Agena Propulsion System Flight Performance Prediction

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An Agena propulsion system performance model predicts the engine propellant flowrate and vehicle mass status; computes the over-all propellant mixture ratio required for propellant tank loading calculations; predicts the engine thrust time history applied to trajectory-shaping calculations; and predicts the engine specific impulse which, in part, determines the mission payload capability. Precision of the performance model is evidenced by good agreement between predicted and actual flight data.

Nomenclature

A = area
a = vehicle acceleration
C = pump cavitation factor, combustion efficiency parameter or coefficient
F = thrust; $F_E = F_{TC} + F_D$
g = units conversion factor or standard gravitational acceleration
H = propellant head
HSV = pump suction head above vapor pressure
I_{sp} = specific impulse; $I_{spE} = F_E/\dot{W}_E$
K = propellant flowrate calibration constant
M = vehicle mass
N = engine turbine speed
P = pressure
R = propellant mixture ratio, $\dot{W}_{ox}/\dot{W}_f$
SG = specific gravity
T = propellant temperature
t = time
$\dot{W}$ = propellant mass flowrate; $\dot{W}_E = \dot{W}_f + \dot{W}_{ox} + \dot{W}_g = \dot{W}_{TC} + \dot{W}_g$
ρ = propellant density

Subscripts

AT = acceptance test
c = units conversion constant
D = turbine exhaust duct
E = engine (thrust chamber plus gas generator)
F = thrust
f = fuel
g = gas generator
ox = oxidizer
P = propulsion
TC = thrust chamber
R = radar
t = nozzle throat
TC = thrust chamber
V = vehicle or vapor

Superscripts

$*$ = characteristic velocity or combustion efficiency
$(\bar{\ })$ = product average value

Introduction

ACCURATE orbit injection of a spacecraft requires that the propulsion system thrust $[F_E(t)]$ and propellant flowrate $[\dot{W}_E(t)]$ histories closely follow the predicted values. Efficient use of the total capability of the boost vehicle requires that I_{spE} (specific impulse) and R_E (mixture ratio) be predictable within narrow tolerances. These four basic parameters—F_E, $\dot{W}_E$, I_{spE}, and R_E—are the primary outputs of a propulsion system flight performance prediction model.

The Agena is equipped with a pump-fed, 16,000-lbf-thrust, bipropellant, multiple-start engine (Fig. 1). A turbine, powered by a gas generator, spins the pumps through a gear box. Gas generator combustion products are routed through the turbine wheel and alongside the thrust chamber through an exhaust duct, which terminates near the thrust chamber nozzle exit. The vehicle propellant capacity is about 9500 lb m of inhibited red fuming nitric acid (IRFNA) and about 4000 lbm of unsymmetrical dimethylhydrazine (UDMH). Approximately 250 similar vehicle configurations have been flown. During this flight experience, progressively advanced methods of preflight propulsion system performance prediction have been developed. This paper presents the current methods of preflight propulsion system performance prediction for Agena.

The first parameters calculated by the model are the oxidizer ($\dot{W}_{ox}$) and fuel ($\dot{W}_f$) flowrates delivered by the turbopump assembly to the gas generator and thrust chamber assemblies. These flowrates form the basis of the thrust-time history and vehicle mass status, which together determine the vehicle acceleration schedule. From $\dot{W}_{ox}(t)$ and $\dot{W}_f(t)$, the over-all oxidizer/fuel ratio for the mission R_E is determined and used to calculate the propellant load unique to that mission and vehicle, based on individual tank volume calibrations. Next, I_{spE} is calculated. A product or family average I_{spE}, determined from flight data, may be used for the entire group of engines, or a unique value may be calculated for a specific engine utilizing the engine acceptance test data and the predicted over-all mission mixture ratio and propellant temperature. The I_{spE} determines the propellant utilization efficiency, and when combined with the individual propellant load and vehicle empty weight, sets the mission payload capability. The product of the I_{spE} and $\dot{W}_E$ yields the F_E history used for mission trajectory shaping.

Presented as Paper 69-453 at the AIAA 5th Propulsion Joint Specialist Conference, U.S. Air Force Academy, Colo., June 9–13, 1969; submitted June 19, 1969; revision received November 24, 1969. The author wishes to express his gratitude to T. Kenyon, who has been responsible for the layout and typing of this and many other technical manuscripts.

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Performance Model Development

Propellant Flowrate and Mixture Ratio Models

The instantaneous propellant flowrates from the pumps to the thrust chamber are calculated with an equation of the form

$$\dot{W} = KC\rho N \tag{1}$$

Each pump is calibrated by the manufacturer, and the constant K which adjusts the $\dot{W}$ vs suction-head characteristics

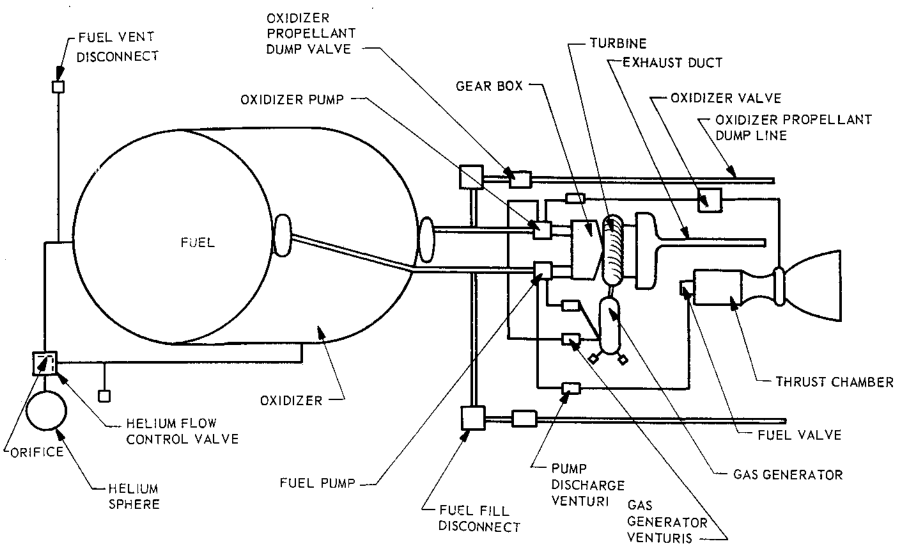

Fig. 1 Agena propulsion system schematic.

of the pump to a standard cavitation curve (C) is determined.

The Agena vehicle is equipped with an orifice controlled pressurization system that delivers a repeatable helium mass flow profile from the helium sphere to the propellant tanks. This pressurant mass flow profile, coupled with the predicted propellant volumetric withdrawal rates, determines the propellant tank pressure schedule. This pressure is empirically adjusted for feedline flow frictional losses, and the propellant acceleration head is added to obtain the predicted pump suction total head. The head calculation is the first step of the performance model. Statistical evaluation of flight propellant pump inlet temperature data revealed that propellant thermal conditioning during loading and vehicle air-conditioning on the pad yields propellant temperature traces that are predictable to within $\pm 5°F$. The temperatures are used in the performance model to determine the instantaneous values of propellant density (ρ).

The engine turbopump assembly is equipped with a passive turbine speed control system. The turbine power available is a function of N^2, whereas the power required by the pumps is a function of N^3; N stabilizes at the point where the power-available and power-required curves intersect. This point is a function of the pump suction head (HSV), propellant temperature (T), the individual unit propellant flow venturis, rotating friction and turbine efficiency. Calibration of the unit at a reference standard H and T provides an initial N value for the performance model. An empirically determined set of influence coefficients is used by the model to adjust the speed to the predicted SG and instantaneous values of T and HSV. With instantaneous values for HSV, T, and N, the pump cavitation factors (C) are calculated and applied to Eq. (1) to determine $\dot{W}_f$ and $\dot{W}_{ox}$. An equation of similar form is used to calculate the gas generator propellant flowrates.

This calculation procedure is repeated, using the predicted propellant tank P and T values to calculate instantaneous values of N, C, and $\dot{W}$. The flowrates are summed to provide continuous vehicle weight data. Lump values of oxidizer and fuel are withdrawn from the propellant tank by the model at each engine start and shutdown. Following this procedure, the vehicle propulsion system mass status is maintained.

After completion of the simulated mission, the model calculates the over-all mixture ratio of the impulse propellant consumed by the propulsion system. This value is used with the calibrated propellant tank volumes to compute the maximum mission propellant load that may be carried without leaving excess deadweight oxidizer or fuel upon depletion of the other propellant.

Engine Specific Impulse and Thrust Models

Thrust is produced by the engine thrust chamber and by the gas generator combustion products expelled through the turbine exhaust duct. The I_{sp_E} is defined as

$$I_{sp_E} = F_E/\dot{W}_E = (F_{TC} + F_D)/(\dot{W}_{TC} + \dot{W}_g) \tag{2}$$

Historically, Agena flight performance predictions have used a product average method (i.e., the sum of flight performance values divided by the number of samples) for determining I_{sp_E}. On many Agena boost missions, the propellant mixture ratio, propellant temperatures and engine-to-engine product variability were found to be repeatable and produced relatively small deviations from the product average I_{sp_E}. Performance studies showed that the product average I_{sp_E} and its deviation met the injection payload requirements; therefore, the simple product average I_{sp_E} model has been used. On long-coast, multiple-burn missions, however, R and T vary significantly among missions and even during a single mission. Recent studies have shown a payload gain for certain engine/mission combinations with an alternative I_{sp_E} model that produces a unique I_{sp_E} for each flight. This method, which is currently being evaluated, is developed as follows. By algebraic manipulation and substitution of the thrust chamber and gas generator specific impulses, Eq. (2) may be expressed as

$$I_{sp_E} = I_{sp_{TC}}[\dot{W}_{TC}/\dot{W}_E] + I_{sp_D}[\dot{W}_g/\dot{W}_E] = [C^*_{TC}C_{F_{TC}}/g_c][\dot{W}_{TC}/\dot{W}_E] + I_{sp_D}[\dot{W}_g/\dot{W}_E] \tag{3}$$

Analyses of ground and altitude test cell firing data have shown measurable variations in the thrust chamber combustion efficiency (C^*) and, to a lesser extent, in the thrust coefficient with R and T. Therefore these parameters were modeled to account for the variances:

$$C^* = C^*_{AT} + [\partial C^*/\partial R]dR + [\partial C^*/\partial T]dT \tag{4}$$

$$C_F = \bar{C}_F + [\partial C_F/\partial R]dR + [\partial C_F/\partial T]dT \tag{5}$$

During calculation of the predicted flight performance with the alternative model, computation of the instantaneous I_{sp_E} is accomplished by calculating the instantaneous values of $\dot{W}_{TC}$ and $\dot{W}_g$, R_{TC}, and predicted T. These values are substituted into Eqs. (4) and (5) with the engine acceptance test value C^*_{AT} and $\bar{C}_F$. The results, in turn, are substituted into Eq. (3) with the calculated value of I_{sp_D}. Using either the primary ($\overline{I_{sp_E}}$) or alternative model [$I_{sp_E}(t)$] for specific impulse, the instantaneous $F_E(t)$ is determined from [$\overline{I_{sp_E}}$ or

$I_{sp_E}(t)]\dot{W}_E(t)$. The instantaneous values of thrust and vehicle weight determine the vehicle acceleration, which is used during the next calculation interval to determine the propellant acceleration head. The calculation sequence is iterated until the desired velocity increment is satisfied, one or both propellants are depleted, or a specified burn duration is reached. The sequence is repeated for multiple-burn missions, and an over-all average value is calculated for each of the primary performance parameters.

Performance Model Verification

Propellant Flowrate and Mixture Ratio

Missions are usually programed to near propellant depletion to take full advantage of Agena capability. Two methods of verifying the propellant flowrate model were developed which capitalize on the low residual propellant values. One method uses the rapid decay of propellant head during terminal drainage of the propellant tanks to approximate, through calibration, the propellant remaining at final shutdown. The remaining propellant is subtracted from the propellant loaded. The average oxidizer and fuel flowrates are calculated, based on the burn duration, and compared to the average value computed by the performance model. On many Agena missions, another method is preferred because residual propellants and pressurization gas are dumped overboard to preclude on-orbit torques caused by random leakage. The flight propellant tank pressures are monitored, and the times required to dump the small quantities of residual propellants are measured. Multiplication of the propellant dump time by the propellant dump flowrates yields accurate values for the residual propellants. The residual propellants are again used to determine the average propellant flowrate, and the results are compared with performance model predictions. The accuracy of this method depends upon the quantity of propellants dumped overboard, since the uncertainty in trapped propellants is limiting. Propellant consumption accuracy values of 0.17% to 0.35% were calculated for residual propellant weights from 50 to 200 lbm. In both cases, results showed that average predicted flowrates were slightly lower than flight values. Performance model flowrates were individually brought into agreement with the flight data by biasing the K's (Eq. 1). The results were then averaged to determine a set of performance model flow biases for subsequent $\dot{W}_E$ and R_E predictions (Fig. 2).

The specific error sources responsible for the biases have not been conclusively identified. It was found, however, that the propellant flowmeters used to determine the K's are calibrated with water at ambient pressure. During engine acceptance tests, propellants are pressurized with N_2 at 24 psia, whereas in flight, the pressurizing agent is

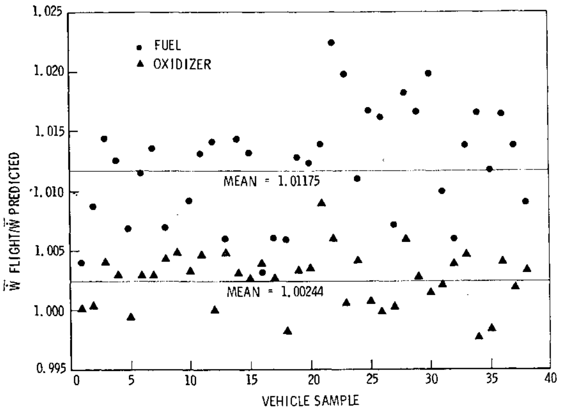

Fig. 2 Ratio of flight to predicted flowrates.

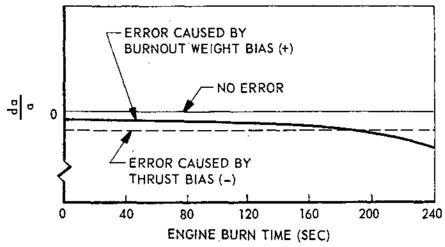

Fig. 3 Acceleration difference vs time characteristics.

He at $\simeq 9$ psia. Until the cause of the flowrate bias is identified and eliminated, the performance model must incorporate the average flow biases.

Engine Thrust and Specific Impulse

When confidence was established in the propellant flow rate model, the accuracy of the predicted thrust model was evaluated by comparing vehicle accelerations calculated with the performance model to those calculated with ground-based-radar vehicle tracking data. The radar tracking data were verified by comparing it with vehicle orbit injection parameters. Newton's second law of motion reveals that variances in vehicle acceleration can be produced by thrust or by vehicle mass errors. The propulsion model and radar tracking accelerations may be arranged to separate these two error sources:

$$a = F/M \qquad (6)$$
$$da/a = (a_R - a_P)/a_P = dF/F - dM/M$$

If there are no errors in the performance model, Eq. (6) will plot as a straight line on the time axis (Fig. 3). If there is a constant bias in the propulsion system thrust model, Eq. (6) will plot as a straight line displaced from the axis (Fig. 3). A vehicle empty-weight bias, however, produces a diverging plot, since it represents an ever increasing percentage of the remaining vehicle mass as the vehicle burntime increases. A constant or time-varying propellant flowrate error initially plots on the time axis and then diverges as the vehicle mass bias grows.

Following this evaluation, a bias in the thrust model of about +0.6% was calculated. Typical results are presented in Fig. 4. Investigations were conducted on the several terms of the basic engine thrust equation ($F_E = C_F P_{TC} A_t + F_D$) to isolate potential bias sources. About 20 thrust chambers were measured for throat and exit areas and found to agree closely with the engine records. Laboratory tests revealed a small (+1.0 psia) bias in the P_{TC} transducer output, which appears to correlate with storage life after calibration. The C^* of the flight thrust chamber was calculated using the adjusted $\dot{W}_{TC}$ and P_{TC} data and was compared to the engine acceptance test values normalized

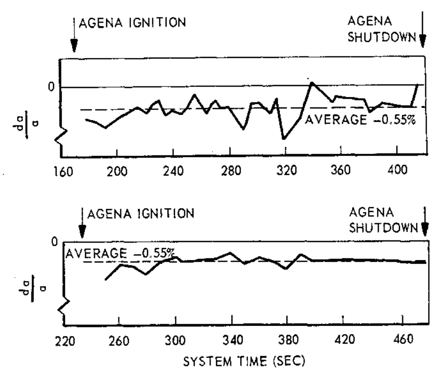

Fig. 4 Flight acceleration difference vs time.

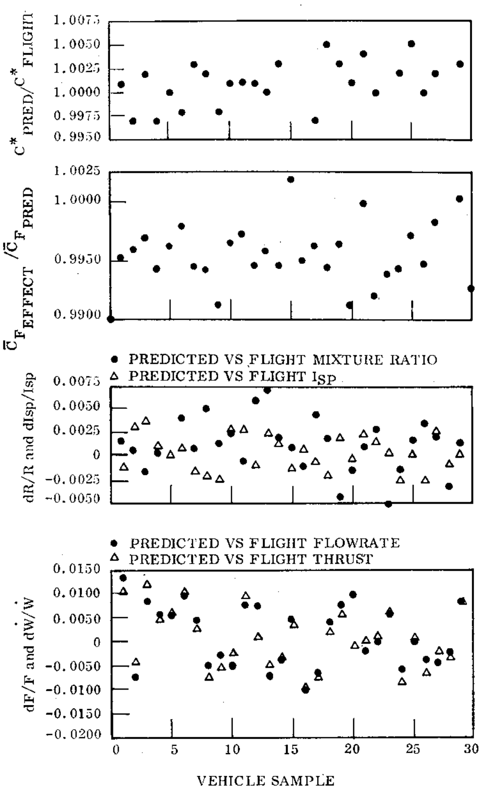

Fig. 5 Agena predicted and flight data.

to R and T (Fig. 5). These comparisons showed good general agreement, which added to the confidence of the over-all propulsion performance model.

After the flight data were corrected for the P_{TC} transducer bias, there still remained a $+0.4\%$ (60 lbf) bias, which was attributed to a positive bias in the thrust coefficient and/or turbine exhaust duct thrust. The exhaust duct thrust has not been measured experimentally, but its calculated value is about 200 lbf. Because the thrust bias is large compared to the exhaust duct thrust, it was concluded that a majority of the 60-lb thrust bias was due to a bias in the thrust coefficient C_F. Evaluation of historical engine development data revealed that a limited number of vacuum chamber tests had been conducted, using prototype thrust chambers and nozzle extensions, to determine the product average C_F. In addition, it was found that toward the end of the development program the thrust chamber propellant injector design had been modified without reevaluation of C_F. Equation (3) shows that a bias in C_{FTC} and/or exhaust duct thrust ($I_{spD}\dot{W}_g$) would produce an I_{spE} bias. Until thrust chamber and exhaust duct vacuum chamber test results allow resolution, the residual $+0.4\%$ thrust bias is incorporated into the primary and alternate performance models by adjusting flight I_{spE} values until the calculated propulsion model and radar vehicle tracking accelerations match. For the alternative model, the adjusted I_{spE} was substituted into Eqs. (3) and (5), with the calculated value for I_{spD} and other flight data and effective $\bar{C}_{FTC}$ value computed (Fig. 5). This method was applied to a statistically significant sample of flight data and the product average values of I_{spE} and effective thrust coefficient determined and incorporated into the primary and alternative engine performance models, respectively. As an independent check, propellant utilization values were calculated for a sample of flights by subtracting the residual propellants dumped overboard after engine shutdown from the loaded propellants, as discussed under the propellant flow model. For each vehicle, the velocity increment and initial weight were used to calculate the I_{spE} satisfying the propellant utilization data. Results were compared to the radar acceleration adjusted values and were found to show close agreement.

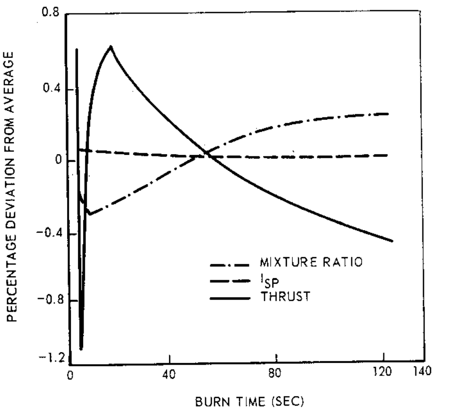

Fig. 6 Predicted Agena performance characteristics.

Agena Propulsion Performance Prediction

The Agena propulsion performance model is used to predict average values and time histories of $\dot{W}_E$, R_E, I_{spE}, and F_E in advance of each mission. Based on these parameters and propellant tank calibration data, the mission propellant load, which will provide maximum impulse and minimum residual propellants for that engine/propellant tank/mission combination, is calculated. These data are then incorporated into a total booster/Agena mission trajectory simulation to determine the mission payload, trajectory shaping, and propellant reserve parameters. Figure 6 presents a typical computer plot of dimensionless R_E, I_{spE}, and F_E. Tabular data and average values are also supplied.

Orbit injection accuracy of an Agena is significantly dependent upon the precision of the propulsion performance model. Precision is defined as the difference between the predicted and flight values of a performance parameter divided by the predicted value and is the most significant measure of performance model capability. A precision analysis that utilizes the product average value for the predicted I_{spE} has been conducted for the basic Agena performance model. Results of this analysis, based on 29 flights, are (3σ): I_{spE}, $\pm 0.55\%$; F_E, $\pm 1.85\%$; $\dot{W}_E$, $\pm 1.81\%$; and R_E, $\pm 0.83\%$. Detailed results are shown in Fig. 5.

A precision analysis was also conducted for the alternative performance model, which uses a unique I_{spE} based on the engine/mission mixture ratio and propellant temperature, the engine acceptance test C^*, and product average effective thrust coefficient. An I_{spE} precision value of $\pm 0.59\%$ (3σ) was determined for the same 29 flights. A correlation analysis was also conducted and showed a relatively low correlation between the predicted and actual I_{spE}. It was anticipated that this method would improve precision and correlation by eliminating product variability. However, any improvement appears to be offset by the lower combined accuracy of the flight and engine acceptance test data. It is concluded that the payload advantage of the alternative model must be deferred until improved accuracy of flight and engine acceptance test instrumentation yields: 1) higher correlation between predicted and actual I_{spE} and 2) a precision value lower than that obtained with the basic model.

Conclusion

A realistic performance prediction model can be developed through application of the governing equations and boundary conditions. Furthermore, empirical adjustment with sufficient flight performance data results in a model of significant accuracy.