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METHOD OF CALCULATING ROCKET ENGINE NOZZLE CONTOUR FOR OPTIMUM THRUST*
by Tian Minhua

1. Principles of Calculating Nozzle Contour for Optimum Thrust

This work considers an ideal two-dimensional, steady-state, irrotational, isentropic flow. A flow field inside a nozzle is axisymmetric and we chose to use a cylindrical coordinate system. In Figure 1 ABTE represents the line of intersection between the contour surface of the nozzle and the meridian plane. The converging section of the nozzle $\widehat{AB}$ is a circular arc wall surface with a curvature radius of R_1. The initial expansion section $\widehat{BT}$ downstream of the throat is a circular arc wall surface with a curvature radius of R_2. We chose a nozzle contour $\widehat{TE}$ which, at a given nozzle length L and ambient pressure p_a, will cause the thrust produced by the nozzle to reach maximum. We called this contour "the nozzle contour for optimum thrust". We chose a point C on the nozzle axis and considered a control surface EC passing through a point E at the nozzle outlet. This control surface $\widehat{EC}$ and the right characteristic curve passing through circular arc separation point T, intersect at point D (see Fig. 1). Engine thrust R can be obtained from the aerodynamic

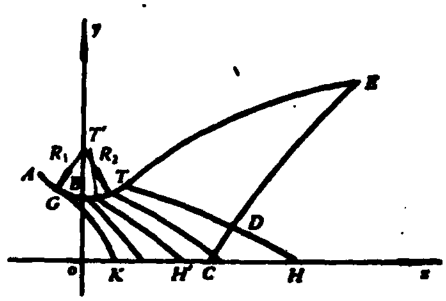

Fig. 1

* This paper was given in April,1980 at the Third All-China Scientific Conference on Engineering Thermophysics at Guilin.

parameters at control surface EC.

$$R = \int_C^E \left[(p - p_a) + \rho v^2 \frac{\sin(\phi - \theta)\cos\theta}{\sin\phi} \right] 2\pi y dy \quad (1)$$

(p, ρ, v, θ and ϕ represent flow pressure, density, velocity, angle of flow, and the angle of incidence between the control surface and the x axis, respectively.)

Since point C is fixed and nozzle length is invariant, the nozzle length after point C is also invariant, i. e.,

$$x_E - x_C = \int_C^E \operatorname{ctg}\phi dy = \text{constant} \quad (2)$$

Moreover, due to the continuity of the flow rate, the flow rate past the control surface must be equal to the flow rate past throat cross section G, i. e.,

$$G = \int_C^E \rho v \frac{\sin(\phi - \theta)}{\sin\phi} 2\pi y dy = \text{constant} \quad (3)$$

The problem winds up as that of finding the condition below which satisfies expressions (2) and (3) and causes thrust R to reach peak value, i. e., causes functional expression (1) to take a conditional peak value. Using the Lagrange product factor[1], the conditional peak value problem becomes an unconditional peak value. Since we can obtain a variational method which causes thrust R to reach maximum value, the flow parameters at control surface EC should satisfy each of the following conditions: (given an ambient pressure $p_a = 0$)

(1) At outlet end point E we have:

$$\sin 2\theta_E = (2/\kappa M_E^2)\operatorname{ctg}\alpha_E \quad (4)$$

(2) Control surface ED serves as the left characteristic curve:

$$dy/dx = \operatorname{tg}(\theta + \alpha)$$

$$\frac{d\theta}{dx} - \frac{\operatorname{ctg}\alpha}{v}\frac{dv}{dx} + \frac{\sin\theta\sin\alpha}{y\cos(\theta + \alpha)} = 0 \quad (5)$$

(3) The aerodynamic parameters at ED also satisfy the two following relationships:

$$M^* \cos(\theta - \alpha)/\cos\alpha = M_E^* \cos(\theta_E - \alpha_E)/\cos\alpha_E \quad (6)$$

$$yM^2\left(1+\frac{\kappa-1}{2}M^2\right)^{-\frac{\kappa}{\kappa-1}}\sin^2\theta\,\mathrm{tg}\,\alpha = y_E M_E^2\left(1+\frac{\kappa-1}{2}M_E^2\right)^{-\frac{\kappa}{\kappa-1}}\sin^2\theta_E\,\mathrm{tg}\,\alpha_E \qquad (7)$$

where

$$M^* = \left(\frac{1}{\kappa-1+2/M^2}\right)^{\frac{1}{2}}$$

(4) Due to the continuity of the flow rate the following expression clearly holds true:

$$\int_D^E \rho v \frac{\sin\alpha}{\sin(\theta+\alpha)} 2\pi y dy = \int_D^T \rho v \frac{\sin\alpha}{\cos(\theta-\alpha)} 2\pi y dx \qquad (8)$$

(κ, α and M represent the ratio of specific heats of the gas, the Mach angle, and the Mach number, respectively.)

2. General Description of the Calculation Process

In nozzle design, the initial conditions which are usually given are nozzle throat radii R_1 and R_2, ratio of specific heats of the gas κ, and nozzle expansion area ratio A_E, in order to determine the nozzle contour for optimum thrust. With regard to rocket engine nozzles, in order to boost the thrust and decrease the length it is necessary to employ a relatively small throat wall curvature radius. We employed the conformal curve coordinate method in [2] to calculate the parameters at initial transonic line GK of the nozzle throat (Fig. 1). This method of calculation is simple, its accuracy is fairly good, and it is suitable for cases where the throat wall curvature radius is relatively small. The entire supersonic flow field was calculated and we made use of a characteristic curve graph[8]. As to the left and right characteristic curve equations, we employed difference equations with two-dimensional accuracy, i. e.,

$$dy/dx = \mathrm{tg}(\theta^* \pm \alpha^*)$$

$$\Delta\theta \mp \mathrm{ctg}\,\alpha^* \frac{\Delta v}{v^*} \pm \frac{\sin\theta^*\sin\alpha^*}{y^*\cos(\theta^*\pm\alpha^*)}\Delta x = 0$$

(The parameters with "*" represent mean values of Δx).

The following relates the method for finding the nozzle contour for optimum thrust TE (see Fig. 1). The flow parameters at right characteristic curve T'H', which runs from the wall to the axis, were determined one by one according to the characteristic curve method. We first assume $M_B^{(1)}$, and can obtain $\theta_B^{(1)}$, by expression (4). Then we look for a point D on T'H' which will allow expression (6) to be satisfied. If we are unable to find point D on T'H', then we can again find the next characteristic curve. When we find point D, we can determine $y_B^{(1)}$, by expression (7) and then can determine the parameters of the various points on ED by expressions (5 - 7) and at the same time use expression (8) to decide whether or not T' is separation point T. If expression (8) holds true, then we find the next right characteristic curve T'H' and point D, until expression (8) is satisfied. This time T'H' will be the TH right characteristic curve which is called for. Then $A_B^{(1)}$ is determined on the basis of $y_B^{(1)}$ and compared with the given A_B. Normally $A_B^{(1)}$ will not be equal to A_B. Using the "shooting" method we can select an appropriate $M_B^{(i)}$, repeatedly using the above described method to obtain $A_B^{(i)}$ until it is sufficiently close to A_B.

Then once more using the parameters at right characteristic curve TD and control surface ED as the initial conditions, on the basis of wall TE serving as the line of flow, the flow field parameters of triangular region TDE (Fig. 1) can be calculated using the characteristic curve method, thereby obtaining the nozzle contour for optimum thrust TE.

3. Calculation Results and Discussion

Table 1 below lists the separation point (point T) parameters and the outlet end point (point E) parameters for the nozzle wall contour for optimum thrust at different expansion area ratios A_B and throat divergent section radii R_2. In the table, L is nozzle length and C_F is the nozzle thrust coefficient.

By the calculated results it can be seen that when the area ratio is fixed, the smaller the nozzle throat divergent section radius R_2, the smaller the nozzle length L. In order to reduce nozzle length we can appropriately reduce R_2. But R_2 cannot be excessively small since the smaller R_2 the larger the expansion angle θ_T at the initial separation point. When R_2 is too small the throat flow can expand too quickly causing flow losses to increase. Consequently, it can be determined by tests down to what value R_2 will be more suitable. Under normal conditions, the throat wall curvature radius can take: $1 < R_1 < 2$ and $0.5 < R_2 < 1$.

Table 1. $R_1 = 2$, $\kappa = 1.23$

A_E	R_2	x_T	y_T	θ_T	L	θ_E	M_E	C_R
20	0.5	0.277	1.084	33°36′	8.399	13°10′	3.5147	1.7605
	1.0	0.551	1.165	33°25′	8.660	13°08′	3.5243	1.7606
	1.5	0.811	1.238	32°43′	8.907	13°06′	3.5326	1.7609
40	0.5	0.295	1.096	36°10′	13.301	11°36′	3.9985	1.8261
	1.0	0.585	1.189	35°49′	13.639	11°34′	4.0062	1.8264
	1.5	0.861	1.272	35°01′	13.955	11°33′	4.0128	1.8265
60	0.5	0.305	1.104	37°37′	17.276	10°49′	4.2903	1.8586
	1.0	0.604	1.203	37°11′	17.669	10°48′	4.2973	1.8588
	1.5	0.888	1.291	36°18′	18.032	10°47′	4.3031	1.8589

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Abstract

Combining the characteristic method with variation principle and applying it to supersonic flow field, a calculating method is presented for the optimum-thrust-nozzle contour design. This method can be used to design nozzle contours with various throat curvature radii and expantion area ratios. It is especially suitable for the design of nozzle contour with large expantion area ratio.

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