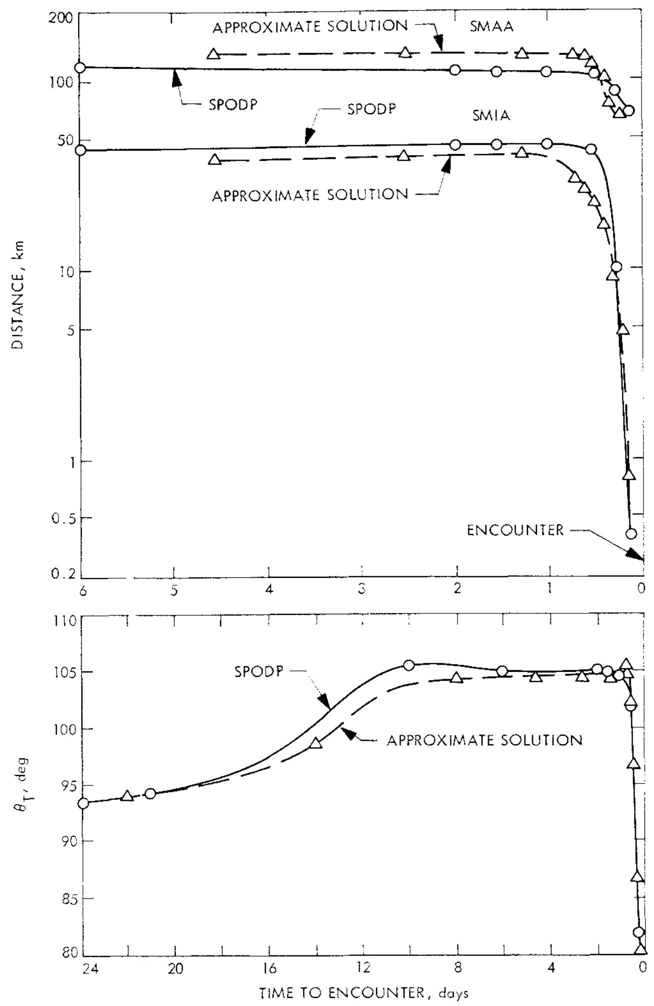

Fig. 4 Size and orientation of SMAA and SMIA: Mars '73.

where $t_F - t_I$ is the duration of the t acking pass (normally 8 to 12 hr). The coefficients a_j, b_j, and c_j vary with time as the spacecraft approaches the planet; they are, therefore, computed at sufficiently frequent intervals throughout the approach phase so that the J^* matrix is reasonably well approximated. The foregoing expression for $(J_i^{*\prime})_{jk}$ represents the "average" data from one station during the time it is tracking the spacecraft, but must be adjusted for the fact that there may be more than one station used and that no one station can track the spacecraft 24 hr/day. Therefore, the final form of J_i^* is adjusted to reflect these considerations as follows:

$$J_i^* = (N/24)J_i^{*\prime} \tag{22}$$

where N is the number of tracking stations (1, 2, or 3).

Results

The results of the approximate technique will be compared with the results of an integrated, real-time, single precision orbit determination program (SPODP) for three representative trajectories: Venus '67, Mars '69, and Mars '73. Identical assumptions and a priori uncertainties were used for both approximate and exact calculations, and are summarized in Table 1.

The results for Venus '67 (Fig. 2) indicate good agreement between the SPODP and the approximate solution. The major difference is that the SPODP indicates a fairly sharp transition in the uncertainty ellipse at encounter minus 1 day, while the approximate solution results in a somewhat earlier and more gradual transition from initial to final orientation of the uncertainty ellipse. In general, the size, shape, and orientation of the uncertainty ellipse in the aim plane are remarkably well approximated and are certainly within the accuracy required for most guidance and navigation studies. For Mars '69 (Fig. 3), the size and orientation of the dispersion ellipse are fairly well modeled, except that the transition from initial to final orientation is slightly delayed.

The selected trajectory for a 1973 Mars mission would permit an arrival at Mars on February 14, 1974. This trajectory is presented because it presently represents the "worst" case encountered among the various comparisons made between the SPODP and the approximate solution. Figure 4 shows that the estimates of the semimajor axes provide a fair approximation to SPODP results. The orientation angle of the dispersion ellipse, however, is reproduced with very remarkable accuracy.

In conclusion, the combination of three hypothetical trajectories in the calculation of approximate partial derivatives appears to have been successful in modeling aspects of a three-body trajectory which previously required numerical inregration. This approximate solution technique described in this Note should be of great value in performing navigational accuracy studies for interplanetary spacecraft trajectories. Further extension of this modeling procedure into other three-body trajectory problems also may be possible.

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ICRPG Liquid Propellant Thrust Chamber Performance Evaluation Methodology

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Development of the Methodology

THE Performance Standardization Working Group (PSWG) of the Inter-Agency Chemical Rocket Propulsion Group (ICRPG) was organized in 1965 to develop a methodology for the experimental determination, analytical prediction, correlation, extrapolation, and flight confirmation of the performance of liquid propellant rocket engines. The Working Group Steering Committee established three working committees: Over-all Concepts, Theoretical Methods, and Experimental Methods. To define a useful objective which could be accomplished within a finite time, the initial effort was

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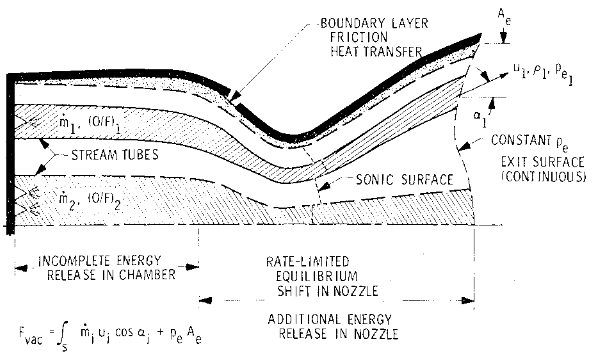

Fig. 1 Internal processes in the real rocket motor.

limited to: 1) thrust chamber assembly performance (feed systems and external aerodynamic and other effects were excluded); 2) those liquid propellants having only gaseous exhaust products and associated with major dollar engine developments would be considered; 3) engines operating in the steady-state mode; 4) engines greater than micro-thrust size; and 5) conventional De Laval nozzles.

The analytical prediction of the fundamental thrust chamber performance parameter (vacuum specific impulse) required first a compilation of the physical processes that are involved in the operation of a real thrust chamber. It was agreed that, in principle, prediction of the performance of a real thrust chamber should account for deviations from the ideal (one-dimensional, equilibrium; ODE) performance due to the conditions and processes shown in Table 1.

A very important concept is that there are interactions between the listed processes and losses. The calculation of performance must include all processes simultaneously; it is not sufficient to calculate each loss independently, and then to subtract the sum of the losses from the ideal performance. The six performance losses listed in Table 1 are believed to account for all the performance inefficiencies that occur within a real thrust chamber. Ultimately, it should be possible to determine analytically all of these performances losses.

The problem of selection or development of computer programs which adequately modeled the various parts of the performance problem, and which could be combined to perform an integrated calculation was then undertaken, and the Working Group contracted for a variety of study efforts.[1-14] The initial phase of the effort culminated in a manual[12] which described the computer programs and calculation procedures and sequence that represented the best current state-of-the-art integrated performance calculation for the five processes and losses which were considered.

The recommended calculation procedure was based on two computer programs, the two-dimensional kinetic nozzle analysis computer program (TDK),[10] and the turbulent boundary-layer nozzle analysis computer program (TBL).[11] The TDK program computes performance for an inviscid reacting gas with finite-rate chemical reactions flowing through an axially symmetric contoured nozzle; two concentric mixture ratio zones can be accommodated, and the effect of incomplete energy release is approximated by using effective heats of formation of the propellant components which are lower than the accepted values. The TBL program accounts for the viscous and heat-transfer effects along the thrust chamber wall by calculating the mass and momentum deficiencies which result. An iterative procedure in which an initial TDK calculation is used to provide input for a TBL calculation, which provides input corrections for a second TDK calculation, is used to determine specific impulse. The sequential operations which constitute a calculation of the performance that includes the effect of interactions between the loss-producing processes are 1) assemble input data; 2) do initial TDK calculation, using real nozzle; 3) use TBL to compute δ^* and θ, using TDK output; 4) use δ^* to generate equivalent potential flow nozzle contour; 5) repeat TDK calculation, using potential flow nozzle (TDK′); 6) compute boundary layer I_{sp} correction, using δ^*, θ, and TDK′ properties; 7) compute multistream tube-core correction, if required; 8) calculate I_{sp}. Stop if η_{ER} was specified in step 1; 9) compare with I_{sp} experimental; and 10) iterate on energy release efficiency, starting at step 5. Stop when predicted performance equals measured performance.

In this current procedure the energy release process is approximated by an admittedly imperfect empirical model, and the appropriate kinetic reaction rates are not known precisely enough for accurate calculation of the performance of certain propellants. The use of the energy release efficiency, η_{ER}, as a free variable when correlating test data means that the value of η_{ER} so determined will also include uncertainties or errors associated with the experimental and physical input data and with the calculation of the other performance losses.

Reference 12 can be used, within the limitations previously listed, to predict performance at altitude, within or near the desired goal of $\pm 1\%$. Normally, this requires evaluation of η_{ER} by correlating test data obtained from a ground-test version of the specific rocket motor. However, in the absence of specific test data, performance can be predicted with almost as good accuracy as the aforementioned by using an estimated η_{ER} based on experience with similar propellant, injector, and thrust-chamber configurations.

Imperfections and Limitations in the Current Methodology

In a real rocket thrust chamber (Fig. 1), the propellant is introduced at the injector and broken into droplets. There may be several "stream tubes," each having a different internal mixture ratio. Combustion starts near the face of the injector. As the propellant droplets move downstream they are shattered and/or evaporated by contact with the hot, high-velocity, combustion gases. The vapors from the propellant droplets mix and react as they are formed, or after some delay. Complete evaporation and reaction and the formation of equilibrium combustion products may or may not be achieved within the low-velocity portion of the combustion chamber. As the gas flows through the nozzle, there may still be some primary evaporation and reaction occurring. In the nozzle the shifts in combustion product composition corresponding to the changes in pressure and temperature are rate limited. At some point in the chamber or nozzle, solid particles may form in the combustion products. The presence of these solid particles affects the mass and temperature distribution within the flow. The actual nozzle flow is three-dimensional, and exits with a divergent velocity component. Friction and heat transfer at the thrust chamber wall combine to produce a boundary-layer flow that is deficient in momentum and energy.

Now consider the computational model as defined by the programs TDK and TBL and by the input data available for use with these programs. Figure 2 attempts to show schematically what is involved in the TDK program. The

Table 1 Real rocket-motor processes and losses

Process	Typical loss, $\% I_{sp_{vac}}$
1) Nonuniform mixture ratio distribution—stream tubes	0–5
2) Incomplete energy release	1–5
3) Multiphase flow, solid particles	Not considered
4) Two-dimensional flow, curvature, and divergence	0.1–3
5) Finite reaction rates, kinetics	0.1–10
6) Boundary layer, friction, and heat transfer	0.5–5

propellant is assumed to be injected into and to react in an infinite reservoir, at stagnation pressure. The η_{ER} effects are currently accounted for by using a decreased effective heat of formation of the inlet propellant, thus decreasing the energy and temperature of, and slightly changing the equilibrium composition of, the combustion gases. From stagnation conditions in the assumed infinite reservoir, the gases expand to the end of the combustion chamber (start of nozzle contraction) by an ODE process. Flow from the nozzle inlet to the throat region is computed assuming a one-dimensional kinetic (ODK) process. Flow through the transonic region is computed using constant gas properties determined at the throat from the ODK calculation. The transonic flow calculation is limited in the sharpness of the throat curvature which it can accommodate. The final expansion of the gases through the supersonic portion of the exhaust nozzle is computed as a two-dimensional method-of-characteristics kinetic-rate-limited flow (TDK), from which the entire program assemblage draws its name. Throughout, the propellant and combustion gas composition are limited to compounds formed from the six elements: C, H, N, O, F, and Cl. All combustion products must be gaseous.

The TBL program assumes that a turbulent boundary layer starts to develop at the upstream (injector face) end of the chamber wall, and continues to develop as the flow progresses toward the nozzle exit.

Physical data input is required to perform many of the physical process calculations previously described. Heats of formation of propellants and equilibrium constants of reaction products are known to a reasonable degree of accuracy. On the other hand, the uncertainty range associated with the existing chemical reaction rate data used for the kinetic program calculations is in some instances sufficient to introduce appreciable uncertainty into the computed performance.

The following desirable improvements in calculation capability would increase the scope of applicability and the accuracy:

1) A physically realistic combustion (η_{ER}) model which relates the multiple stream tube generation of product gases in the finite combustion chamber to the injected propellant mass flux and droplet size distributions.

2) An expanded list of chemical elements and compounds which can be handled.

3) Consideration of the effects of solid particles (as well as liquid droplets) in the combustion chamber.

4) Consideration of the effect of mixing along stream tube boundaries.

5) Refinement of the nozzle convergent region treatment to include a) two-dimensional flow, b) kinetic effects, c) solid particles and liquid droplets in flow, d) continuing evaporation and chemical reaction with gas generation and heat release in this region, and e) multiple stream tubes.

6) Refinement of the transonic nozzle analysis to handle a) small throat curvature ratios, b) multiple stream tubes (discontinuous sonic surface), c) liquid droplets, evaporation, and continuing reaction, d) kinetics, and e) solid particles in the flow.

7) Inclusion of the effects of a) multiple stream tubes, b) liquid droplets, evaporation and continuing reaction, and c) solid particles in the flow in the nozzle supersonic region treatment.

8) Modification of the boundary-layer treatment to account for geometry effects on its development.

9) Acquisition of additional and more precise physical data on a) droplet size and mass distribution resulting from injection processes, b) droplet shattering and evaporation in a hot gas stream, c) chemical reaction rates, and d) multistream tube mixing rates.

What has been discussed thus far is improvement based on the initial limitation of effort, except that a broader range of propellants and the effects of particles in the gas would be considered. Still to be included would be the capability of handling transient operation, micro-thrust engines, other than De Laval nozzle shapes, and mass addition, such as film coolant or turbine gas exhaust, at the chamber or nozzle wall.

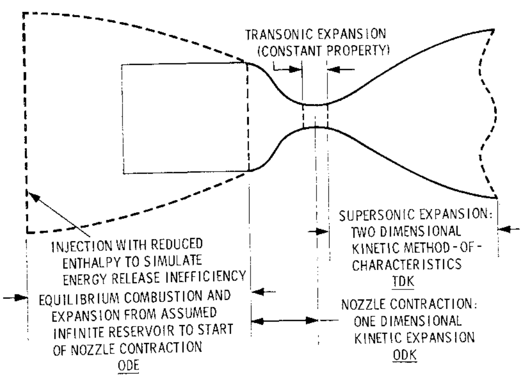

Fig. 2 Internal steps in the TDK computer program.

Concluding Remarks

The ICRPG liquid propellant thrust chamber performance evaluation methodology described here is now being used, where applicable, by government agencies (Army, Navy, NASA, Air Force) as a reference procedure in evaluating proposals and results on appropriate contracts. It is adequate for correlation and prediction of performance (in conventional thrust chambers) of propellants in large scale use during 1964–1968. Improvement in the accuracy of performance prediction, and extension of this capability to the higher performance fluorinated or "space storable" propellants, are desirable and are expected to be done as rapidly as the necessary technology can be developed.

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Streamwise Directed Vortices and Crosshatched Surfaces of Re-Entry Vehicles

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RECENTLY systems of streamwise directed vortices and crosshatched surfaces of re-entry bodies have attracted a good deal of attention. The streamwise directed vortices appear to play an important role in various flow situations involving curved flow, and they give rise to important physical phenomena within practical heat transfer and ablation situations. This note draws attention to some results obtained by means of a water jet in exploring these phenomena.

The experiments were made in an apparatus which delivered a vertical water jet through nozzles of three different sizes. The jet velocity could be regulated with great accuracy within the range 7–32 fps. The models consisted of a 30° cone, a 90° cone and a flat plate which were coated with paint and exposed axisymmetrically to the jet. M. Scherberg at Aerospace Research Labs. Wright-Patterson Air Force Base, suggested that the experiments might also be used to exhibit the cross-hatching process connected with ablation, an idea that has been substantiated by the latest findings of the author.

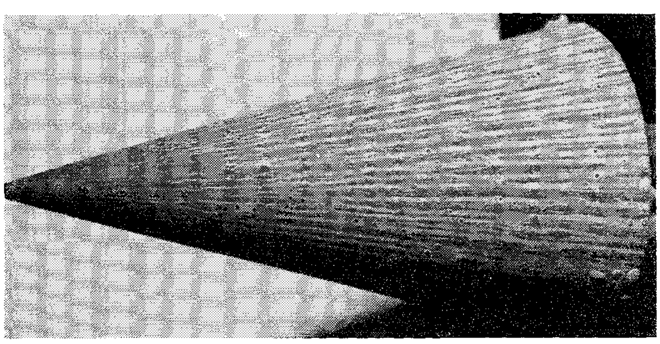

Fig. 1 Striations on a 30° cone model.

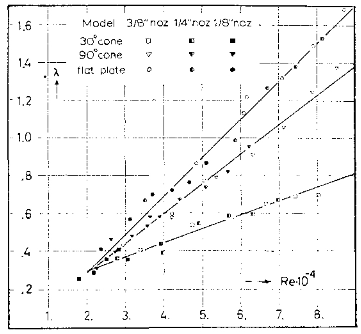

Fig. 2 Number of striations vs jet Reynolds number.

When the models were coated and exposed to the jet with the paint still wet, striations would occur in the coat as exemplified by the 30° cone model shown in Fig. 1. These striations show such a remarkable regularity and correlate so well with the parameters of the flow as to suggest that the striations are indicative of a three-dimensional flow close to the surface, which may be characterized as streamwise directed vortices. The regularity of the striations is clearly brought out in Fig. 2 where the number λ of striations per degree angle as function of the Reynolds number of the jet is plotted for the three different models with three nozzle sizes. A detailed report on these experiments[1] is at present in print and further work is in progress to recorrelate the data making them directly applicable to flight vehicles.

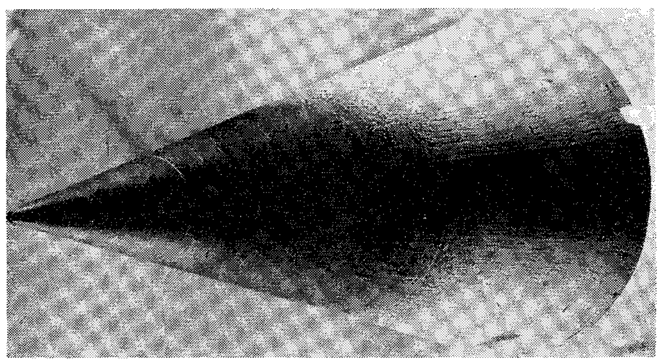

Fig. 3 Re-entry vehicle in the Sparta program (Chrysler Corporation).

Through the courtesy of the Chrysler Corporation, Missile Division, a good deal of material from their Sparta program was placed at the author's disposal. Figure 3 shows one of their recovered re-entry vehicles in the form of a 27° cone, and the striations found in the surface of this cone show a remarkable similarity to those found in the model in Fig. 1. Superimposed on the Sparta streamwise pattern one finds a crisscross pattern which is exhibited in Fig. 4 showing details from rubbings (a paper is wrapped around the model and rubbed) of other Sparta cones.

It turns out that the present water analogy experiments are very well suited for investigations of the ablation process.

Fig. 4 "Rubbing" from recaptured re-entry cones (Chrysler Corporation).

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