, the performance for a 0.25° wall angle is shown in the new Figure 2d. The performance is also listed in Table I as No. 3a. For $p/p_0 = 0.02$, this nozzle gives the highest particle velocity for all nozzles with a throat diameter of 3.97mm. The difference in particle velocities between these four nozzles is not very pronounced. Considerations for the design and fabrication of nozzles with such small wall angles may dictate the actual choice. Also, the nozzle length may be a decisive factor. With a small sacrifice in particle velocity, fairly short nozzles with a large wall angle give good results. For instance, case No. 2 in Table 1 assumes a wall angle of 0.5° and results in a nozzle length of only 344 mm to achieve a particle velocity of 2014 m/sec.

A note for Table II of AFWAL-TR-81-3105: the indication of a minimum wall angle in case No. 9 was the result of misinterpreted computer results. The given value is incorrect. No attempt has been made to find the correct value. In concurrence with the above discussion on optimum wall angles, one can only conclude that for the present wall friction assumptions the minimum wall angles for which the flow becomes choked are in all cases extremely small with case No. 9 being no exception.

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1. S. H. Hasinger, "80-Micron Particle Acceleration in an Expanding Gas", Flight Dynamics Laboratory, Air Force Wright Aeronautical Laboratories, Technical Report AFWAL-TR-81-3105, Wright-Patterson AFB, Ohio, November 1981.

2. A. H. Shapiro, "The Dynamics and Thermodynamics of Compressible Fluid Flow", Volume I, The Ronald Press Company, New York, 1953, p. 219.

3. S. H. Hasinger, "A New Method for Calculating Ducted Flows", submitted to the AIAA for publication as Technical Note.

TABLE I

CALCULATION RESULTS

No.	Throat Diam. mm	Nozzle Wall Angle subs.	Nozzle Wall Angle supers.	$p/p_o = 0.02$ L mm	$p/p_o = 0.02$ v_g m/sec	$p/p_o = 0.02$ v_p m/sec	$p/p_o = 0.01$ L mm	$p/p_o = 0.01$ v_g m/sec	$p/p_o = 0.01$ v_p m/sec	$p/p_o = 0.005$ L mm	$p/p_o = 0.005$ v_g m/sec	$p/p_o = 0.005$ v_p m/sec
Helium ($\gamma = 1.666$) $T_o = 3200°K$ $p_o = 100$ atm												
1	3.97	7^o	0.34^o	555	3216	2104	870	2923	2134	1327	2640	2143
2	3.97	7^o	0.5^o	344	3786	2014	531	3588	2080	800	3362	2113
3	3.97	7^o	0.7^o	232	4162	1877	350	4048	1968	519	3895	2022
3a	3.97	7^o	0.25^o	821	2657	2112	1305	2318	2118	2005	2032	(2118)
4	5.85	10^o	0.5^o	497	3807	2263	776	3605	2329	1170	3380	2359
5	5.85	10^o	0.7^o	333	4174	2133	507	4059	2227	759	3903	2282
6	8.34	7^o	0.7^o	487	4162	2345	736	4047	2440	1090	3890	2494
7	8.37	15^o	1.0^o	308	4477	2191	463	4441	2310	681	4358	2387
Air (heated, $\gamma = 1.32$) $T_o = 3200°K$ $p_o = 100$ atm												
8	3.90	7^o	0.7^o	281	1695	1203	418	1672	1236	604	1633	1255
8a	3.90	7^o	0.34^o	612	1305	1179	934	1208	1181	1393	1110	(1181)
Air (heated, $\gamma = 1.32$) $T_o = 2000°K$ $p_o = 200$ atm												
9	20.59	20^o	5^o	212	1650	1237	293	1715	1299	401	1765	1345
9a	20.58	15^o	1^o	1005	1456	1320	1495	1467	1349	2178	1462	1364

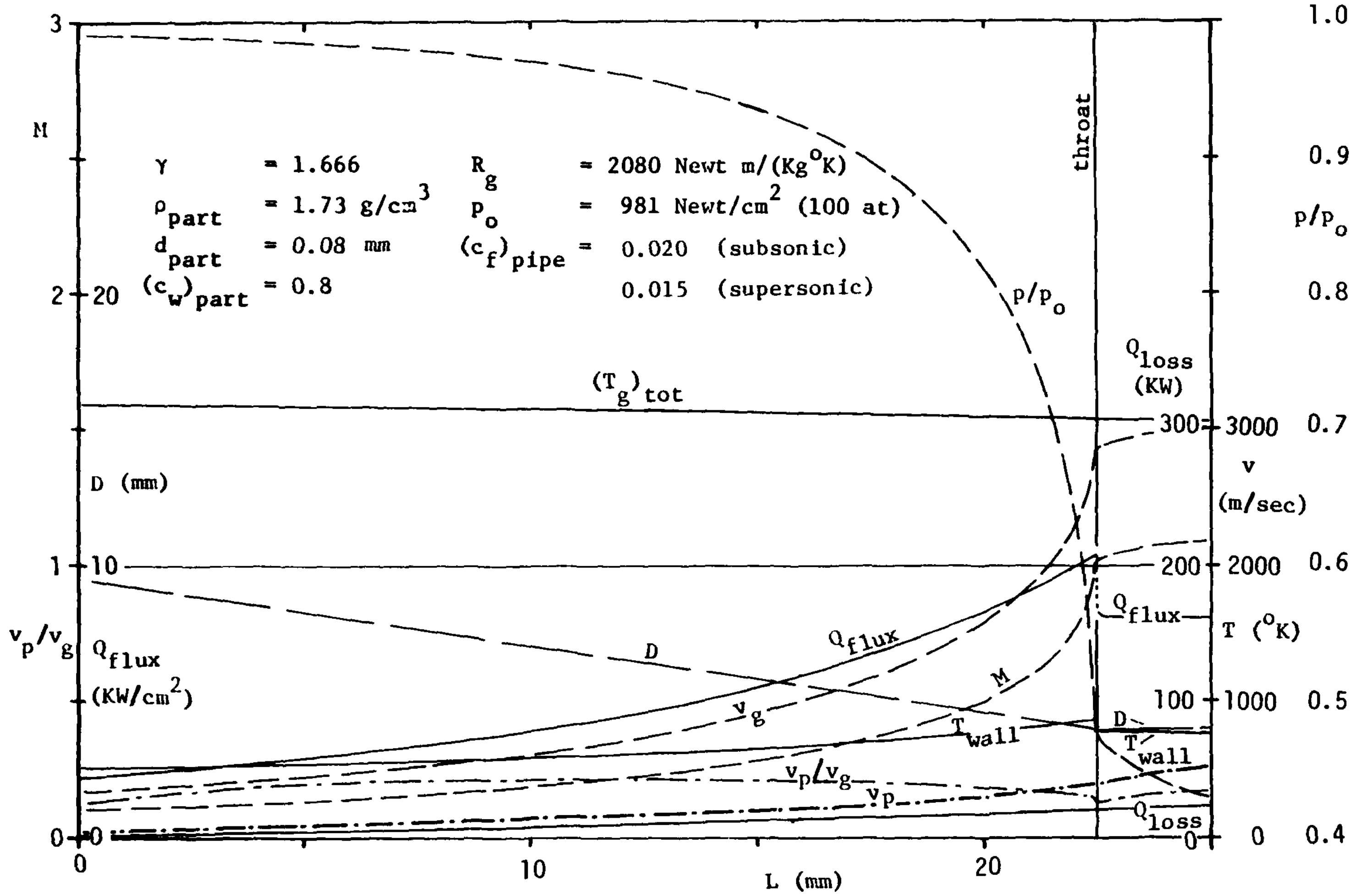
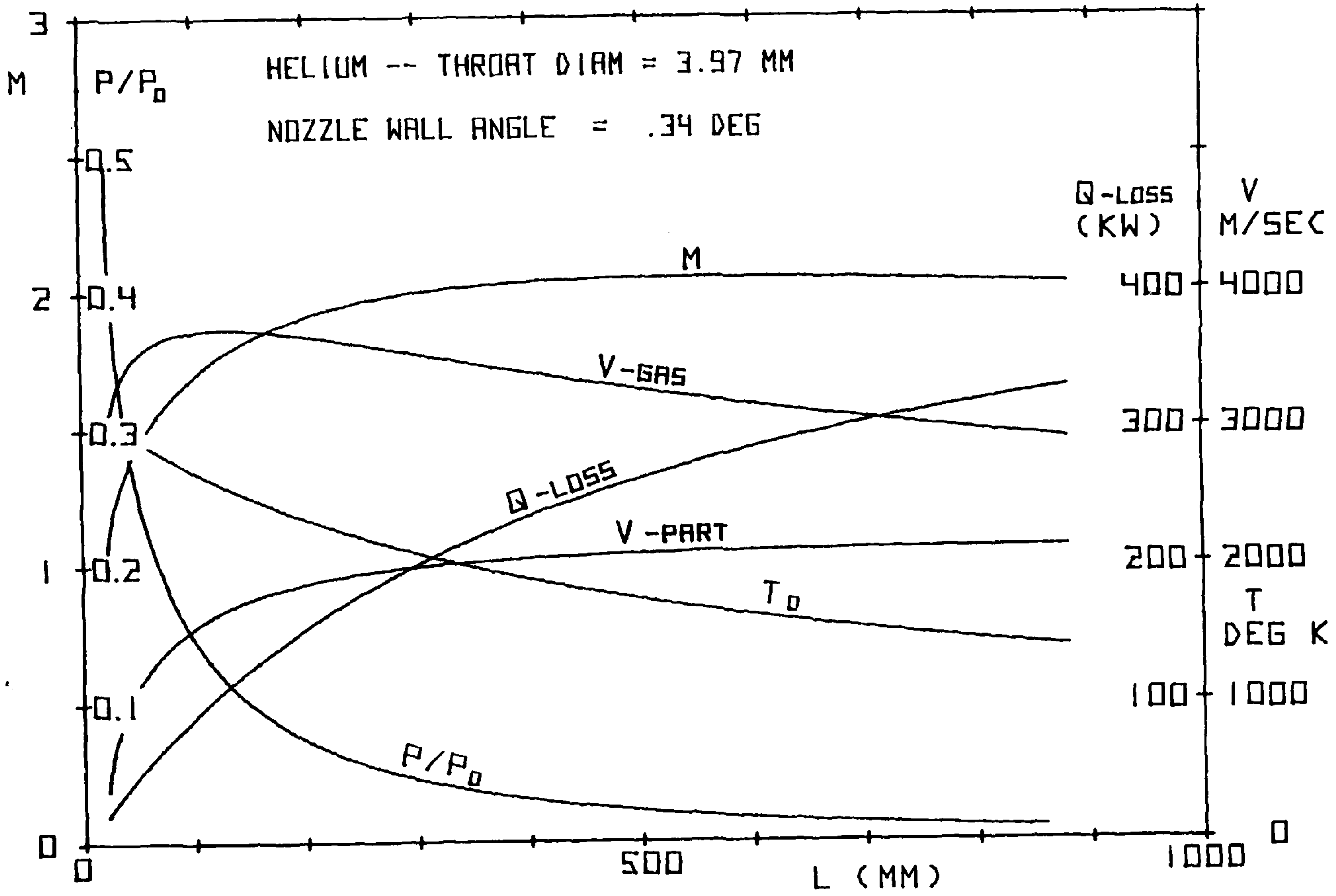
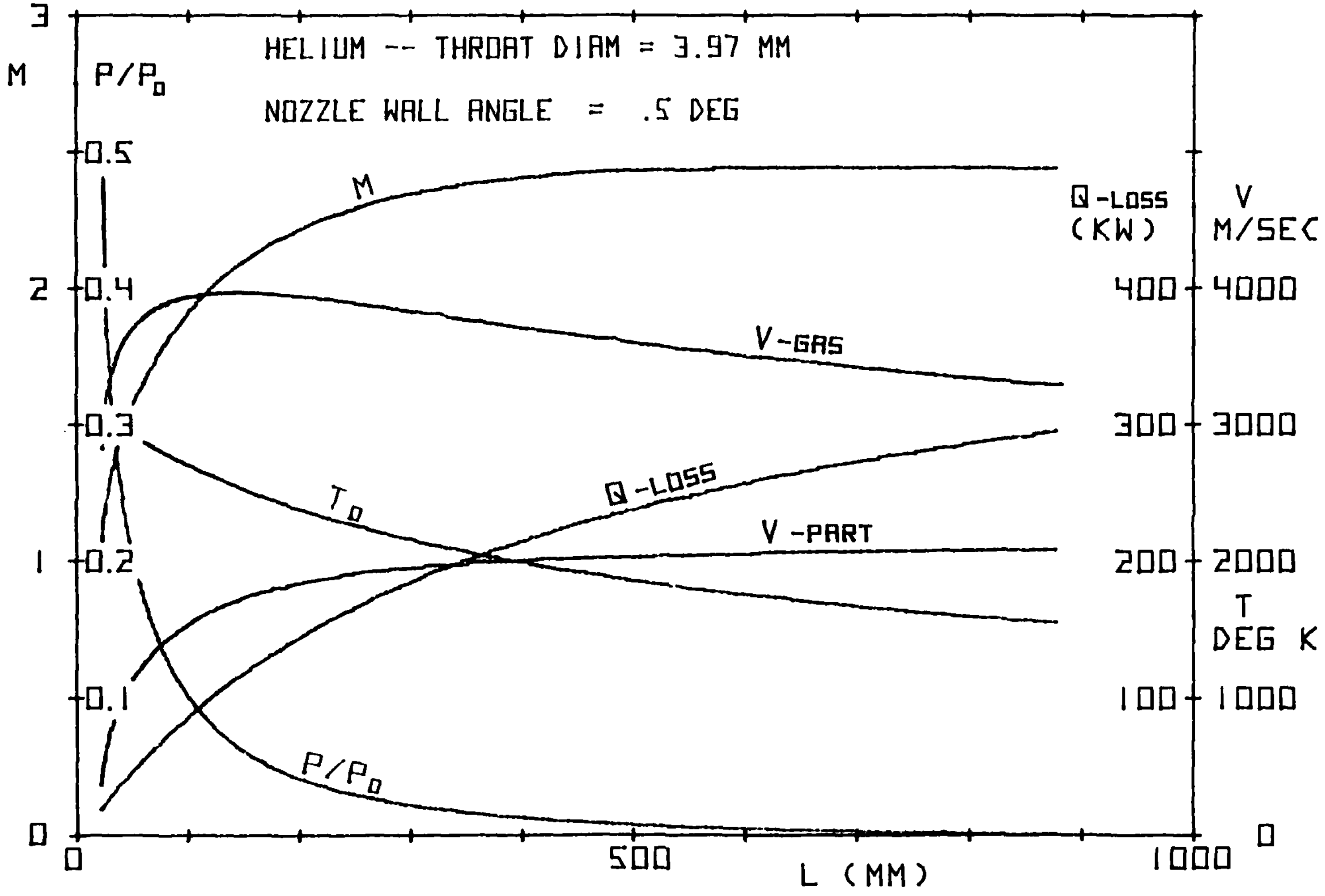


Figure 1. Flow Conditions, Heat Loss, and Particle Acceleration for the Subsonic Portion of the Expansion Nozzles Considered in Figures 2a to d for Their Supersonic Nozzle Portions



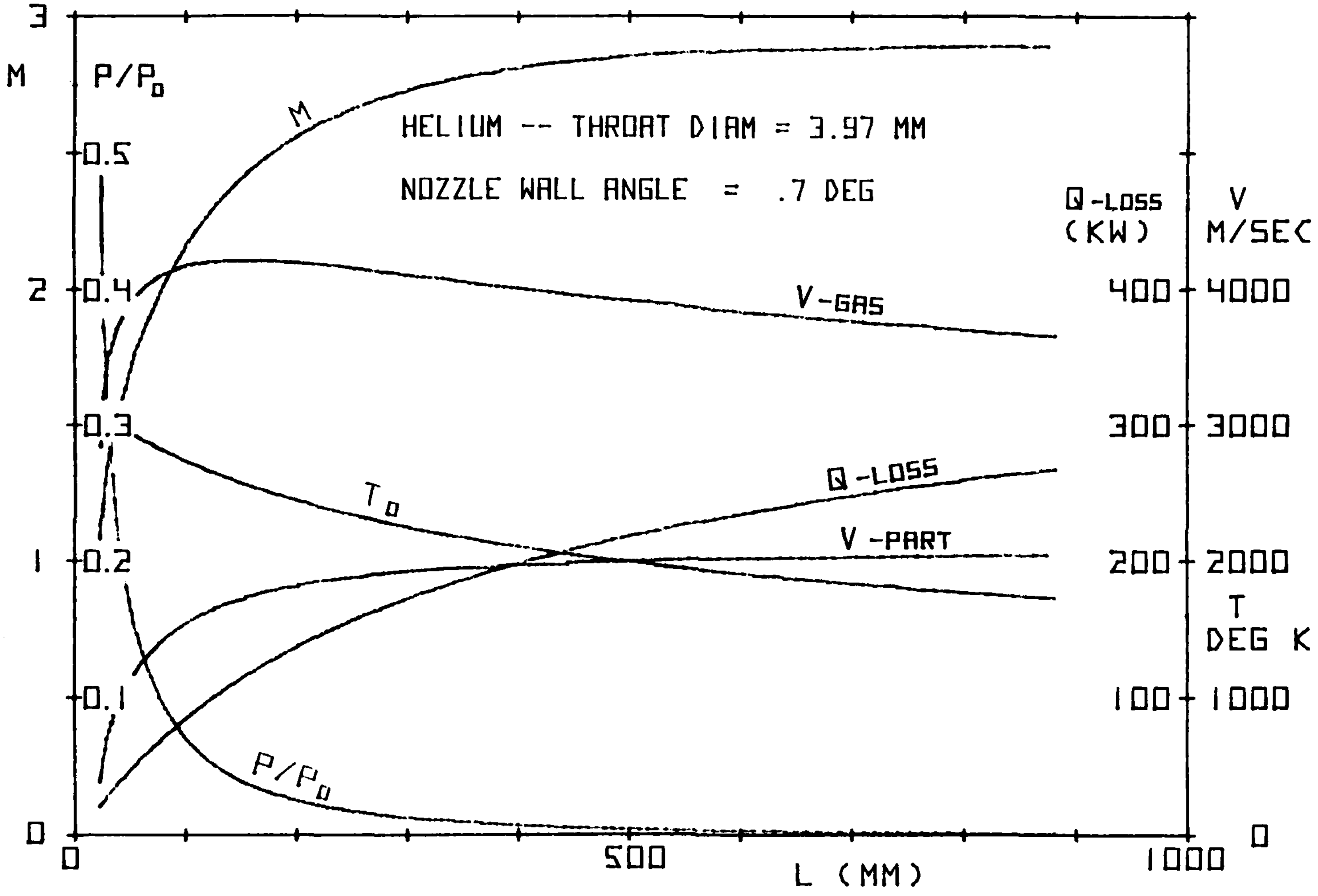
a. Nozzle Wall Angle 0.34°

Figures 2a to 2d. Flow Mach number M, Gas Velocity V-Gas, Particle Velocity V-Part, Total Gas Temperature T_o, Expansion Pressure Ratio P/P_o and Accumulated Heat Loss from the Nozzle Q-Loss for the Supersonic Nozzle Portion at Different Wall Angles (friction coefficient $c_f = 0.015$).



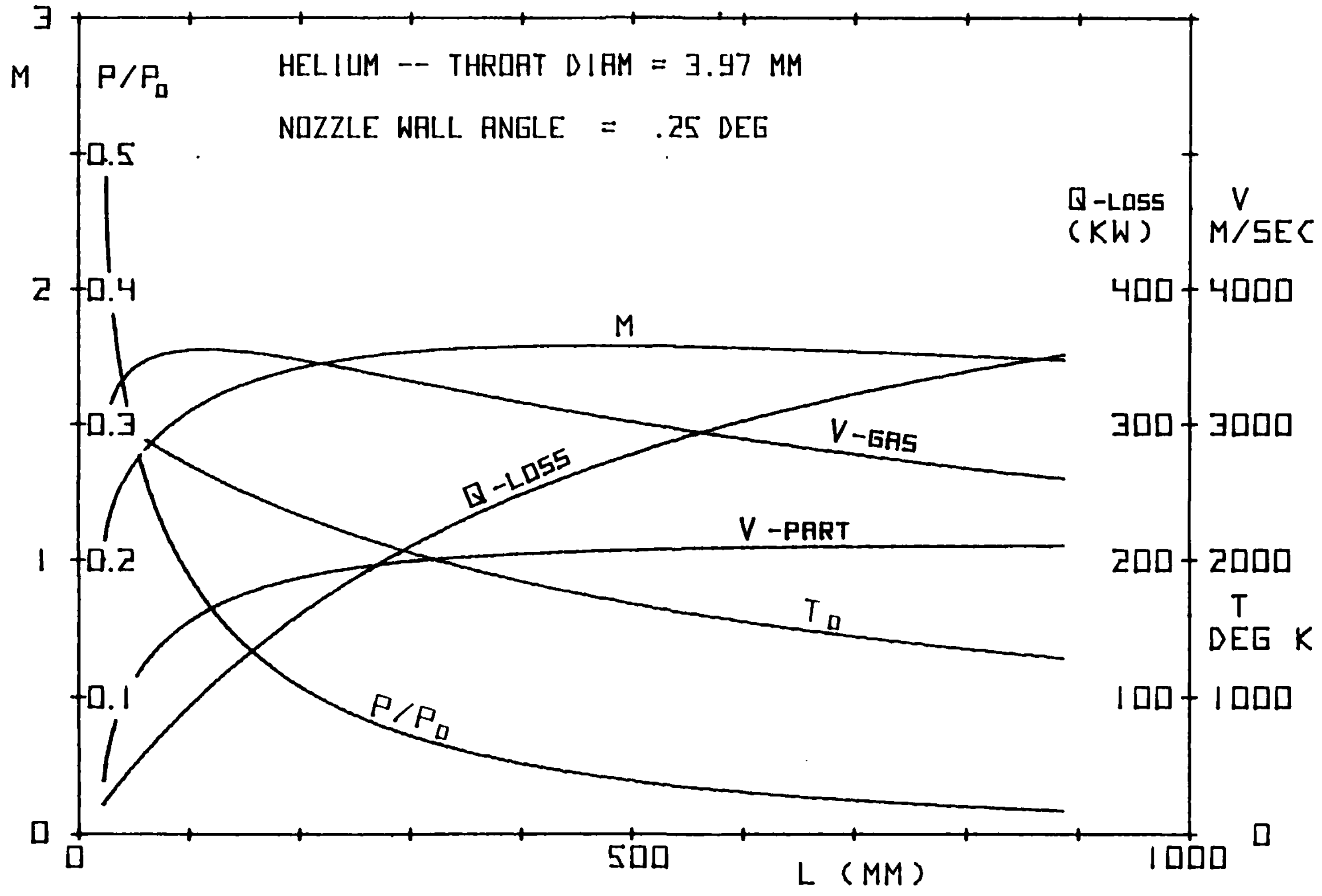
b. Nozzle Wall Angle 0.5°

Figures 2a to 2d.



c. Nozzle Wall Angle 0.7°

Figures 2a to 2d.



d. Nozzle Wall Angle 0.25°

Figures 2a to 2d.

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