Rocket Drag Prediction Method for Optimizing Design and Flight Performance

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ABSTRACT

This paper discuses about rocket aerodynamic drag prediction methods for optimization of range. Methods presented in this paper could be used as an initial guideline or a quick reference to approximate aerodynamic drag of a rocket. Parameters associated to a rocket's aerodynamic drag can be established in three distinctive types: wave drag, skin friction drag and base drag. Details on these drag types generated by the various sections of a rocket's body are explained under various speed conditions.

Keywords: Rocket aerodynamic drag, drag prediction method, rocket design.

1.0 INTRODUCTION

This paper presents a work in progress to study flight performance of a ballistic rocket for optimization of range. Based on trajectory shaping techniques presented by Saeedipour and Mohd Yusof [1] & [2], this paper addresses various methods used to predict a rocket's aerodynamic drag. Being the most dominating factor to a rocket's flight performance, drag prediction method has to be established in greater details at various flight conditions for subsonic flight (M<0.8), transonic flight (0.8≤M≤1.2), low supersonic flight (1.2≤M≤ 1.8) and moderate/high supersonic flight (1.8≤M≤6). Body and fin/wing will be treated separately in order to simplify the prediction method.

Table 1 below is used to explain methods employed to address drag prediction methods at various speed conditions:

Table 1. Drag prediction method at various Mach number.

Mach Number		Subsonic M<0.8	Transonic 0.8<M<1.2	Low supersonic 1.2<M<1.8	Moderate/ high supersonic 1.8<M<6
Wave drag	Body	Empirical method	Semi-empirical method based on Euler solution	Second order Van Dyke(HTVD) plus Modified Newton Theory(MNT) method	2nd Order Shock Expansion Theory (SOSET) plus Modified Newton Theory (MNT method.
				Bonney method	
	Wing	n/a	Empirical method	Linear theory plus Modified Newton Theory (MNT) method	Second order Shock expansion Theory (SOSET) plus Modified Newtonian theory (MNT) along strips method.
Skin friction drag		Van Driest II method Jerger Method			
Base body drag		Improved empirical method. Simplified method			

2.0 ROCKET DRAG

Drag is a major design parameter in satisfying the flight range requirement of tactical rockets, especially supersonic rockets. It is a function of drag coefficient, dynamic pressure, and reference area, given by the equation:

$$D = q\, C_D\, S_{Ref} \tag{1}$$

For a rocket configuration, the reference area is the body cross sectional area. Substituting rocket diameter gives $D = 0.785 q C_D d^2$. The ratio D/C_D is a function of diameter and dynamic pressure commonly used as a typical performance parameter for tactical rockets.

For rocket body-alone, nose wave drag, body friction drag and base drag have significant effect to the rocket flight performance. Similarly for fin/wing alone, wing wave drag and wing friction drag have significant effect to the rocket flight performance.

In order to facilitate understanding on these terms, the following are definiton of the types of drag acting on rocket:

<u>Wave drag</u>

Wave drag is an aerodynamics term that refers to a sudden and very powerful form of drag that appears on flying object at high speeds normally happen at high subsonic or higher. It is caused by shock wave around the flying object. Shock waves radiated away a considerable amount of energy, energy that is "seen" by the flying object as drag.

Skin friction drag

This arises from the tangential stresses due to the viscosity or "stickiness" of the air. When air flows over any part of the flying object there exists, immediately adjacent to the surface, a thin layer of air called the boundary layer, within which the air slows from its high velocity at the edge of the layer to a standstill at the surface itself. Surface friction drag depends upon the rate of change of velocity through the boundary layer, i.e. the velocity gradient. The velocity and hence pressure variations along the length of any surface can have adverse effects on the behavior of the boundary layer.

Base drag

A component of aerodynamic drag caused by a partial vacuum in the rocket's tail area. The vacuum is the hole created by rocket's passage through the air. Base drag changes during flight. While the motor is firing, the drag is minimal since the tremendous volume of gas generated by the motor fills this void. The drag takes a sharp jump at burnout when this gas disappears (note: tracking smoke has very little effect on base drag due to its low density). Base drag can be reduced by the use of a boat tail to transition the main body diameter down to the motor diameter which helps direct air into the evacuated area. When properly designed, a boat tail can reduce base drag below zero (i.e. actually generate a small amount of forward thrust) by making use of the "pumpkin seed" effect.

2.1 Nose wave drag (body alone)

In this section, four cases are presented covering all speed conditions.

2.1.1 Case for Subsonic Flow M<0.8

Empirical method. For subsonic flow, M<0.8 is low enough so that it can be assumed no compressibility effect occur. Hence, it is called viscous separation drag. According to reference 4, for cone half angles, θ_c the flow over cone instead of remaining attached, separate due to the very strong adverse pressure gradient and reattaches downstream. These separations prevent the pressure from decreasing as much as it would in inviscid flow and produce a drag. As a result, an empirical expression for this viscous separation drag, where the important parameter is the angle, δ^* which the nose mates with the shoulder after body [4]. This relation is

$$C_{D,vis} = 0.012\,(\delta^* - 10^\circ), \quad \delta^* > 10$$
$$C_{D,vis} = 0 \qquad \delta^* < 10 \tag{2}$$

2.1.2 Case for Transonic Flow 0.8<M<1.2

Semi empirical method-based on Euler solution. For transonic flow, the flow field solution is nonlinear. Alternatives to obtain the transonic wave drag are:
Full numerical solution
Numerical solution of full potential equation
Numerical solution of Euler equation
Numerical solution of Navier-Stokes equation
Experimental data.

Data from Table 2 [3] & [5] can be used to estimate wave drag bodies for transonic Mach number by interpolation or extrapolation. These computational results are solution from the full potential equation.

Table 2: $C_{D,W}$ for blunted tangent ogives

M_∞	l_n	r_n/r_b = 0	0.5	1.0
0.8	0.75	0.04	0.045	0.092
	1.0	0.025	0.030	0.092
	1.25	0.01	0.015	0.092
	1.5	0.01	0.015	0.092
	2.0	0.01	0.015	0.092
	3.0	0.01	0.015	0.092
	4.0	0.01	0.015	0.092
	5.0	0.01	0.015	0.092
0.95	0.75	0.16	0.189	0.279
	1.0	0.08	0.118	0.279
	1.25	0.04	0.070	0.279
	1.5	0.02	0.042	0.279
	2.0	0.02	0.035	0.279
	3.0	0.02	0.025	0.279
	4.0	0.02	0.020	0.279
	5.0	0.02	0.020	0.279
1.05	0.75	0.280	0.322	0.405
	1.0	0.200	0.252	0.405
	1.25	0.155	0.201	0.405
	1.5	0.135	0.152	0.405
	2.0	0.110	0.111	0.405
	3.0	0.078	0.062	0.405
	4.0	0.055	0.055	0.405
	5.0	0.036	0.050	0.405
				Continue...
1.2	0.75	0.419	0.460	0.55
	1.0	0.331	0.364	0.55
	1.25	0.283	0.286	0.55
	1.5	0.247	0.231	0.55
	2.0	0.194	0.155	0.55
	3.0	0.108	0.102	0.55
	4.0	0.065	0.095	0.55
	5.0	0.038	0.090	0.55

2.1.3 Case for Low Supersonic Flow 1.2<M<1.8

Second order Van Dyke (HTVD) plus Modified Newton Theory (MNT) method. For blunt nose configurations, modified Newtonian theory is used because it is easy and an accurate. Newtonian impact theory assumes that, in the limit of high Mach number, the shock lies on the body, meaning that the disturb flow field lies in an infinitely thin layer between the shock and the body [6]. So the pressure coefficient of the surface;

$$C_p = 2\sin^2\delta \tag{3}$$

Where δ is the angle between the velocity vector and a tangent to the body at the point.

According to Lees [6], the better accuracy can be obtained by replacing the constant '2' by the stagnation pressure coefficient, C_{pO}, where

$$C_{p0}=\frac{2}{\gamma M_\infty^2}\left(\left[\frac{(\gamma+1)M_\infty^2}{2}\right]^{\frac{\gamma}{\gamma-1}}\left[\frac{\gamma+1}{2\gamma M_\infty^2-(\gamma-1)}\right]^{\frac{1}{\gamma-1}}-1\right) \quad (4)$$

In a special case where $\delta=90°$ for blunt nose, the pressure coefficient is derived as follows:

$$C_p=0.9C_{p0} \quad (5)$$

Second order Van Dyke theory [7] combines a second order axial solution for potential equation with first order cross flow solution. The advantage of this method is it is more accurate in the axial direction. So the pressure coefficient at each body station is:

$$C_p(x,\varphi)=\frac{2}{\gamma M_\infty^2}\left(\left[1+\frac{\gamma-1}{2}M_\infty^2\left(1-\frac{U^2+v^2+w^2}{V_\infty^2}\right)\right]^{\frac{\gamma}{\gamma-1}}-1\right) \quad (6)$$

And the force coefficients are

$$C_A=\frac{2}{\pi r_r^2}\int_0^l\int_0^\pi C_p(x,\varphi)\frac{rdr}{dx}d\varphi dx \quad (7)$$

$$C_N=\frac{2}{\pi r_r^2}\int_0^l\int_0^\pi C_p(x,\varphi)\cos(\varphi)rd\varphi dx \quad (8)$$

$$C_M=\frac{1}{\pi r_r^3}\int_0^l\int_0^\pi C_p(x,\varphi)\cos(\varphi)xrd\varphi dx \quad (9)$$

At low supersonic Mach number, the pressure over expands on a blunt nose tip as it proceeds around the blunt portion from the stagnation point to the given portion of the noose. HTVD is used to cover this overexpansion near its maximum acceptable slope and allow the pressure to expand around the surface. The MNT is started at the stagnation point and allowed to expand until the pressure coefficient of the MNT and HTVD were equal. Upstream of the match point, (C_p MNT=C_p HTVD) MNT is used in force and moment calculation and HTVD downstream.

2.1.4 Moderate/high supersonic (1.8<M<6)

Second Order Shock expansion Theory (SOSET) plus Modified Newtonian theory (MNT) method. Combination between SOSET and MNT showed good agreement with experimental data for M > 2.3 but only fair result agreement for lower supersonic Mach number. Dejarnette et al. [8] developed an empirical equation for the Mach number to match MNT to SOSET. This approach gave improved result particularly at low supersonic Mach number. Pressure coefficient on a blunt nose derived by Dejarnette is:

$$C_p=C_{pMNT}-\delta\cos^m\delta[\cos\delta-\cos(\delta)_m] \quad (10)$$

Where m=2.78, $(\delta)_m$=25.95 and

$$\delta=\frac{\left(C_{p0}\sin 2\delta\right)_{MNT}+1.5\left[\left(C_p+\frac{2}{\gamma M_\infty^2}\right)\left(C_{p0}\sin 2\delta\right)\right]_{MNT}^{1/2}}{\sin(\delta)_m\cos^m(\delta)_m}$$

From Ref. 8, constant value of a match point θ_m = 25.95 degree and pressure at this point given by:

$$\frac{p_1}{p_\infty}=1+\frac{\gamma_\infty M_\infty^2}{2}C_{p1} \quad (11)$$

C_{p1} is calculated from equation 9. The total pressure is:

$$P_0=\frac{1}{2}\rho_\infty V_\infty^2C_{p0}+p_\infty \quad (12)$$

The local Mach number can be found:

$$M_1=\left\{\left(\frac{2}{\gamma-1}\right)\left[\left(\frac{p_0}{p_1}\right)^{\frac{\gamma-1}{\gamma}}-1\right]\right\}^{\frac{1}{2}} \quad (13)$$

Figure 1 [3] is an example application of combination of SOSET and MNT on blunt cone at Mach number = 2.96. Note the excellent agreement of theoretical and experimental pressures along the surface.

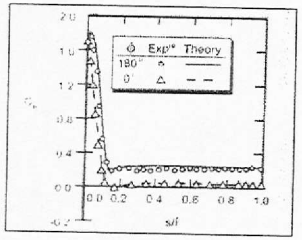

Figure 1 Pressure distribution on a blunted cone R_N/R_B = 0.35, α = 8° and θ_c = 11.5°

Bonney method. Based on Bonney [9] the equation for body wave drag coefficient is:

$$C_{D0B\,Wave}=(1.59+1.83/M^2)\{\tan^{-1}[0.5/(l_n/d)]\}^{1.69} \quad ; M>1 \quad (14)$$

Even though this method is limited for M > 1, it is still useful because most of the rocket model flying at M > 1.

Wing wave drag (wing/fin alone)

Shock waves are typically associated to supersonic flow. Similar phenomenon can be seen at much lower speeds at areas on the aircraft where the Bernoulli effect accelerates local airflow to supersonic speeds over curved areas. This effect is typically observed at speeds of about M = 0.8 so wing wave drag can be neglected for subsonic Mach number.

2.2.1 Transonic flow 0.8≤M<1.2

Empirical method [3]. For wave drag prediction of wing alone, data from table 2 can be used with the assumption that the drag varies linearly between its value calculated at M = 1.1 from linear theory and a value of 0 at M = 0.9.

2.2.2 Low supersonic flow 1.2<M<1.8

Linear theory plus Modified Newton Theory (MNT) method [10] The pressure coefficient at any point on the wing surface is:

$$C_p = -2\Phi_x(x, y, 0) \qquad (15)$$

where Φ_x is the perturbation velocity at any given point and is dependent on location of the point with respect to line of sources and sinks that generates the wing leading edge or other discontinuity.

The pressure coefficient can be calculated at a given number of spanwise and chordwise locations [3]. The drag coefficient of given airfoil section at the spanwise station y = y_A is:

$$c_d = \frac{2}{c(y_A)} \int_0^{c(y_A)} C_p(x, y_A)\, w(x, y_A)\, dx \qquad (16)$$

The total drag coefficient for one fine of semi span b/2 is:

$$C_D = \frac{1}{S_w} \int_0^{b/2} c_d\, c(y)\, dy \qquad (17)$$

where $S_w = \frac{b}{2}(c_r + c_t)$.

For cruciform fins, the total drag coefficient is:

$$C_D = \frac{4}{S_w} \int_0^{b/2} c_d\, c(y)\, dy \qquad (18)$$

If it is desired to base the drag coefficient on the body cross-sectional area it must be multiplied by S_W/S_{ref}.

2.2.3 Moderate/high supersonic (1.8<M<6)

Second order Shock expansion Theory (SOSET) plus Modified Newtonian theory (MNT) along strips method. [11] The wave drags prediction based on MNT and given as:

$$C_{D_{wave}} = n_w \left[\frac{2}{\gamma M^2_{NLE}}\right] \left\{ \left[\frac{(\gamma+1)M^2_{NLE}}{2}\right]^{\frac{\gamma}{\gamma-1}} \left[\frac{\gamma+1}{2\gamma M^2_{NLE} - (\gamma-1)}\right]^{\frac{1}{\gamma-1}} - 1 \right\} \sin^2\delta_{LE} \cos\Lambda_{LE} \frac{S_{LE}}{S_{ref}} \qquad (19)$$

Newtonian theory is modified by calculating the pressure across the normal shock as a function of Mach number. It is noted that a thin wing with a small leading edge section angle has smaller wave drag than that of a wing with a blunt leading edge. Also, leading edge sweep reduces the effect of Mach number by the factor of cos Λ_{LE}, maintaining a subsonic leading edge until M cos Λ_{LE} = 1. A third potential contributor to wing drag is the base drag due to flow separation on the aft surface of the wing. However, for a thin wing with a sharp trailing edge, there is negligible flow separation and negligible base drag.

2.3 Skin frictions drag (body and wing/fin alone)

Van Driest II method. Most of the tactical rockets have a majority of flow consisting of about 10 to 20% laminar flow followed by 80 to 90% turbulent flow. For turbulent boundary layer skin friction coefficient, C_{fT} is calculated by using Van Driest method. [12] It assumes zero pressure gradient and Prandtl number equal to 1. The solution can be found by numerically solving the equation below:

$$\frac{0.2424}{A(C_{fT})^{1/2}}\left(\frac{T_w}{T_\infty}\right)^{1/2}(\sin^{-1}C_1 + \sin^{-1}C_2) = \log_{10}(R_{N\infty}C_{fT}) - \left(\frac{1+2n}{2}\right)\log_{10}\left(\frac{T_w}{T_\infty}\right) \qquad (20)$$

where,

$$C_1 = \frac{2A^2 - B}{(B^2 + 4A^2)^{\frac{1}{2}}} \; ; \; C_2 = \frac{B}{(B^2 + 4A^2)^{\frac{1}{2}}}$$

$$A = \left[\frac{(\gamma-1)M_\infty^2}{2T_w/T_\infty}\right]^{1/2} \; ; \; B = \frac{1 + (\gamma-1)/2M_\infty^2}{T_w/T_\infty} - 1$$

$$R_{N\infty} = \frac{\rho_\infty V_\infty \ell}{\mu_\infty} \qquad (21)$$

$$\frac{T_w}{T_\infty} = 1 + 0.9\frac{\gamma-1}{2}M_\infty^2 \qquad (22)$$

$$\frac{\mu}{\mu_\infty} = \left[\frac{T_\infty}{T_s}\right]^n \qquad (23)$$

where n = power in the power viscosity law, 0.76 for air.

For laminar flow, skin friction drag, C_{fl}

$$C_{fl} = \frac{1}{\mathrm{Re}_x}\left[1.328 - 0.0236M_\infty - 0.00338M_\infty^2 + 0.000349M_\infty^3 - 8.54\times10^{-6}M_\infty^4\right] \qquad (23)$$

Hence the skin friction drag is simply the mean skin friction coefficient multiplied by the ratio of wetted area to reference area.

$$C_{DF} = C_{fl}\frac{(A_{wetted})_l}{A_{ref}} + C_{fT}\frac{(A_{wetted})_T}{A_{ref}} \qquad (24)$$

For body alone: equation 24 must be multiplied by 1.14 and for wing surfaces no factor is needed.

friction given by equation 25 and 26, but this method is determining the friction drag for turbulent boundary layer only.

$$C_{D0\,Body,\,Friction} = 0.053\,(l/d)\,[M/(q\,l)]^{0.2} \qquad (25)$$

$$C_{D0,Wing,Friction} = n_W \{ 0.0133 [M / (q\, c_{mac})]^{0.2} \} (2 S_W / S_{Ref}) \quad (26)$$

2.4 Body base drag (body alone)

Improved empirical method. Base drag arises at the base of the rocket or a rear of the blunt trailing edge of a fin due to separation flow. Equations 27-33 below show the summary of the empirical method of the Ref. 13.

A. Body Alone

$$(C_{P_B})_{NF,\alpha} = (C_{P_B})_{NF,\alpha=0}[1+0.01F_1] \quad (27)$$

where F_1 = Body Alone, α = Effects

B. Body with Tail Fins

1. Deflection and Thickness Effects

$$(C_{P_B})_{\alpha,\delta,F,x/c=0} = (1+0.01F_2)(C_{P_B})_{NF,\alpha=0} + 0.01F_3(t/d) \quad (28)$$

where $F_2 = |\alpha+\delta|$ effects

F_3 = Additional effects due to thickness

2. Fin Location Effects when $x/c \neq 0$

$$(C_{P_B})_{\alpha,\delta,F,x/c} = (C_{P_B})_{NF,\alpha} + 0.01(\Delta C_{P_B})_{\alpha,\delta,F,x/c=0} \quad (29)$$

C. *Power On*

ΔC_{P_B} Due to Power-On by Modified Brazzel Method Added to Above Values of C_{P_B}

D. *Boat tail*

$$C_{A_B} = -C_{P_B}(d_B/d_{ref})^3 \quad (30)$$

E. *Flare*

$$C_{A_B} = -C_{P_B}(d_B/d_{ref})^2 \quad (31)$$

For Mach number above 4.3 an empirical equation for base pressure coefficient, C_{PB} is:

$$C_{PB} = \frac{0.18}{M_\infty^2} + \frac{0.266}{M_\infty} - 0.018 \qquad ; 4.3 < M_\infty < 16 \quad (32)$$

$$C_{PB} = 0 \qquad ; M_\infty > 16$$

The power on base pressure coefficient is

$$(C_{PB})_{ON} = \frac{2}{\gamma M_\infty^2}(p_b / p_\infty - 1) \quad (33)$$

Simplified method. Body base drag can be a major contributor to the total drag during coasting flight, due to the low pressure in the base. For a high fineness nose, the base drag can be larger than the wave drag. It can be divided into two types [11]: Body base drag for coasting flight and powered flight. During powered flight the base drag is reduced by the factor $(1 - A_e/S_{Ref})$.
The base drag equation for coasting flight at supersonic Mach number ($M>1$) is

$$C_{DB,Coast} = 0.25/M \quad (34a)$$

$$C_{D0\,Base,Powered} = (1 - A_e / S_{Ref}) (0.25 / M) \quad (34b)$$

For coasting flight at subsonic Mach number,

$$C_{DB,Coast} = 0.12 + 0.13M^2 \quad (35a)$$

$$C_{D0\,Base,Powered} = (1 - A_e / S_{Ref}) *$$

$$(0.12 + 0.13\, M^2) \quad (35b)$$

If the nozzle exit area is nearly as large as the rocket base area, the base drag may be negligible during powered flight

3. CONCLUSION

Selection of rocket's aerodynamic drags prediction method usually based on the accuracy of the method, ease to use and simple. This paper is only the summary of the methods in order to give general understanding for rocket aerodynamic mainly rocket drag. The objective of this paper is to predict the aerodynamics drag at a various Mach number to optimize the rocket range and it also can be used as an initial guidelines and a quick reference for approximate aerodynamic drag method in general.

NUMENCLATURE

M	= Mach number
D	= Drag force (N)
T	= Temperature (° R)
q	= Dynamics pressure
p	= Pressure (N)
C_D	= Drag force coefficient
S	= Area of the section
d	= Diameter
R, r	= radius
l	= Length
C_P	= Pressure coefficient
C_A	= Axial Force coefficient
C_N	= Normal force coefficient
C_M	= Moment pitching coefficient
γ	= Specific heat ratio
x, r, φ	= Polar or cylindrical coordinate with x along body axis, r along radius and φ around body
θ	= Reynolds Number
A_e	= Nozzle exit area
M	= Mach angle, $=\sin^{-1}(1/M)$
$C_{D0\,Body,Wave}$	= body zero-lift wave drag coefficient
$C_{D0\,Base}$	= body base drag coefficient,

$C_{D0\ Body,\ Friction}$	= body skin friction drag coefficient
$C_{D0\ Body}$	= body zero-lift drag coefficient

Subscript

Ref	= Reference
vis	= Viscous
∞	= Free stream
0	= Stagnation point
N, n	= Rocket's noose
B, b	= Rocket's body

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