NOZZLE PERFORMANCE PREDICTIONS USING THE TDK 97 CODE

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Nomenclature

A	Cross Sectional Area
C*	Characteristic velocity, $P_c A^*/\dot{m}$
Isp	Specific Impulse, $F/\dot{m}$
F	Thrust
LRE	Liquid Propellant Rocket Engine
M	Mach Number
$\dot{m}$	Mass flow rate
NTO	Nitrogen Tetroxide
p	Pressure
r	nozzle radius
Re	Reynolds' Number
RP-1	Kerosene Based Fuel
UDMH	Unsymmetrical dimethyl-hydrazine
y	Distance from the nozzle wall

Greek

α	Nozzle wall angle
δ	Boundary layer thickness
δ^*	Boundary layer displacement thickness
γ	Ratio of specific heats, Cp/Cv
ε	Expansion ratio (A_e/A^*)
η	Efficiency
θ	Boundary layer momentum thickness also nozzle half angle

subscripts

a	ambient
aw	adiabatic wall
c	chamber
div	divergence
e	nozzle exit
eq	chemical equilibrium
f	finite contraction ratio
froz	chemically frozen
i	i th zone or striation
kin	finite rate chemistry
th	theoretical or ideal
xz	interzonal
∞	infinite contraction ratio
0	stagnation, equation (1)

superscripts

*	refers to nozzle throat plane

Abstract

The Two-Dimensional Kinetics (TDK) computer program is a primary tool in applying the JANNAF liquid rocket thrust chamber performance prediction methodology. Over the past decade work has been completed which extends the applicability of TDK to high expansion ratio space engines, scramjet engines, plug nozzles, and to engines that include tangential injection of gas generator products into the exhaust nozzle and transpiration cooling of the nozzle wall.

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The code can now be applied to analyze Orbit Transfer Vehicle (OTV) engine designs, and also designs for Space Transportation Booster Engines (STBE's), which feature dual fuel concepts. Many improvements have been made to the code, especially with respect to treatment of the wall boundary layer. For example, a new Mass Addition Boundary Layer (MABL) module has been added to the code. The MABL module allows secondary exhaust products to be injected tangential to the primary flow. The products are then mixed along the shear layer interface and allowed to react chemically. A generalized chemistry capability is included. Provision has also been made to treat the effects of wall surface roughness, transpiration cooling, radiation cooled walls, and laminar-turbulent transition. The finite rate chemistry of TDK has been improved in several respects. The method of characteristics solution has been modified in that complete numerical stability is achieved for very large engines, e.g., engines operated with LOX/hydrocarbon propellants combusted at pressures in excess of 200 atmospheres. The generalized chemistry has been extended to include global as well as elementary finite rate reactions. Reactions of the global type are useful in characterizing the initial steps of hydrocarbon decomposition. Heterogeneous reactions have also been provided, and several different types of rate expressions can be specified.

Introduction

The topic of assessing thrust chamber performance is not new and there are many excellent sources of information on this topic. In the United States, Refs. 1 and 2 address all of the issues discussed here and embody the JANNAF performance prediction methodology. The books of Sutton[3] and Zucrow and Hoffman[4] cover many of the topics in detail.

Liquid Propellant Rocket Engines, LRE's, are devices that convert the latent energy of the propellants into sensible heat in the combustion chamber and then convert it again into kinetic energy in the nozzle. In order to make comparisons between different engine and system designs, it is necessary that we be able to assess the performance of the LRE. The thrust chamber of a LRE produces the measurable output of a liquid propellant powered rocket, i. e., thrust, and hence the assessment of its performance is of great importance in evaluating the overall performance of the entire system. Because of the tremendous energy flow in LRE's, these engines are characterized by small performance losses due to heat loss, friction, vaporization and mixing inefficiencies, etc. However, even small losses have a large impact on delivered payload or range of the system and are therefore important.

In order to assess the performance of a system, one must establish a yard stick or figure of merit that characterizes the system. The figure of merit must also be a measurable quantity. The most used such quantity for LRE thrust chambers is the specific impulse, Isp. The specific impulse is defined as the engine thrust divided by the mass flow rate of the propellants and thus tells us how effectively the thrust chamber converts propellant into thrust. Both the specific impulses delivered to vacuum and to ambient pressure conditions are commonly used.

Maximum performance of a thrust chamber, sometimes called ideal or theoretical performance, is achieved if the propellants entering the thrust chamber react completely and chemical equilibrium is maintained throughout the expansion process. Additionally, the flow should be isentropic and one dimensional. Under these ideal conditions, the thrust chamber performance is dependent on the physical, chemical, and thermodynamic properties of the propellants and their combustion products, and on the operating conditions of the engine, i.e., propellant mixture ratio, O/F, chamber pressure, p_c, expansion ratio, ε, and ambient pressure, p_a. Hence, for the ideal thrust chamber, we have removed the real-world design parameters such as nozzle geometry, size, injector element design, engine coolant configuration, baffles, etc.

Thus the theoretical maximum performance is defined as an isentropic one dimensional flow in chemical equilibrium (often called shifting equilibrium) at the thrust chamber O/F and chamber pressure (infinite or finite contraction ratio). In TDK, the theoretical Isp is calculated using the ODE module which was adapted from the CEA/CET codes of Gordon and McBride.[5,6]

The JANNAF performance prediction methodology is based on estimates of the magnitude and interactions of various loss mechanism which occur in an LRE. The losses considered by the JANNAF methodology and the estimates of the interactions are given in Table 1 which was taken from Ref. 1.

Table 1. Interaction of Physical Phenomena With Performance Loss Calculations

Performance Losses / Phenomena	Divergence Loss	Boundary Layer Loss	Finite Rate Kinetics Loss	O/F Mal-distribution Loss	Energy Release Loss
Non- 1-D Flow	-	1 st Order (>.2%)	2 nd Order (<.2%)	Not Imp.	Not Imp.
Viscous and Heat Transfer	Not Imp.	-	Not Imp.	Not Imp.	Not Imp.
Finite Rate Chemistry	Not Imp.	2 nd Order (<.2%)	-	Not Imp.	Not Imp.
Non-Uniform Mixture Ratio	Not Imp.	1 st Order (>.2%)	1 st Order (>.2%)	-	Not Imp.
Incomplete Energy Release	Not Imp.	2 nd Order (<.2%)	1 st Order (>.2%)	Not Imp.	-

1 st Order (>.2%) = Primary Importance (Could Be >0.2% on Isp); *2 nd Order (<.2%)*= Secondary Importance (Probably <0.2% on Isp, *Not Imp.* = Generally Not Important

The TDK[7] code treats all of the losses above except the energy release loss. The TDK code consists of several modules which computes the Isp for a variety of input mixture ratios and enthalpies. The code uses a finite rate kinetics MOC solver to compute the core flow and a finite rate kinetics finite difference boundary layer module employing a Cebeci-Smith[8] eddy viscosity turbulence model to compute the boundary layer loss. The O/F mal-distribution or macromixing loss is treated by inputting to the code the O/F and energy content of each striation considered. Gas turbine exhaust dumps can be treated as being injected into either the boundary layer or the core flow. Both cold wall, radiation cooled, transpiration cooled, and adiabatic wall heat transfer treatments are allowed. Furthermore, the solutions of the core flow and boundary layer can be iterated by displacing the potential wall either inward or outward by the boundary layer displacement thickness.

The method of approach used by TDK and the JANNAF methodology is to couple the divergence, finite rate chemistry, and non-uniform mixture ratio losses into one calculation which is performed by the TDK MOC module and a decrement, Δ, for the boundary layer loss which is calculated by the Mass Addition Boundary Layer (MABL) module. That is

$$Isp_D = Isp_{TDK-MOC} - \Delta Isp_{MABL} \tag{1}$$

The MABL module is an extension of the work by Levine[9].

The TDK Model

This section describes the modeling approach used in TDK. Starting first with the calculation of ideal performance, we will then cover the various other losses.

Ideal or Theoretical Performance

As stated above, the ODE module is used to calculate the theoretical performance of the propellants at a given chamber pressure, mixture ratio and propellant energy content The usual assumptions made are that the composition and enthalpy (heat of formation plus sensible heat) of the propellants are known. The enthalpy of the propellants in the tank is often used if known, otherwise the enthalpy at the normal boiling point (NBP) is a good choice. The chamber pressure is assumed and an enthalpy-pressure (HP) solution is found. The products of combustion are then expanded to different pressures using the entropy of the products (PS) in the chamber to close the set. From the conservation of mass and energy, the area ratios and velocities can be found at each solution point. The ODE module is used to calculate the chamber conditions to start the kinetics calculation for each O/F specified.

Two Dimensional Flow Field Model

As stated previously, TDK uses an MOC solution to compute the nozzle performance considering divergence, finite rate kinetics, and O/F non-uniformities. This method is used to solve the Euler equations which include the diabatic heat release terms from the chemistry. Solutions to these equations while isoenergetic along streamlines, allow for shock waves and variations in the total enthalpy of the flow.

Because the steady state Euler equations only have real roots for supersonic flow, a supersonic start line must be constructed. Gas and species properties are computed by the ODK module which computes the one dimensional flow equations with finite rate kinetics from the chamber to the throat region using a pressure define stream tube. The series solution by Sauer[10] as modified by Nickerson[11] gives a good estimate of the nozzle discharge coefficient and transonic start line for the method of characteristics used in TDK.

The divergence loss is conceptually the simplest of all the losses. The velocity vectors of the gases exiting the nozzle are not necessarily aligned with the axis of the nozzle or vehicle. The result of the misalignment is that not all of the kinetic energy of the flow results in axial thrust.

When the highly energetic propellants are burned in the combustion chamber, the resulting high temperatures cause many of the normally stable molecules to dissociate. During the subsequent expansion in the nozzle, the kinetic rate process tends to re-combine these molecules making sensible heat available to further drive the expansion. Most notably it is the recombination of hydrogen molecules to form H_2 and the formation of CO_2 from CO and O which releases the bulk of the energy. It is the short residence time in the nozzle coupled with rapidly decreasing pressure and temperature which does not allow the flow to stay in chemical equilibrium (maximum heat release).

The calculation of finite rate chemically reacting flows require that for each species the net production rate, ω_i, be calculated. These values, together with the species thermodynamic data, are sufficient to allow non-equilibrium properties to be included in the fluid flow equations. The finite rate chemical kinetics loss is computed using the law of mass action and Arrhenius form rate data. First and second order reactions along with unidirectional reactions are allowed.

For most propellant systems, the reaction rate mechanisms and rate data are reasonably well understood. The rate data are usually known within an order of magnitude which is adequate for determining the finite rate kinetics loss. There are several ways in which this loss is expressed. The two most common are the ratio of specific impulse with and without the loss, η_{kin}, (used with the ODK module of TDK) and the fraction of the difference between equilibrium and frozen flow, η'_{kin}.

$$\eta_{kin} = Isp_{kin} / Isp_{eq} \tag{2a}$$

$$\eta'_{kin} = (Isp_{kin} - Isp_{froz})/(Isp_{eq} - Isp_{froz}) \tag{2b}$$

The loss is mainly a function of the propellant system, chamber pressure, and residence time in the nozzle. High pressure systems tend to have smaller losses due to the large number of molecular collisions. The nozzle length scale is also an important parameter in that it sets the residence time in the nozzle. While high area ratio nozzles tend to have larger losses, this effect is less important than the other factors discussed. The table given below shows some typical calculated values for the kinetic loss efficiencies.

Table 2. Kinetic Loss Ratios

Engine	Propellants	η'_{kin}
F-1	LOX/RP-1	.98
Atlas Booster	LOX/RP-1	.9
Atlas Sustainer	LOX/RP-1	.9
TR201	NTO/A50	.5
R-4D	NTO/MMH	.3
Titan III (Stage I)	NTO/A50	.75

As can be seen from the above table, amine fueled (e.g. hydrazine, UDMH, MMH) engines usually exhibit low kinetic efficiencies.

For most systems the finite rate kinetics loss is less than 2%. However, low pressure amine and fluorinated systems can have higher losses. In TDK, either a one dimensional (ODK) or two dimensional model (MOC) can be used to estimate the kinetics loss. Arrhenius rates are known within an order of magnitude for most of the important reactions in chemical systems of current interest. The difference in calculated kinetics loss between one and two-dimensional solutions is usually very small. However, in order to handle striations in the flow, TDK uses the two dimensional solver with finite rate kinetics.

The rates chosen for these calculations have an obvious impact on the magnitude of the loss. The rates shown in Table 3 are recommended for CHON systems in the TDK code documentation[7]. Recommended third body efficiencies for various species are shown in Table 4. Perturbing the H+H and CO+O recombination rates by factor of 30 downward has the effect on kinetic efficiency shown in Table 5 for an engine with 100:1 area ratio, NTO/A50 propellants at a stoichiometric mixture ratio, and a chamber pressure of seven atmospheres. As can be seen from the table, the changes in performance are minimal.

Table 3. Reaction Rate Data for the CHON System:
(for chemical reactions between species, CO, CO_2, H, H_2, H_2O, N, NO, N_2, O, OH, and O in initial equilibrium expanding through an adiabatic nozzle)

Reactions	A	N	B	Meas. M	Reference
$H+H+M=H_2+M$	6.4E17	1.0	0.0	Ar	BAULCH 72 (A) 30U
$H+OH+M=H_2O+M$	8.4E21	2.0	0.0	Ar	BAULCH 72 (A) 10U
$O+O+M=O_2+M$	1.9E13	0.0	-1.79	Ar	BAULCH 76 (A) 10U
N+O+M=NO+M	6.4E16	0.5	0.0	N2	BAULCH 73 (C) 10U
$N+N+M=N_2+M$	3.0E14	0.0	-.99	N2	BAULCH 73 (C) 10U
$CO+O+M=CO_2+M$	1.0E14	0.0	0.0	Ar	BAULCH 76 (B) 30U
O+H+M=OH+M	3.62E18	1.0	0.0	Ar	JENSEN 78 (B) 30U
$O_2+H=O+OH$	2.2E14	0.0	16.8		BAULCH 72 (A) 1.5U
$H_2+O=H+OH$	1.8E10	-1.	8.9		BAULCH 72 (A) 1.5U
$H_2+OH=H_2O+H$	2.2E13	0.0	5.15		BAULCH 72 (A) 2 U
$OH+OH=H_2O+O$	6.3E12	0.0	1.0		BAULCH 72 (A) 3 U
$CO+OH=CO_2+H$	1.5E7	-1.3	-.765		BAULCH 74 (A) 3 U
$N_2+O=NO+N$	7.6E13	0.0	75.5		BAULCH 73 (C) 3 U
O_2+NO+O	6.4E9	-1.0	6.25		BAULCH 73 (C) 2 U
CO+O=CO	2.5E6	0.0	3.18		BAULCH 76 (B) 2 U
$CO_2+O=CO+O$	1.7E13	0.0	52.7		BAULCH 76 (B) 3 U

$k = A\,T^{-N}\ \exp(-1000B/RT)$; in units of cc, °K, mole, sec.

Table 4. Third Body Recombination Efficiency Ratio (CHON System):
(as recommended by Kushida[12])

Species	H+H	H+OH	O+O	N+O	N+N	CO+O	O+H
Ar	1.	1.	1.	.8	1.	1.	1.
CO	1.5	3.	4.	1.	1.	1.	4.
CO_2	6.4	4.	8.	3.	2.	5.	5.
H	25.	12.5	12.5	10.	10.	1.	12.5
H_2	4.	5.	5.	2.	2.	1.	5.
H_2O	10.	17.	5.	7.	3.	1.	5.
N	1.	1.	10.	10.	10.	1.	1.
NO	1.5	3.	4.	1.	1.	1.	4.
N_2	1.5	3.	4.	1.	1.	2.	4.
O	25.	12.5	12.5	10.	10.	1.	12.5
OH	25.	12.5	12.5	10.	10.	1.	12.5
O_2	1.5	6.	11.	1.	1.	25.	5.

Table 5. Variation In Kinetic Efficiency With Rate Data

Reaction	η'_{kin}	η_{kin}
Reference/No Changes	.552	.9513
$H+H+M=H_2+M$	.546	.9507
$CO+O+M=CO_2+M$	.551	.9512
Change Both Rates	.545	.9505

Another consideration is the starting point for the expansion. If the species in the combustion chamber do not start out in a state near chemical equilibrium, then there is the potential that they will not approach equilibrium within the nozzle. The non-equilibrium starting condition problem is especially important at off optimum mixture ratios.

Interzonal variations in mixture ratio are caused by decisions made in the design of the thrust chamber. The most common cause of interzonal striations is the use of a fuel (or oxidizer) film to keep the chamber walls from exceeding their maximum design temperature. Other striations can be caused by the presence of baffles used to suppress acoustic waves.

The design values of mass flow for the fuel and oxidizer are used for initial studies. Cold flow data can supply updated values once testing has begun. Except in rare instances, striations can only be inferred from hot flow heat transfer data.

In the TDK, the data are used to generate striation profiles which are then run though the MOC module. The boundary between the striations is a slip line or contact discontinuity. The effect of striations on the boundary layer edge condition can be very significant. Great care has been taken not to include this loss more than once.

Thus, in the full up TDK calculation, the divergence, finite rate kinetics, and macromixing losses are coupled together in the two dimension finite rate MOC module.

Boundary Layer Loss

Propulsive LRE's are generally characterized by high Reynolds number flow. The table below lists the Reynolds numbers based on throat conditions for a variety of engines. Because the mass flux is highest in the throat region, the throat Reynolds number is almost always the largest encountered during the nozzle expansion. Other characteristics of importance in LRE's which affect boundary layers are the methods of wall cooling. Since the enthalpy of the combustion products is very high, the chamber and nozzle walls need to be protected. Some standard ways of protecting the walls include regenerative cooling, barrier or film cooling, radiation cooling, ablative walls, and slot injection or transpiration cooling. A high Reynolds number means that the viscous layer next to the wall is very thin, which in turn indicates that the classical thin shear or boundary layer assumptions are valid. Hence, except for the smallest engines, the core flow in the engine can be treated as inviscid and the solution of the wall shear layer can be uncoupled from the core flow. The true singular perturbation nature of the boundary layer equations becomes quite apparent in rocket engine flows since the outer or core flow is not uniform and can vary significantly in the radial direction over the distance of a boundary layer thickness. In addition, when film cooling is used in the engine, there is a significant total enthalpy gradient near the wall and hence the outer flow can be highly rotational. The standard simplistic ways of looking at the boundary layer thickness can be very misleading and questions as to how much mass flow is in the boundary layer have limited meaning.

Table 6. Nozzle Characteristics For Various Engines

Engine	Thrust (10^3N)	p_c (bars)	r* (mm)	ε	Re_{r^*}
Hughes 5 lbf	.11	1.72	2.37	296.6:1	1.10×10^4
NASA/LeRC Hi-E	2.40	24.82	12.7	1025:1	1.73×10^5
XLR-134	2.28	35.16	10.06	767.9:1	1.80×10^5
STS/RCS	3.84	10.34	25.93	28.46:1	1.75×10^5
Adv. Space Engine	100.67	157.68	31.85	400.7:1	2.20×10^6
RL 10	60.05	27.19	65.28	205.03:1	1.29×10^6
RD-170	7915.73	244.65	117.75	36.9	1.62×10^7
SSME	2062.45	226.49	130.88	77.5:1	1.18×10^7
F1	7786.55	68.4	444.5	15.76:1	1.81×10^7

Since solution procedures for the boundary layer equations are well established, the only real questions are what physical phenomena need to be modeled. Smaller engines tend to have laminar boundary layers while the larger engines are almost always turbulent. One rule of thumb is that engines with less than 4,500 kgf (10,000 lbf) thrust are laminar. A slightly more appealing transition criteria is that transition occurs when the Reynolds Number, base on the boundary layer momentum thickness, Re_θ, exceeds 360. The MABL boundary layer module of TDK uses the algebraic eddy viscosity model of Cebeci-Smith. Coats et al[13] have estimated that the maximum calculated variation in boundary layer loss result is approximately 25% when a κ-ε turbulence model is compared to the Cebeci-Smith model. Since the boundary loss is seldom more than 4% of the total performance, the variation of calculated loss with turbulence model will be less that 1% of the total performance. Without high quality experimental data to validate turbulence models for rocket engine flows, there is no way of knowing which of the available turbulence models should be used.

Other questions arise in the use of TDK as to which chemistry model should be used in the boundary layer calculation. For most simple flows, i.e., single O/F core flow, almost any chemistry model will give results within the known accuracy range of the boundary layer equations. However, a consistent chemistry model should be used for the core flow and the boundary layer. If heat transfer results are required in addition to performance losses, then the choice of chemistry model can be quite important. For example, the difference in adiabatic wall temperature at the nozzle exit plane for the Vulcain engine between the assumption of finite rate kinetics and chemical equilibrium and is 350 °K hotter, a very non-trivial difference if you are determining the cooling requirements of the engine! Another consideration in selecting the boundary layer chemistry model is the need to predict what happens to turbine exhaust gases that are injected into the engine downstream of the throat. These injected gases have a pronounced effect on the boundary layer profiles as shown in Figure 1 and can lead to either endothermic or exothermic reactions. Transpiration cooling modeling requirements will also have an impact on the chemistry model selection.

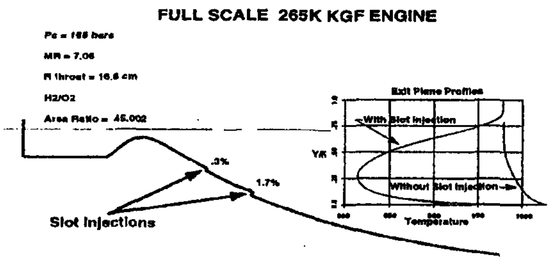

Figure 1. Boundary Layer Profile
With Tangential Slot Injection

One curiosity of the JANNAF procedure is the method of computing the boundary layer loss. The standard JANNAF equation for the boundary layer loss is

$$\Delta Isp_{BL} = 2\pi r \rho_e u_e^2 \theta \cos(\alpha)\left[1 - \frac{\delta^*}{\theta} P_e / \rho_e u_e^2\right] / \dot{m} \qquad (8)$$

which is a combination of both the inner and outer solutions applied all at one time. Kehtarnavaz et al[14] have extended the derivation of Eq. (8) to thick boundary layers.

Energy can also escape the thrust chamber in the form of radiation. The foremost methods are from the combustion chamber walls, the nozzle, and from the hot gases. The first two losses are coupled with the boundary layer loss while the second is generally small. For radiation cooled nozzles the performance loss is treated as a boundary condition for the MABL calculation.

There is always a concern about the coupling between the core flow and the boundary layer solution, especially for small or large area ratio nozzles. Kushida et al[15] have reported very good agreement between predictions and measured data when using the TDK MOC/boundary layer method for a very small high area ratio thruster (Hughes 5 lbf engine of Table 6). The computed boundary layer thickness for that nozzle was 28% of the radius at the nozzle exit plane!

Mixing Losses Treated By TDK

We have already mentioned that the engine wall is sometimes cooled by injecting a fuel (or oxidizer) film spray on the wall. The lower (or higher) O/F in this region reduces the flame temperature and thus the heat transfer rate. Because these propellants do not combust in a way that releases the maximum amount of heat, there is a loss associated with this process when compared to our theoretical performance at the overall engine mixture ratio. This loss is referred to as a coarse mixing, interzonal mixing, or macromixing loss and is treated by TDK using the methods describe above.

Sometimes all of the above losses are lumped together and referred to as combustion efficiency or energy release losses. The most direct measurement that we have of these losses is the measured C* efficiency of the engine.

Mixing Losses Not Treated By TDK

Liquid rocket engines do not always vaporize all of the propellants within the combustion chamber. With many engines using hydrocarbon fuels, the fuel tends to vaporize much slower than the oxidizer (fuel controlled burning). This slow vaporization can cause a large shift between the injected O/F and the effective gas phase O/F at the exit of the combustion chamber. Engine designers often tradeoff combustion chamber length and ease of injector fabrication for vaporization efficiency.

The major problem with modeling intrazonal or micromixing loss is that it can not be measured directly in either a rocket engine or a reasonable simulation device. The micromixing losses are always inferred by first subtracting out other losses such as finite rate kinetics, vaporization, and macromixing losses. Both theoretical and empirical micromixing models exist. On the empirical side, C* correlations based on similar engines are used to estimate the total energy release efficiency loss.

Since TDK does not treat these losses, we recommend the use of engine specific empiricisms to estimate the total energy release efficiency loss. From the total loss, subtract out the vaporization and macromixing losses, and then adjust the input enthalpy to match the measured or estimated performance.

Results

The following sections gives some examples of the various capabilities of the TDK code. Included are calculated results for F-1 engine which shows the effects of a multiple louvered nozzle, a dual bell nozzle, a plug nozzle, a scramjet nozzle, and an engine employing transpiration cooling of the nozzle wall.

F-1 Engine

The F-1 Saturn V Booster Engine is the largest liquid rocket engine ever built by the United States. It produced approximately 1.5 million pounds of thrust at sea level. A cluster of five engines were used to power the first stage of the Saturn V vehicle, and produced a sea level thrust of 7.5 million pounds. A picture of the F-1 is shown in below. The engine operates with a run duration of 164 seconds and a design life of 2250 seconds (20 starts).

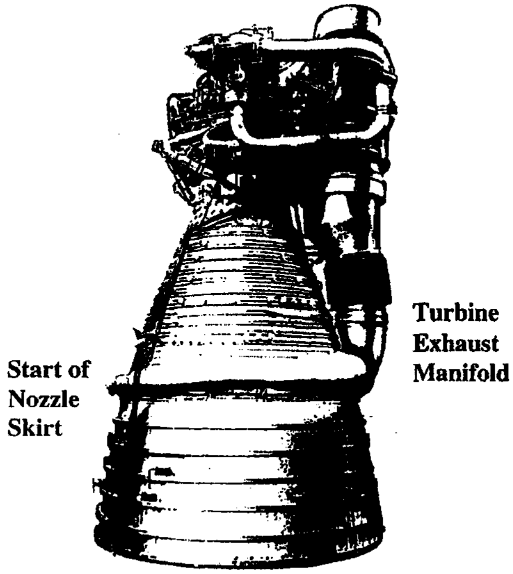

Figure 2. F-1 Saturn V Booster Engine

The F-1 engine utilizes kerosene (RP-1) as the fuel and liquid oxygen (LOX) as the oxidizer. A single regeneratively cooled thrust chamber extends to the 10:1 expansion ratio. A nozzle extension is attached to the thrust chamber and further expands the primary flow to an expansion ratio of 16:1. The nozzle extension is cooled by injection of the turbine exhaust manifold gases. These gases are injected along the extension nozzle inner wall through slots formed by 23 rows of overlapping shingles. The injection is tangential to the primary thruster flow.

The presence of the turbine exhaust dumps through a series of shingles makes this engine an excellent choice to demonstrate the tangential slot injection capability of TDK. The injection of mass is handled in the MABL module with the restriction that the pressure of the injectant must equal the core flow pressure at the point of injection. The following two figures show the boundary layer momentum thickness, θ, as a function of axial distance. Both figures clearly show the simulation of the shingles. The momentum thickness grows because of the increase in momentum in the boundary layer due to the injected gas and the thrust decrement decreasing for the same reason.

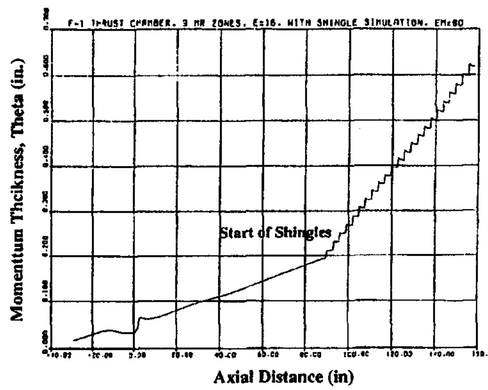

Figure 3. F1 Boundary Layer Momentum Thickness

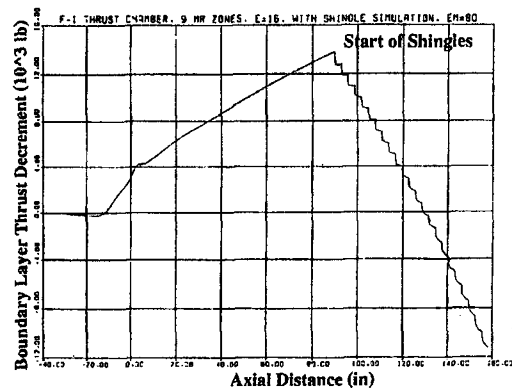

Figure 4. F1 Boundary Layer Thrust Decrement

Dual Bell Nozzle

The dual bell nozzle concept is intended to produce an altitude compensating nozzle. It consists of a regular bell nozzle with another bell nozzle attached, hence the name dual bell. The idea behind the concept is that at low altitude, the boundary layer in the second bell will separate producing a full flowing first bell with a totally separated second bell. Because the separation is intentionally induced, the risk of side load damage is reduced and the performance of the nozzle is higher. At altitude, both bell nozzles would flow full so that the performance increase from the higher area ratio can be realized. The next two figures show a characteristic plot of the dual bell with both bells flowing full and the wall pressure versus axial distance.

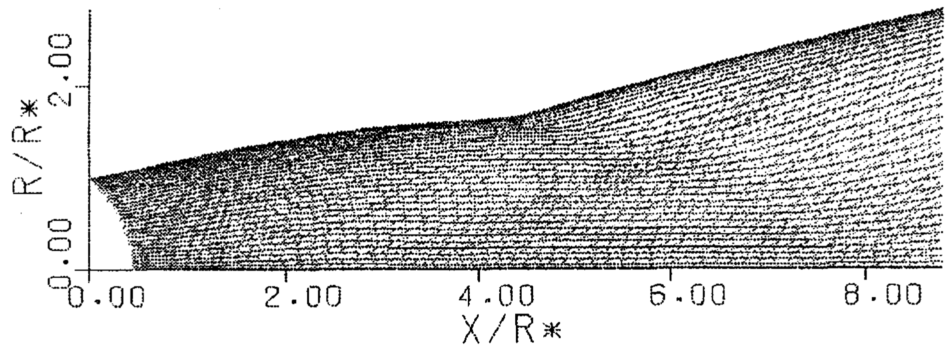

Figure 5. Dual Bell Characteristic Mesh Plot

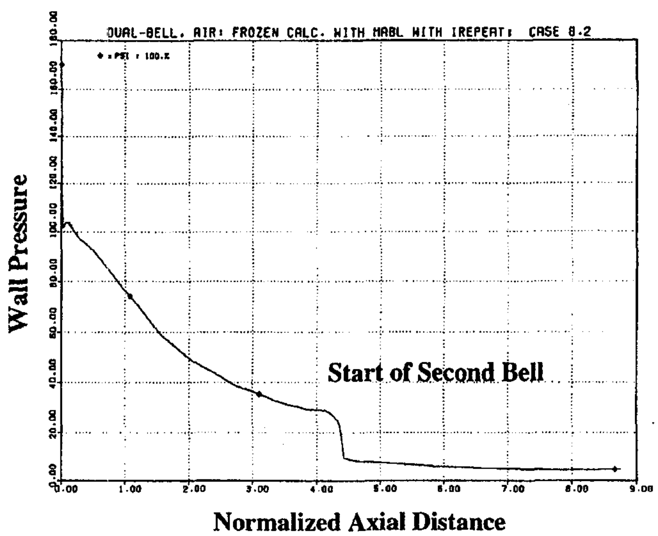

Figure 6. Dual Bell Wall Pressure

Plug Nozzle

The plug nozzle is another altitude compensating nozzle concept. The idea behind this concept is that the external total pressure will hold the flow close to the plug keeping the pressure on the nozzle wall high. As such, the performance of the nozzle is very trajectory sensitive.

The TDK code will treat both 2-D and axisymmetric plug nozzles. The external flow is modeled with a Newtonian pressure boundary and the boundary layer loss is computed for both the upper and lower walls. The main current limitation on the plug nozzle calculation is that there is no base pressure treatment. The basic features of the flow from a plug nozzle are shown in the Fig. 7 followed by a characteristic mesh plot computed by the MOC module in Fig. 8.

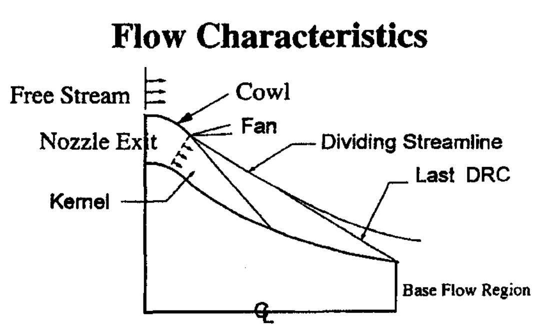

Figure 7. Flow Features Of A Plug Nozzle

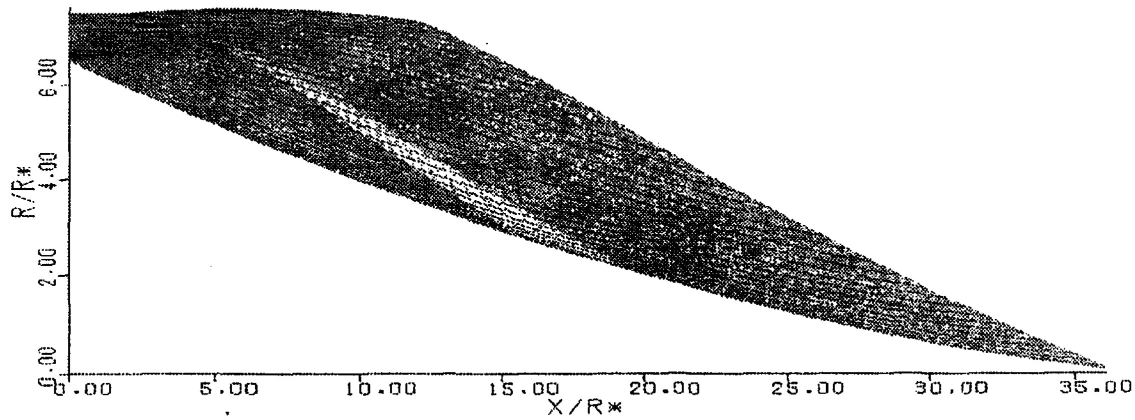

Figure 8. Characteristic Plot For A Plug Nozzle

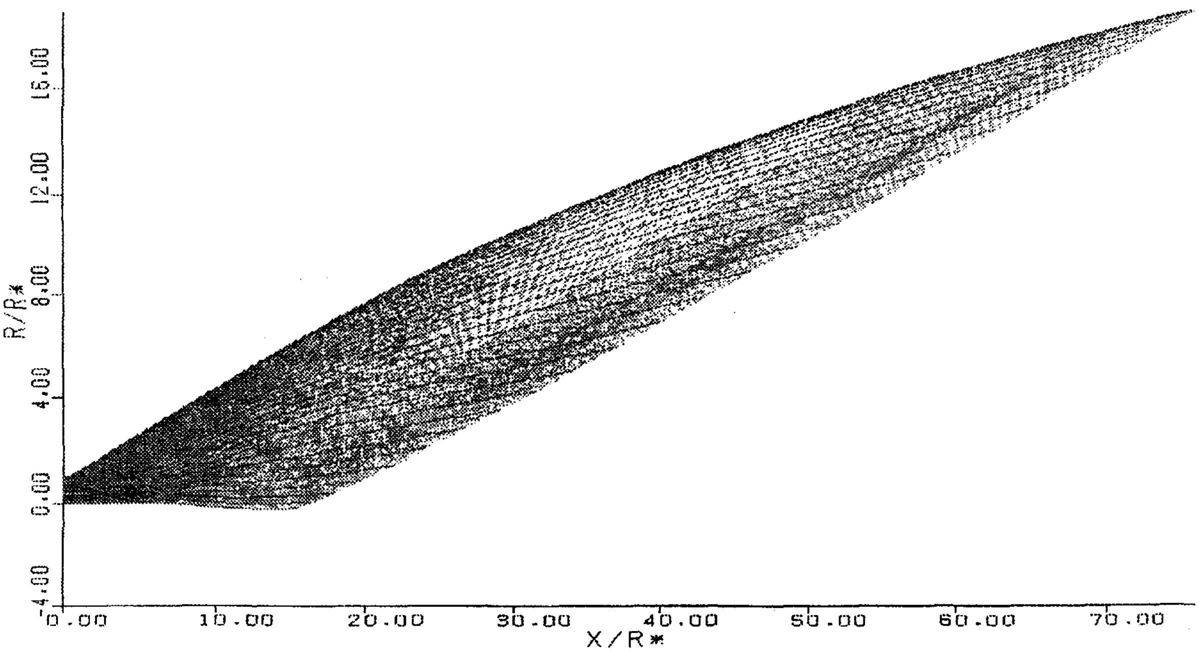

Figure 9. Scramjet Characteristic Mesh Plot

Scramjet Nozzle

The TDK code has a basic scramjet capability. Both 2-D and axisymmetric nozzles can be modeled. A Newtonian pressure boundary is used for the free stream interaction and boundary layer losses are computed for both the nozzle and the cowl. The flow is assumed to start at the burner exit with both the flow properties and species concentrations known. Results in the form of a characteristics plot for a NASP type configuration are shown in Figure 9 above. The expansion fan and subsequent compression from the cowl lip can be seen.

Transpiration Cooled Nozzle

Transpiration cooling can be a very efficient way to keep the nozzle wall cool, especially in the throat region which is subjected to the maximum heating rates. TDK can use frozen, chemical equilibrium, and finite rate kinetics chemistry models to simulate transpiration cooling. An SSME engine configuration was run for the case of no transpiration cooling and with injectants of methane and hydrogen. The blowing rate selected was .05 lbm/sq. ft./sec and the adiabatic wall temperature was calculated assuming that the chemistry was in chemical equilibrium. The effect on adiabatic wall temperature is dramatic with the lighter molecular weight hydrogen being a more efficient coolant.

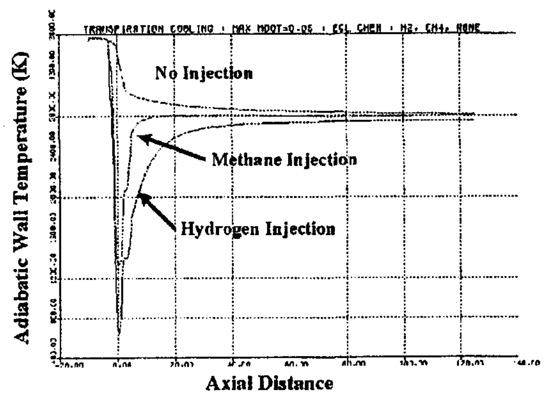

Figure 10. Effect of Transpiration Cooling On T_{aw}

Conclusions

In the United States, the practices used at each engine manufacturer and cognizant analysis organization can vary significantly. However, for engines employing standard bell nozzles, the JANNAF procedures as outlined in CPIA 246 are generally followed. That is, the TDK (Two Dimensional Kinetics) computer program is used to model all of the losses which we have described above.

Over the past 29 years[16], the TDK code has been shown to be an accurate and efficient flow solver. Improvements to the code have made it more applicable to the task of nozzle performance prediction than ever. Table 7 shows a comparison of predicted versus measured results for the TDK code for various bell nozzles.

Table 7. Comparisons of TDK and Measured Nozzle Performance

Engine Name	TDK/MABL Isp Prediction	Measured Isp
Adv. Space Engine	473.58	477.9
Hughes 5 lbf	216.65	214.52[15]
NASA/LeRC Hi-E	480.31 (458.7)[a]	468.9
SSME	457.7	452.6
RL 10	463.03	458.7
XLR-134	468.68	-

[a] corrected for 95.5% measured C* efficiency

The state of the art in nozzle loss prediction is much better than that of injector performance. However there are still issues which need to be resolved. The most important of the issues for nozzle losses is the establishment of an adequate turbulence model for the boundary layer calculations. A unified model which is applicable for all speed regimes and includes finite rate chemistry is required.

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