MILITARY TECHNICAL COLLEGE

CAIRO - EGYPT

1 # AN ANALYSIS OF THE
2 # LAMINARIZATION PHENOMENA IN ROCKET NOZZLES

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4 ABSTRACT

The accelerations imposed on the gas flow inside nozzles of rocket engines are extremely high. By such high acceleration applied to the flow, the turbulent layer structure at the wall is suppressed and a partial or complete conversion to a laminar boundary layer is obtained. By such laminarization, intensive reduction in heat transfer rates occurs. In this paper;theoretical and experimental investigations were conducted to determine the impact of such laminarization phenomena on the local values of heat transfer coefficients in rocket nozzles.

A simple hypothsis is also given for correlating the measured gas-film heat transfer coefficients at the throat of such nozzles.

By such hypothesis, it was possible to develop a theoretical seheme based on the conservation equation to predict the temperature distribution, in the nozzle-walls.

1. INTRODUCTION :

The work presented here is mainly concerned with a study, analytical and experimental,

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originated to find the main parameters that influence the structure of the boundary layer flow along a convergent-divergent nozzle of a rocket motor.

It is known that if a sufficiently strong accelerating force is applied to a flow, the turbulent boundary layer structure at the wall can be suppressed and a partial or complete conversion to a laminar boundary layer is obtained[1]. Such laminarization influences directly the heat transfer rates along rocket nozzles since the flow along such nozzles is characterized by a strong accelerating force that is mainly dependent on the value of stagnation pressures. So the presented study covers a wide range of stagnation pressures from 10 bars - 120 bar. Launder [2] has stated that the boundary layer shape will suffer destortions as the accelerating parameter :

$$K = \frac{\mu}{\rho u^2} \frac{du}{dx}$$

has an approximate value of 10^{-6}. This value is mostly attainable in rocket nozzles near the throat[3]. Calculation of the heat transfer coefficients for flow in nozzles was done by the use of the simplified Bartz or colburn equations [4], of the form

$$St = 0.0296 \, Re_d^{-0.2} \, Pr^{-0.67} \qquad (1)$$

This equation provides an order of estimate of the local convective heat transfer coefficients in small nozzles where an essentially turbulent layer flow is assumed. But such equation cannot be used for rocket nozzles where the used stagnation pressures are extremely high.

In the first part of this paper an experimental analysis of the laminarization phenomena is performed where the accleration parameter reaches the value $2.5.10^{-6}$ to $5.4.10^{-6}$ at the throat section of rocket nozzles. Measurements is concerned with the heat transfer rates along the nozzles which is directly related to the boundary layer structure.

In the second part of this paper, analysis of the found experimental data is performed and correlation of the found data is done relaying upon a concept of the shape of distorsion of the boundary layer along the nozzle.

2.EXPERIMENTAL ANALYSIS.

A need for a high pressure gas generator is satisfied by the use of a combustion chamber of a rocket motor that utilizes a douple base solid propellant of tubular grains. Such chambers are characterized by short duration time, 10-20 second, so the measurement instrumentations have small inertia time to follow up the transient measurement data [3].

The gases produced from the gas generator were passed through a long pipe before entering the nozzle to have a partially developed gas flow. The length of the pipe was chosen in the range :

$$L = (3\text{-}5)\ d$$

Where d : diameter of the pipe

L : total length of the pipe

This leads to gas parameters at the inlet section of the nozzle of the values :

Pressure : 20-120 bar

temperature : 1200-1800 °K

The nozzle geometrical cofiguration was considered as another parameter that influences the structure of boundary layer flow along the nozzle. So, the convergence angle of the nozzle is considered as a variable that ranges from 30-60 degrees.

The gas generator was placed on a horizontal test stand. Inlet and throat pressures were measured by pressure transducer that were connected to an analogue digital converter of a Packard computer that was programmed to work as a data acquistion system. Wall temperatures at 20 locations along the nozzle were measured by means of NiCr/Ni thermocouples and the induced e.m.f were amplified and fed to the computer A/D converter. Other thermocouple sets were fixed at 20cm distance from the exhaust-nozzle section for determination of the exhaust gas temperature.

Calibration of the measuring devices were done before each measurement round. Each round was repeated three times to have the confident measurement data.The thermocouples were located at five sections along the nozzles, fig 1., where at each section were fixed four thermocouples with different depths through the nozzle walls. From the four measured temperatures at each section, it was possible to estimate the heat transfer rates across the wall and the gas temperature at each section[4], through the experimentation time.

The number of rounds that have been done, to get a confident dependence of the heat transfer rates across the nozzles on the gas flow parameters and nozzle geometry, were 120 rounds (corresponding to 40 different values of the stagnation pressures and nozzle-convergence angles)

3. RESULTS ANALYSIS

From the experimentally determined heat transfer rates and gas temperatures along the nozzle walls, the dimensionless heat transfer coefficients, Stanton Number, have been calculated. Comparing the experimental values of heat transfer coefficients to that expected from equation (1), it can be observed, fig 2., that the heat transfer falls progressively below that for simple turbulent flow as long as the acceleration parameter is above a certain value, here marked in fig 2. as 2.4×10^{-6}, No sudden changes in Stanton Number occurs, but the rate of development becomes more appropriate to value of that defined by Pohlahausen[5] for essentially a laminar boundary layer flow :

$$St = 0,332 \; R_{e_d}^{-0,5} \; Pr^{-0,67} \qquad (2)$$

During the performed experimental analysis it was possible to provide different values for the acceleration parameter K in the range of

$$2.5 \times 10^{-6} - 5.4 \times 10^{-6}$$

The corresponding values of Stanton numbers were measured and compared to that predicted from equations(1) and(2). It was found that the boundary layer flow can be considered essentially turbulent for K below a value of 1.26×10^{-6}, and the measured values of heat transfer rates were close to that predicted from Bartz equation(1). For the acceleration parameter K above the value 3.4×10^{-6}, the boundary layer flow can be considered to become essentially laminar and the measured heat transfer rates were close to that predicted from Pohlehausen equation(2). Measurements of the heat transfer rates corresponding

± 20% in such type of work is acceptable according to the references in such field of work[1].

In addition to the process of laminarization reducing the heat transfer to the nozzle walls, a sharp drop of the heat transfer has been found when the flow becomes supersonic. Evidence from the measurement results that this change in heat transfer rates occurs just upstream of the geometric throat, at a point at which Mach number close to the wall will Just be unity, fig.3. The heat transfer rates at such point is close to the values predicted from equation (2).

4. THEORETICAL ANALYSIS

In this part, relaying upon the mathematical modeling of the whole processes occuring in rockets, the temperature distribution in the walls of the nozzle can be found by making use of the imperical equations obtained from the previous section.

Firstly, to find the gas temperature at the inlet section of the nozzle a simple idealization of the steady state deflagration of the tubular double base propellant[8] is followed. We consider that the propellant burns with a rate "S" whose pressure dependence is represented imperically by the equation :

$$S=0.32 + 0,023\ P \quad \text{cm/second},$$

where P is the chamber pressure in bars, as determined from the conservation equations for the steady state combustion and that includes those for mass, chemical species, and energy [8].

Then by an axisymmetric simulation of flow inside the nozzles. a (2-D) elliptic equations are found that considers pressure-velocity (p-u-v) formulations. By to values of K in the range $1.26 \times 10^{-6} \leq K \leq 3.4 \times 10^{-6}$ leads to the existence of a progressive change in the boundary layer shape throughout these values. As there no discontinuity should be in the Stanton number curve during the laminarization process; so a transition equation may be found. By introducing Krojilini integral equations [5] and the boundary layer shape factor [6], it was possible to correlate the measurement results by an imperical equation of the form :

$$St = A(K)\, R_{e_d}^{-0,4} \quad Pr^{-0,67} \qquad (3)$$

$$\text{for} \qquad 1.26 \times 10^{-6} \leq K \leq 3.4 \times 10^{-6}$$

Where the term A(K) is a function of the acceleration parameter K, and it is mainly dependent on the rate of distorsion of the boundary layer shape. The functional dependence of such term was found from statistical analysis of the measurement data to be of the form :

$$A(K) = \frac{0.102}{1+10^{10} K^2} \qquad (4)$$

Fig(2) shows the scattering of the measurement data with respect to the graphical repsentation of equations (1) and (2).

However, it must be beared in mind that this equation represents a step change in Stanton number during laminarization process. So, as has been found from measurements, when the acceleration parameters fall again, there is a contunious transition back to a turbulent boundary layer. During such transition, the same equations (3) and (4) fit correctly the measurement data (However, a difference of the use of the Imperial College "TEACH" (Teaching Elliptic Axisymmeric Characteristies Heuristically) computer program[9], a solution by Gaus-Siedel line by line iterative procedure, the gas flow parameters along the nozzle are obtained. The solution scheme depends on writing the conservation equations for continuity, momentum, and energy in two dimensional finite difference form and solving them simultaneously to find the whole aspects of the flow. The whole solution-procedure is dealt with in previous work[9].

The boundary conditions at the nozzle walls can be described by the following equation[10]:

$$-\frac{d}{dx}\left(r \rho_e u_e \emptyset (H - H_w)\right) = r\, St\, \rho_e u_e (H_{aw} - H_w) \qquad (5)$$

where H_w is the enthalpy of the flowing gases at each section, $\emptyset$ is the energy-boundary layer thickness, and the suffixes e,w, and aw denote the conditions at the free stream edge of boundary layer, wall conditions, and adiabatic wall conditions [10].
The results of solution by TEACH program, and the value of Stanton number as found from equations (1),(2) or (3) are introduced into equation(5) to find the nozzles-wall temperatures.

Such procedure was applied at three cases concerning nozzles operating at stagnation pressures of 40,80,and 120 bar.Fig(4) shows the results for a nozzle operating at stagnation pressure of 80 bars. The obtained results were found to be in agreement with the measured temperatured in the convergent part of the nozzle but it overestimates the actual temperatures near the exhaust section. However, such difference is mainly due to flow seperation at such zone and this phenemonon should be dealt with seperately, in the future works.

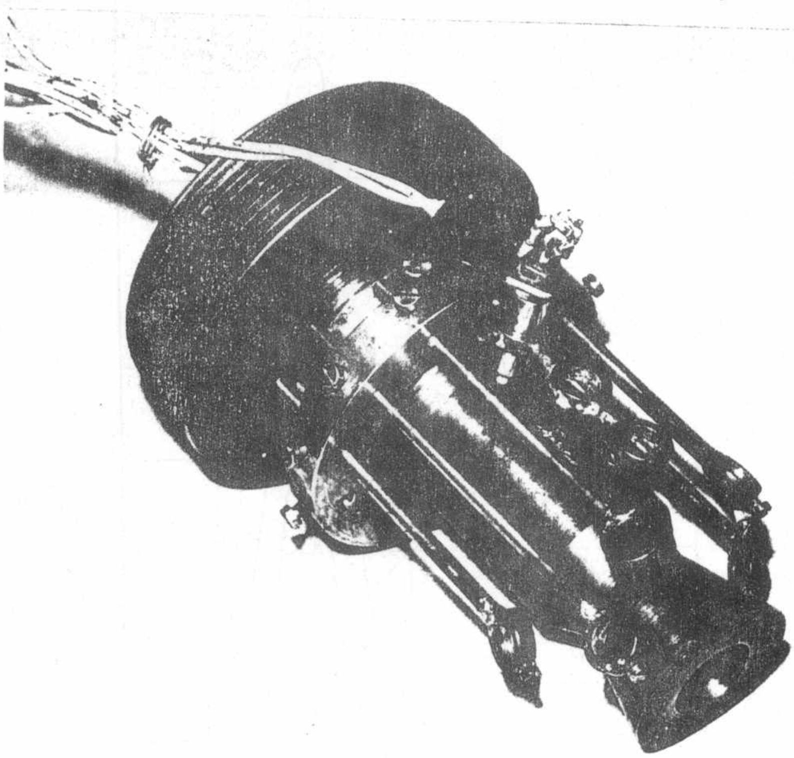

Fig. 1. THERMOCOUPLES FIXATION ALONG THE NOZZLE

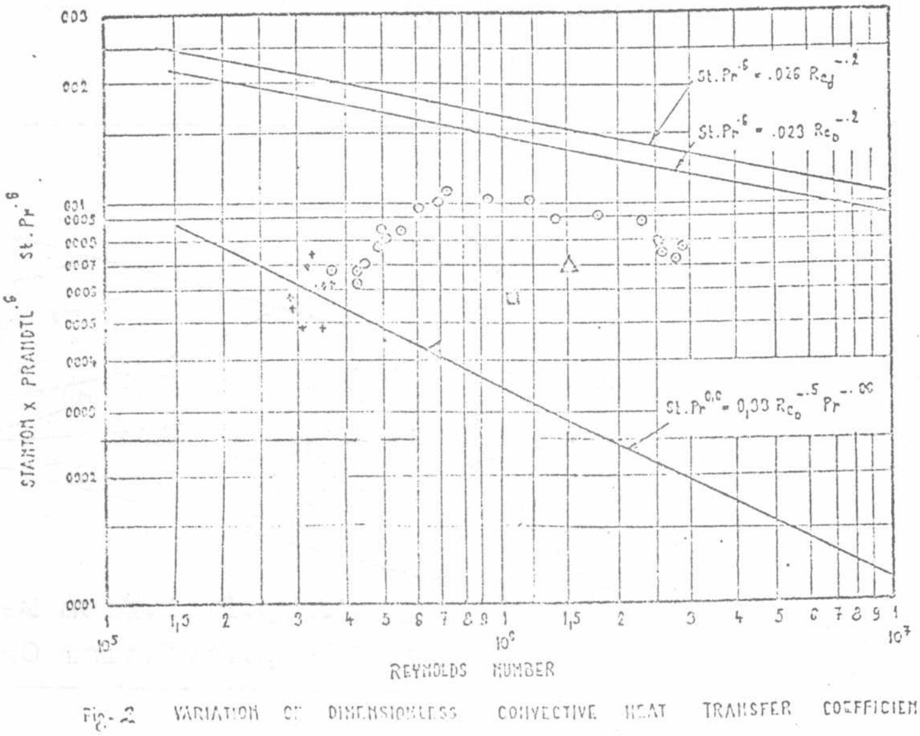

Fig. 2 VARIATION OF DIMENSIONLESS CONVECTIVE HEAT TRANSFER COEFFICIENT

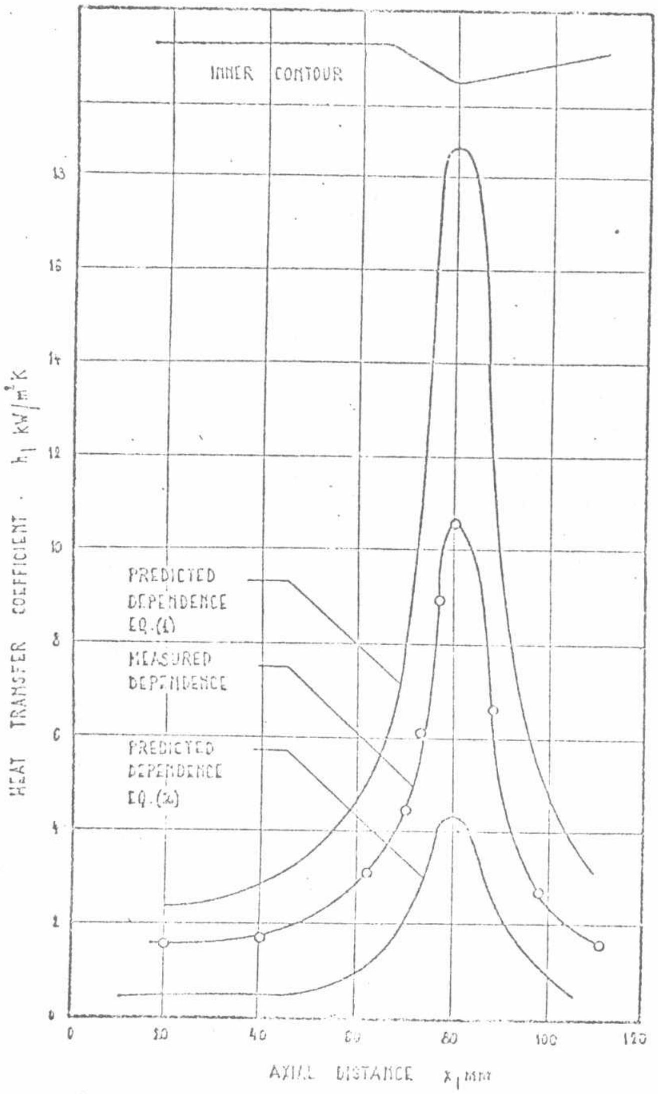

Fig. 3 PREDICTED AND MEASURED HEAT TRANSFER COEFFICIENT ALONG THE NOZZLE.

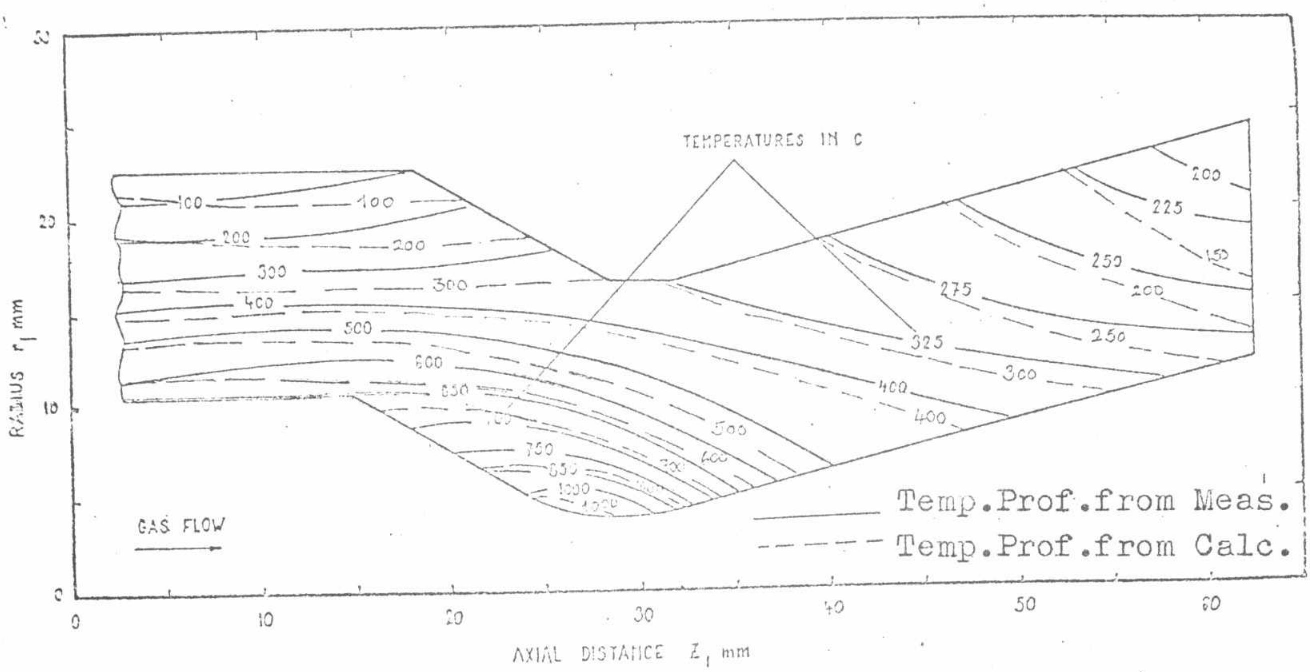

Fig. 4. TEMPERATURE PROFILE IN NOZZLE WALLS.

5. CONCLUSION

By a study of the main parameters that influence the structure of the boundary layer flow along a convergent-divergent nozzle of a rocket engine, it was possible to correlate the heat transfer rates along such nozzle. This was done by introducing an accelerating parameter K in the imperical equation utilized for prediction of the heat transfer coefficients. Such parameter has been found to be the source of suppressing the turbulent boundary layer and reduction of the heat transfer rates.

Hence, a semi-imperical scheme of solution that finds the temperature distribution in the rocket walls was proposed and the found results are acceptable.

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6. REFERENCES :

1. L.H Back, P.E. Massier, and R.E. Guffel : " Flow phenomena and Convective Heat Transfer In Conical Supersonic Nozzles ", Journal of Spacecrafts and Rockets, 4,8,1047, 1977.

2. B.E. Launder, F.C. Lockwood,"An Aspect of heat Transfer in Accelerating Boundary," Journal of Heat Transfer, 91,2,299-234,1969.

3. A.M.Salama, " Investigation of Heat Transfer Rates in Rocket Nozzles Operating at High Stagnation Pressures, " Proceeding of the First Conference in Mechanical Engineering, Cairo University, May 1977.

4. G.Liebmann; " Solution of Transient Heat Transfer Problems by Analogue Methods ", Tran. of ASME , 87,1267-1282, 1965.

5. M.I.Rashed, Two Dimensional Boundary Layer ", Cairo University, Cairo, 1963.

6. L.Shoenmann; P.Block; " Laminar Boundary Layer Heat Transfer in low thrust Rocket Nozzles, Journal of Space-Crafts and Rockets, 8,9,1089-1968.

7. A.D Gosman, W.H pun ", Heat and Mass transfer in two dimensional flow", Accademic Press, London, 1969.

8. M.S.Miller, " In Search of an Idealized Model of Homogenous Solid Propellant Combustion ", Combustion and Flame, 46,51+73,1982.

9. A.M.Salama, " A Modified Mathematical Modeling in Gas Turbines-A 2-D Scheme ", Conf.of Applied Mech,MTC,1984.

10. E.R.G.Eckert,R.M.Drake; " Analysis of Heat and Mass Transfer ", Mc Graw-Hill, NewYork, 1972.

Nomenclature

d: diameter
H: enthalpy
K: acceleration
P: Pressure
Pr: Prandtl Number
q : heat flux
r : radius
Re: Reynolds Number
St : Stanten Number
u : Gas velocity component parallel to the walls
x: distance along the wall
$\emptyset$: energy thickness
μ: Viscosity
ρ: Local Gas density

Suffixes :

d : diameter as ref. dim.
e : condition at free stream edge of boundary layer
w : wall conditions