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Gas Effect at High Temperature on the Supersonic Nozzle Conception

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Abstract

The aim of this work is to develop a new computational program to determine the effect of using the gas of propulsion of combustion chamber at high temperature on the shape of the two-dimensional Minimum Length Nozzle giving a uniform and parallel flow at the exit section using the method of characteristics. The selected gases are H_2, O_2, N_2, CO, CO_2, H_2O, NH_3, CH_4 and air. All design parameters depend on the stagnation temperature, the exit Mach number and the used gas. The specific heat at constant pressure varies with the temperature and the selected gas. The gas is still considered as perfect. It is calorically imperfect and thermally perfect below the threshold of dissociation of molecules. A error calculation between the parameters of different gases with air is done in this case for purposes of comparison. Endless forms of nozzles may be found based on the choise of T_0, M_E and the selected gas. For nozzles delivering same exit Mach number with the same stagnation temperature, we can choose the right gas for aerospace manufacturing rockets, missiles and supersonic aircraft and for supersonic blowers as needed in settings conception.

Key words: Supersonic Minimim Length Nozzle, High temperature, Calorically imperfect gas, Numerical integration, Gas

1. Introduction

Gases play an important role in aerospace propulsion including supersonic jets gear. The use of such gas of propulsion essentially affects the behavior of the supersonic flows, and in particular on all design parameters. The choice of such gas to propel aerospace vehicles is done on the basis of the need in the design parameters and for construction specifications. For example, the nozzles used for supersonic propulsion of rockets, missiles and supersonic aircraft engines; it is desired to have short lengths and mass of the nozzles to have a reduced weight of the vehicle. As for the construction of supersonic blowers, it is desired to have nozzles that in some cases have a low temperature distribution and a large exit section enough to place the aircraft or missiles prototype and instruments and measuring devices without damage [1-4]. For both constructs is still desired to have a uniform and parallel flow to the exit section to prevent loss of thrust.

The model with constant C_P is typically used at low temperatures to illustrate the design parameters of various forms of supersonic nozzles [2, 5].

In references [3-4, 6-7], studies at *HT* on the supersonic nozzles design only for air are presented.

Thermodynamic parameters of a supersonic flow at *HT*, with application for air in supersonic nozzle are presented in reference [8].

The aim of this work is to develop a new computational program to study the effect of using of propulsion gases at

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HT on the design and sizing of the 2D supersonic *MLN* giving a uniform and parallel flow at the exit section. Then this study is to allow a suitable choice of gas in accordance with parameters such as the required coefficient C_F, Exit Mach number, choice of construction material and the stress applied on the wall. The selected substances are limited to 9 gases indicated by table 1 [9-12]. The application range of the temperature is chosen [500K, 3000K]. While for the Mach number is [1.00, 5.00].

The treated gases are selected from the group of gases found in the literature, having differents thermodynamic properties. We focus on the specific heat at constant pressure $C_P(T)$ at *HT* and the constant R. This function $C_P(T)$ is available depending on the temperature in several references [9-12].

The molecules of these gases have one or two kind of atoms which are H_2, O_2, N_2, CO, CO_2 and H_2O, NH_3, CH_4 and air. The selected gases require that the temperature is within a specific interval for not having the molecules dissociation. This interval varies from one gas to another. The gas is considered as perfect. Including the equation of state ($P=\rho RT$) is still valid, except it will be considered calorically imperfect and thermally perfect.

Figure 1 shows the various internal regions of the 2D MLN supersonic flow to have a uniform and parallel flow to the exit section. So the area *OAB* is appointed by Kernel region. It is of not simple type. The region *ABE* is named by transition region. It is of simple region and the BSE region is appointed by uniform region. However, the Mach number is constant at all points in this region. The wall of the nozzle is a priori

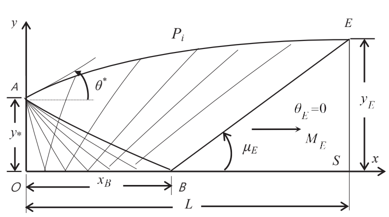

Fig. 1. Presentation of the flow field in the nozzle

Table 1. Coefficients of $C_P(T)$ function and constants thermodynamic for selected gases

N	Gas	a' J/(K mol)	b' J/(K^2 mol)	c' J K / mol	R J/(kg K)
1	H_2	27.28	3.26	0.50	4157.250
2	O_2	29.96	4.18	-1.67	259.828
3	N_2	28.58	3.76	-0.50	296.946
4	CO	28.41	4.10	-0.46	296.946
5	CO_2	44.22	8.79	-8.62	188.965
6	H_2O	30.54	10.29	0.08	461.916
7	NH_3	29.75	25.10	-1.55	489.088
8	CH_4	23.64	47.86	-1.92	519.656
9	Air	Polynomial of 9th degree [3, 9-11]			287.102

unknown. It is determined numerically to have the desired condition. The search of the wall and the calculation of the internal flow are done by the method of characteristics in the case of *HT* assumptions [3].

In the literature, we find a change according to equation (1) of the specific heat at constant pressure. Constants of this function and the thermodynamic constants of gas are shown in the table 1 [9-12].

$$C_P(T)=a'+b'T+\frac{c'}{T^2} \tag{1}$$

For air, the law of variation of $C_P(T)$ is chosen as a polynomial of 9th degree [8].

The $C_P(T)$ function is found in Joule/(mol K). One needs to convert it to Joule/(kg K). For this, we used the following relationship (2).

$$C_P\left(\frac{Joule}{kg\ K}\right)=C_P\left(\frac{Joule}{mol\ K}\right)\times\frac{R}{8.3145} \tag{2}$$

2. Mathematical Formulation

The calculation is based on the use of the method of characteristics at *HT*. The equations of compatibilities and characteristics respectively valid on the upward and downward characteristic, as presented in Fig. 2, are represented by [3]:

$$\begin{cases} -\dfrac{C_P(T)}{2\,H(T)}\sqrt{M^2(T)-1}\,dT+d\theta=0 \\ \dfrac{dy}{dx}=\tan(\theta-\mu) \end{cases} \tag{3}$$

$$\begin{cases} -\dfrac{C_P(T)}{2\,H(T)}\sqrt{M^2(T)-1}\,dT-d\theta=0 \\ \dfrac{dy}{dx}=\tan(\theta+\mu) \end{cases} \tag{4}$$

With

$$H(T)=\int_T^{T_0} C_P(T)\,dT \tag{5}$$

The values of *M* and *T* at *HT* are connected by [3]:

$$M(T)=\frac{\sqrt{2H(T)}}{a(T)} \tag{6}$$

$$a^2(T)=\gamma(T)\,R\,T \tag{7}$$

$$\gamma(T)=\frac{C_P(T)}{C_P(T)-R} \tag{8}$$

$$\mu=arcsin\left(\frac{1}{M}\right) \tag{9}$$

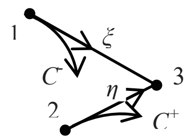

Fig. 2. Illustration of the upward and downward Mach lines.

The ratios ρ/ρ_0 and P/P_0 are calculated by [8]:

$$\frac{\rho}{\rho_0}=\mathrm{Exp}\left(-\int_T^{T_0}\frac{C_P(T)}{a^2(T)}\,dT\right) \tag{10}$$

$$\frac{P}{P_0}=\frac{T}{T_0}\frac{\rho}{\rho_0} \tag{11}$$

As the flow through at the throat and the exit section of the nozzle is unidirectional, the ratio of critical sections at HT, given by equation (12) remains valid for the convergence of the found numerical results [8]:

$$\frac{y_E}{y_*}=Exp\left(\int_{T_E}^{T_*}C_P(T)\left[\frac{1}{a^2(T)}-\frac{1}{2\,H(T)}\right]dT\right) \tag{12}$$

3. Internal Flow

First, one must determine the temperature $T*$ and T_E of the throat and the exit section by solving the equation (6) when (M=1.00, T=$T*$) and (M=M_E, T=T_E) by the method of bipartition respectively [3, 13-14]. The ratio of critical sections given by equation (12) is determined by using the Simpson quadrature [3, 13-14].

The calculation of the Prandtl Meyer function value $v=v_E(T_E)$ corresponding to $T=T_E$ of the exit section is needed to calculate the internal flow by the following relationship [15-16]:

$$\nu(T)=\int_T^{T_*}\frac{C_P(T)}{2H(T)}\sqrt{M^2(T)-1}\;dT \tag{13}$$

The discretization of the system (3) and (4) gives four algebraic equations with four unknowns (x, y, T and θ) at each point of the flow field. The resolution was made by the finite difference method with predictor corrector [2-3, 13, 17-18].

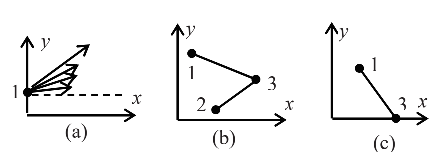

Fig. 3. Different points of the flow field. (a) : Point at the expansion center (Point A in Fig. 1). (b) : Internal point. (c) : Point on the axis of symmetry.

We can have three types of points in the flow as presented in Fig. 3.

The flow parameters in the points 1 and 2 of the Fig. 3 are known. The problem is to determine the parameters in the point 3 by solving the system of equations (3) and (4) simultaneously.

Parameters at a point depend only on the properties at the upstream points. It is a property of a supersonic flow [2-3, 17-18].

The Kernel region is characterized by the length of the segment OB of Fig. 1. The calculation is limited only to the Kernel region since the flow is 2D at all points of the intersection of characteristics. At the beginning of each downward charactéristic, the incrementing angle of deflection of the flow at the point A by $\Delta\theta=\theta^*/N=v_E/(2N)$ and calculating the corresponding temperature by reversing the Prandtl Meyer function (13). As a result, we calculate the Mach number corresponding by equation (6). Calculating the integral (13) is donne by using simultaneously the Gauss Legendre quadrature [18] and the bipartition algorithm [3, 13].

4. Calculation procedure of the wall

The wall of the nozzle passes through point A. At this point, we have x_A=0.0, y_A=$y*$, M_A=M^* and θ_A=θ^*. The value of M^* is equal to the Mach number at point A just after the expansion, corresponding to the Mach number at the first point on the last descendant characteristic AB, see Fig. 1.

The properties of the flow on each upstream line in the transition region ABE are constants. Then the parameters (T, θ, M, P, ρ) on points P_i (i=1, 2, ..., N) of the wall are known and we have only to determine the position (x, y) of each points. The position of point P_i can be determined by the following relationships:

$$x_{P_i}=\frac{y_i-y_{P_{i-1}}+x_{P_{i-1}}tg(\alpha_{i-1})-x_i\;tg(\theta_i+\mu_i)}{tg(\alpha_{i-1})-tg(\theta_i+\mu_i)}\qquad i=2,..N \tag{14}$$

$$y_{P_i}=y_{P_{i-1}}+\left(x_{P_i}-x_{P_{i-1}}\right)tg(\alpha_{i-1})$$

with

$$\alpha_{i-1}=\frac{\theta_{i-1}+\theta_i}{2} \tag{15}$$

For i=1, we have x_{P_1} point A of the Fig. 1. When i=N, the length of the nozzle is obtained in the non-dimensional form, by:

$$\frac{L}{y_*}=\frac{x_E}{y_*}=\frac{x_{P_N}}{y_*} \tag{16}$$

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The exit section corresponding to the discretization of N points is given in non-dimensional form by:

$$\frac{y_E}{y_*}(computed) = \frac{y_{P_N}}{y_*} \tag{17}$$

To achieve convergence, it is necessary that the value given by equation (17) tends towards the value given by equation (12). In this case, all the design parameters also converge to the exact solution. The convergence of the results depends on the number N of downward characteristics from the expansion center A.

The calculation of the mass of the nozzle is therefore given, for a nozzle having the same wall thickness, in the non-dimensional form, by [3]:

$$C_{Mass} = 2\sum_{j=1}^{j=N-1}\left[\left(\frac{x_{j+1}}{y_*}-\frac{x_j}{y_*}\right)^2+\left(\frac{y_{j+1}}{y_*}-\frac{y_j}{y_*}\right)^2\right] \tag{18}$$

The axial pressure force exerted on the wall can be calculated again, in the non-dimensional form, by [3]:

$$C_F = 2\sum_{j=1}^{j=N-1}\left(\frac{P}{P_0}\right)_{(j)}\left[\frac{y_{j+1}}{y_*}-\frac{y_j}{y_*}\right] \tag{19}$$

By adjusting the various gases, stagnation temperature and the exit Mach number, we can determine the design parameters for each used gas.

5. Comparison with air

The air is generally used as a propellant gas in the most aerospace applications as there are with quantities in nature. Similarly, the studies presented in references [3] on the supersonic nozzles design are made only for air.

To present the benefits of such a gas, the error between the air parameters and settings for other gas is determined by the following relationship:

$$\varepsilon(\%) = \left|1-\frac{Parameter_{Gas}}{Parameter_{Air}}\right| \times 100 \tag{20}$$

6. Results and comments

Curve 1 in the illustrated figures shows the variation of the parameter for the H_2 gas. Curve 2 for O_2. Curve 3 for N_2. Curve 4 for CO. Curve 5 CO_2. Curve 6 for H_2O. Curve 7 for NH_3. Curve 8 for CH_4 and the curve 9 for air. The Fig. 22 contains 8 curves for the same number of gas. The Fig. 5, 6, 7, 8 and 21 are followed by tabulated results for each gas to view the numerical founded values.

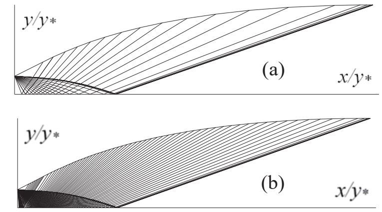

Fig. 4. Meshes in characteristics. (a) : Large mesh. (b) : Fine mesh.

The results for air (curve 9) in the figures can be found in reference [3]. They are presented for the purpose of the comparison with other gases.

Figure 4 shows an example of meshing characteristics. We can notice the characteristics in the Kernel and transition regions. Large and fine meshes are shown in Figs. 4a and 4b. The convergence of the design results depends on the considered mesh for the calculation, which itself depends on the number of selected characteristics in the Kernel region, downswing from point A of Fig. 1. Then a fine mesh gives good results. For example in the Fig. 4a where used only 20 characteristics and in Fig.4b where used 75 characteristics.

Figures 5, 6, 7 and 8 illustrate the effect of the propellant gas on the shape of the 2D supersonic *MLN*, giving M_E at the exit section, respectively for M_E=2.00, 3.00, 4.00 and 5, 00 when T_0=2000 K. The design results are shown in tables 2, 3, 4 and 5.

It is clear that the propellant gas effect on the nozzle shape, and therefore on all design parameters, in accordance with tabulated results. For example CH_4 gas provides a large length and mass as well as the very high C_F. While the H_2 gas, N_2, O_2 and CO give small parameters. For supersonic nozzles construction applied to missiles and aircraft is

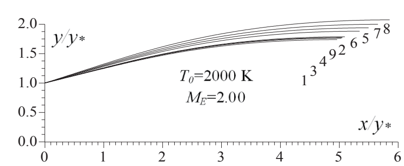

Fig. 5. Effect of gas on the nozzles shape giving M_E=2.00.

Table 2. Numerical values for Fig. 5.

N	Gas	θ^* (deg)	L/y_*	C_{Mass}	C_F	y_E/y_*
1	H_2	13.900	4.963	5.034	0.168	1.737
2	O_2	14.526	5.092	5.171	0.181	1.785
3	N_2	14.236	5.032	5.107	0.175	1.763
4	CO	14.300	5.044	5.121	0.176	1.768
5	CO_2	16.383	5.499	5.604	0.222	1.937
6	H_2O	15.718	5.346	5.441	0.206	1.880
7	NH_3	17.079	5.658	5.774	0.238	1.997
8	CH_4	17.903	5.857	5.988	0.259	2.073
9	Air [3]	14.313	5.051	5.128	0.177	1.770

recommended for use for example the H_2 gas, N_2 or CO instead of CH_4, NH_3 even the air. For example, CH_4 and NH_3 gas are not recommended. While for supersonic blowers can be used which have large exit sections. In this case the CH_4, NH_3, H_2O and CO_2 are recommended. The influence of M_E and T_0 is noted on the form and the parameters. H_2, N_2, O_2 and CO gases gives better performance compared to the air for applications in aerospace manufacturing. The shapes of the nozzles in the case of air can beings found in reference [3].

Figure 9 presents the influence of the gas and T_0 on the

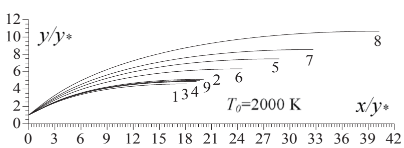

Fig. 6. Effect of gas on the nozzles shape giving M_E=3.00.

Table 3. Numerical values for Fig. 6.

N	Gas	θ^* (deg)	L/y_*	C_{Mass}	C_F	y_E/y_*
1	H_2	26.228	18.173	18.724	0.323	4.582
2	O_2	27.885	20.134	20.796	0.357	5.113
3	N_2	27.101	19.175	19.782	0.341	4.853
4	CO	27.239	19.326	19.942	0.344	4.895
5	CO_2	33.441	28.783	29.970	0.482	7.476
6	H_2O	31.078	24.550	25.476	0.426	6.319
7	NH_3	35.440	32.685	34.126	0.529	8.556
8	CH_4	38.503	40.277	42.218	0.607	10.661
9	Air [3]	27.465	19.671	20.306	0.349	4.986

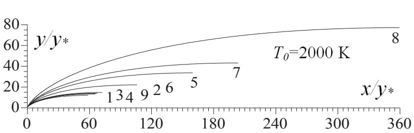

Fig. 7. Effect of gas on the nozzles shape giving M_E=4.00.

Table 4. Numerical values for Fig. 7.

N	Gas	θ^* (deg)	L/y_*	C_{Mass}	C_F	y_E/y_*
1	H_2	34.474	58.565	60.393	0.411	11.907
2	O_2	37.009	71.764	74.154	0.460	14.714
3	N_2	35.793	65.036	67.137	0.437	13.282
4	CO	35.967	65.819	67.955	0.440	13.450
5	CO_2	46.386	159.659	165.967	0.660	33.596
6	H_2O	41.851	105.817	109.713	0.559	22.016
7	NH_3	49.382	203.502	211.865	0.728	43.133
8	CH_4	55.525	359.559	375.239	0.878	77.147
9	Air [3]	36.264	67.485	69.693	0.446	13.804

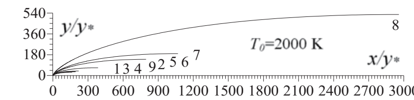

Fig. 8. Effect of gas on the nozzles shape giving M_E=5.00.

Table 5. Numerical values for Fig. 8.

N	Gas	θ^* (deg)	L/y_*	C_{Mass}	C_F	y_E/y_*
1	H_2	40.060	163.669	168.029	0.461	27.842
2	O_2	43.209	220.341	226.488	0.519	37.766
3	N_2	41.685	190.471	195.674	0.491	32.531
4	CO	41.869	193.094	198.384	0.494	32.996
5	CO_2	55.743	795.536	820.270	0.771	139.516
6	H_2O	49.160	385.330	396.813	0.636	66.878
7	NH_3	59.082	1068.77	1102.74	0.847	188.424
8	CH_4	68.155	2957.76	3054.48	1.057	526.941
9	Air [3]	42.069	195.055	200.423	0.499	33.353

shape of the supersonic nozzle when M_E=3.00 for T_0=1000 K, 2000 K and 3000 K. The selected gases are the CH_4 and H_2 respectively shown in Fig. 9a and 9b. The increasing of T_0 demand large lengths and mass and different shapes of the nozzle to keep the uniform and parallel flow at the exit section. The numerical results of the curve 2 can beings found in table 3. The curve 2 in Fig. 9a and 9b, respectively is the curve 1 and the curve 8 of Fig. 6.

Figures 10, 11, 12, 13, 14 and 15 show the effect of gas on the variation of various parameters as a function of the exit Mach number of the nozzle M_E, for T_0=2000 K. In Figs. 11, 12, 13 and 15 was preferred the presentation of the natural Logarithmic scale of parameters of L_d/y_*, L/y_*, C_{Mass} and y_E/y_* respectively since the found values are very large for some gases such as CH_4, NH_3, CO_2 and H_2O and very small values for the other gases such as H_2, O_2, N_2, CO and air at the same figure. The presentation will be bad in the real scale. The gas giving very small values is the H_2, while the CH_4 gives the larger values. At low M_E until about 2.00, it is not influenced by a choice remark such gas on the design parameters. While most M_E becomes big and starts to exceed 2.00 about the

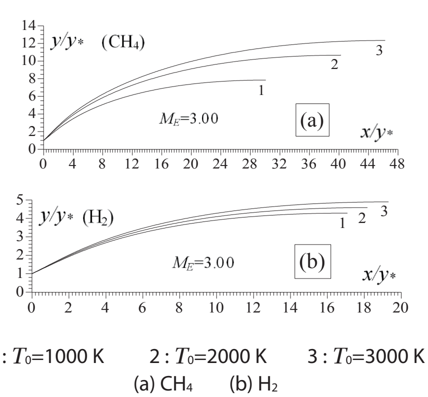

Fig. 9. Effect of T_0 on the shape of the nozzles for CH_4 and H_2 gases.

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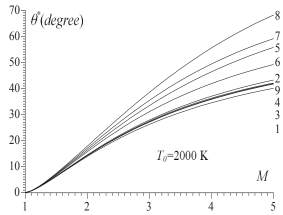

Fig. 10. Effect of the gas on θ^* of the throat.

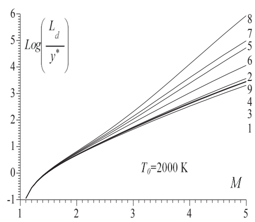

Fig. 11. Effect of gas on the L_d/y_*.

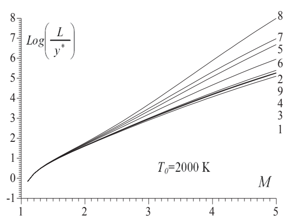

Fig. 12. Effect of gas on the length L/y_*.

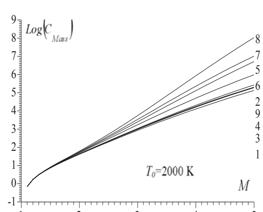

Fig. 13. Effect og gas on C_{Mass} of the nozzle.

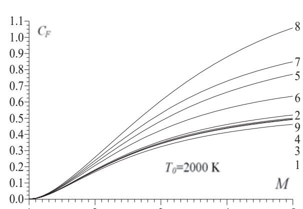

Fig. 14. Effect of gas on the coefficient C_F.

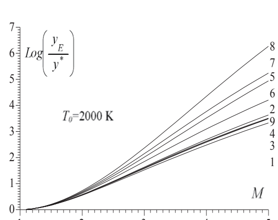

Fig. 15. Effect of gas on y_E/y_* of nozzles.

choice of such a gas is essential for propulsion.

Figures 16, 17, 18 and 19 represent the variation of the thermodynamic parameters and the Mach number through the wall of the nozzle of Fig. 6 for T_0=2000 K and M_E=3.00. It is noted an expansion of gas from M=M^* just after the expansion (at the throat) to the exit for M=M_E. The value of M^* is different from one gas to another, affecting critical parameters and settings to the throat just after the expansion.

In Fig. 17 we see that the temperature through the wall is high enough for the CH_4 gas, NH_3, CO_2 and H_2O relative to the air. While for the H_2 gas, N_2, and CO is cold enough with respect to air. So we must chose a suitable building material resistant to the distribution shown in Fig. 17 according to the selected gas.

Figure 19 shows that there is an expansion of Prandtl Meyer type from θ=θ^* at the throat to θ=0 at the exit. It also

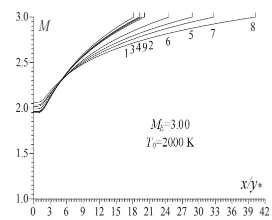

Fig. 16. Effect of gas on the variation of the Mach number through the nozzle wall.

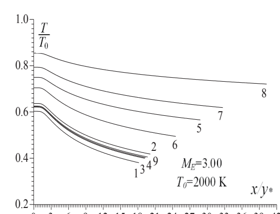

Fig. 17. Effect of gas on the variation of T/T_0 through the nozzle wall.

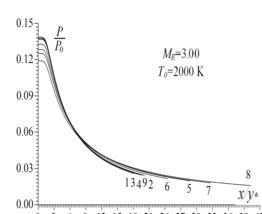

Fig. 18. Effect of gas on the variation of P/P_0 through the nozzle wall.

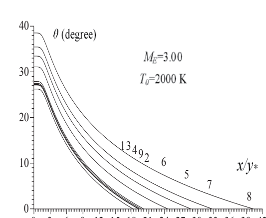

Fig. 19. Effect of gas on the variation of θ through the nozzle wall.

shows that the flow is horizontal to the exit section.

Figure 20 shows the variation of the Mach number across the axis of symmetry of the nozzle of Fig. 6 when M_E=3.00 for T_0=2000 K. It is noted that there is an expansion from M=1.00 to M_E=3.00. We still notice the existence of the Kernel and uniform regions. For CH_4 gas, NH_3, CO_2 and H_2O in the uniform area length is larger than the air because in this case the nozzle has a greater length than the case of air. While for H_2, N_2, CO and O_2 have even smaller areas. So for supersonic wind tunnels, we have a large space to place the prototype aircraft in the uniform region and the effect of the wall is not important what is seen far enough compared to the prototype.

Figure 21 show the gas effect on the shape of the nozzles having same radius of the exit section y_E/y_* as the case of air. The example taken is for M_E=3.00, and T_0=2000 K.

Notice that if we keep the same design parameter for the design of nozzles, the exit Mach number, the shape of the nozzle and the other design parameters are not the same for all gases.

When we take a single parameter of the same design for all gases, as if the air according to the figures, the H_2 gas

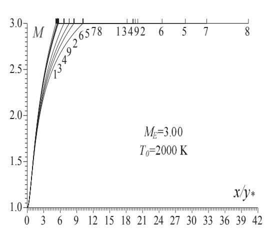

Fig. 20. Effect of gas on the variation of the Mach number along the axis of symmetry of the nozzle.

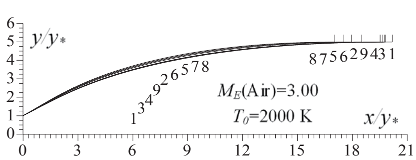

Fig. 21. Effect of the gas on the shape of the nozzles having the same exit section.

Table 6. Numerical values of the nozzles for Fig. 21.

N	Gas	M_E	θ^* (deg)	L/y_*	C_{Mass}	C_F
1	H_2	3.0850	27.069	20.184	20.806	0.333
2	O_2	2.9769	27.628	19.522	20.161	0.354
3	N_2	3.0257	27.373	19.821	20.453	0.344
4	CO	3.0176	27.425	19.770	20.403	0.346
5	CO_2	2.7230	29.110	17.953	18.635	0.418
6	H_2O	2.8148	28.590	18.516	19.183	0.392
7	NH_3	2.6575	29.586	17.534	18.230	0.438
8	CH_4	2.5789	30.132	17.039	17.750	0.466
9	Air [3]	3.0000	27.465	19.671	20.306	0.349

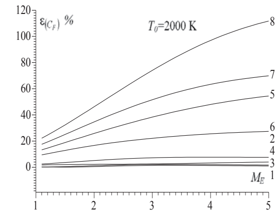

Fig. 22. Effect of the gas on the error given by C_F as compared to air.

delivers the highest M_E, which exceeding the air. Same goes for N_2 gas, CO gas. While for O_2 gas, CO_2, H_2O, NH_3 and CH_4 deliver a lower than air M_E. CH_4 is the lowest. The shape, length, radius of the exit section for the H_2 are very large in comparison with other gas and air. CH_4 gas always delivers high compared to other gas. The design parameters are also different. The values of y_E/y_* for air are presented in table 3 and can still beings found in references [3]. The numerical results are shown in table 6.

It was noted that if we keep the same settings for all gases, the other parameters will be different including the shape of the nozzle. Now if we keep the same shape of the air nozzle delivers an exit Mach number M_E for given T_0, and other gas by air is changed, the first point mentioned is that the flow through the exit section remains more uniform and parallel. It is estimated that we will see a degradation of M_E for CO_2, H_2O, CH_4 and NH_3 compared to air and increased M_E for H_2, O_2, N_2 and CO. The distribution areas of flow of Fig. 1 will not be respected, and there is only one area of not simple flow throughout the nozzle type and the BSE uniform area to disappear. So to change the air by other gases, we must devise another form of the nozzle to keep the condition of uniform and parallel flow at the exit section according to the results presented in this work.

Figure 22 shows the error committed by the use of such a propulsion gas instead of air on the C_F coefficient according to M_E for T_0=2000K. Of all design parameters were chosen only C_F because it is interest and all other parameters have the same rate of change as the C_F except the change in values. We note the significant difference for CH_4, NH_3, CO_2 and H_2O relative to the air. While for H_2, O_2, N_2 and CO error is very small. For M_E=5.00, $\varepsilon_{CH_4}(\%)$= 111.71% and $\varepsilon_{H_2}(\%)$=7.53%, which shows the possibility of using the H_2 gas instead of air, even for N_2, CO and O_2 and impossibility of using CH_4, NH_3, H_2O and CO_2 instead of air.

7. Conclusion

This work allows us to study the effect of using gas of propulsion on the design parameters of two-dimensional supersonic nozzles type *MLN* at *HT*. We can draw the following conclusions

1. The computer program can do any gas found in the nature. It should be added in this case, the $C_P(T)$ and the gas constant R and calculate the corresponding $H(T)$.
2. The $C_P(T)$ function, R and $\gamma(T)$ characterize the calorific value of the gas and mainly affects all design parameters.
3. For applications of missiles and supersonic aircraft, it is recommended to use propellant gas having the smallest ratio γ to have a small mass and large C_F. Among the selected gas, CH_4 is a bad choice and the H_2 is a good choice.
4. The convergence of design parameters and sizing depend on the choice of the number of downward characteristics from the center of expansion of the nozzle.
5. For construction applications of supersonic blowers, it is recommended to use the propellant gas having a ratio γ as large as possible to have a large nozzle and an ratio $A_E/A*$ big. Among the selected gas, CH_4, NH_3, CO_2 and H_2O is a good choice and the H_2, O_2, N_2 and CO is a bad choice instead of air.
6. The stagnation temperature and the exit Mach number also affect the shape of the nozzle and all design parameters.
7. If we keep the given form of the nozzle for the air, the use of another gas instead of air will lose the condition of uniform and parallel flow at the exit section. Among the selected gas, the use of H_2, N_2, CO by increasing the exit Mach number and the use of gas CH_4, NH_3, CO_2 and H_2O will degrade the exit Mach number.
8. The convergence of results is controlled by the convergence of the ratio of critical sections, calculated numerically by the relationship (17), to that given by equation (12). Others design parameters also converge.
9. Endless shaped nozzles can be found by varying three parameters which are M_E, T_0 and the gas itself ($C_P(T)$, R).
10. The propellant gas affects the thermodynamic parameters P/P_0, T/T_0, ρ/ρ_0 and geometric $y_E/y*$ of the flow.

As prospects, we can study the effect of gas of propulsion at *HT* on the design and sizing of the axisymmetric *MLN* and other types such as Plug Nozzle, Expansion Deflexion nozzle 2D and 3D by the method of characteristics. These studies will not be possible unless it is still studying the effect of gas on the Prandtl Meyer function at *HT*. In the case of air, these studies are presented in references [3-4, 6-7].

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Appendix A

M	Mach number
x	Abscissa of the nozzle section
y	Radius of the nozzle section
a	Speed of sound
μ	Mach angle
v	Prandtl Meyer function
θ	Deviation of the flow
N	Number of downward characteristics
P	Pressure
T	Temperature
R	Thermodynamic constant of gas
H	Enthalpy
C_P	Specific heat at constant pressure
C_F	Pressure force coefficient
L	Length of the nozzle and for Kernel region
C_{Mass}	Mass of the nozzle in nondimensionnal value
γ	Specific heats ratio
ρ	Density
ε	Error of computation
a', b', c'	Coefficients of the $C_P(T)$ function
HT	High Temperature abreviation
MLN	Minimum Length Nozzle abreviation

Indices

0	Stagnation condition (combustion chamber)
*	Critical condition
E	Exit section
i	Point
A	Throat of the nozzle
d	Kernel region