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DISCUSSION OF PERFORMANCE CALCULATIONS OF NUCLEAR ROCKET ENGINES

E. L. Semple

POWER PLANT PERFORMANCE ANALYSIS UNIT

MASTER

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The purpose of this report is to discuss some of the fundamental relationships in a nuclear rocket engine and to present the equations required to calculate the performance of the rocket. The problems associated with these calculations are pointed out.

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Introduction

This report presents a brief discussion of the aerodynamic and thermodynamic relationships pertinent to performance calculations of nuclear rocket engines. The material was prepared originally in fulfillment of part of a contract. Since little work has been done on nuclear rockets, the discussion is concerned primarily with the problems which will be encountered in the thermodynamic design of these power plants. No attempt has been made to suggest operating temperatures, component efficiencies, or possible missions.

Nomenclature

Station Designations

- 1 - propellant (coolant) storage tank
- 2 - storage tank discharge - pump or valve inlet
- 3 - pump or valve discharge - collector inlet
- 4 - reactor inlet
- 5 - reactor discharge - nozzle inlet
- 8 - nozzle throat
- 9 - nozzle discharge
- 0 - ambient

Symbols

- A - area, sq in.
- C_F - nozzle flow coefficient, actual area/effective area
- C_v - nozzle velocity coefficient, actual thrust/theoretical thrust
- F_n - net thrust, lb
- g - acceleration due to gravity, ft/sec²
- h - enthalpy, Btu/lb
- J - mechanical equivalent of heat, ft lb/Btu
- M - Mach number, speed/sonic speed
- P - pressure, psia
- R - gas constant, ft³/lb °R
- s - entropy, Btu/lb °R
- T - temperature, °R
- W - propellant weight flow, lb/sec
- ϕ - entropy function, Btu/lb °R
- γ - ratio of specific heats
- %H - percent of monatomic hydrogen

Subscripts

- s - static
- T - total

Discussion

The nuclear rocket to be considered in this report consists of a propellant storage tank, a valve or pump, a heat transfer reactor, and an isentropic nozzle. The coolant is expanded or pumped from the storage tank through the reactor and then expands through an isentropic nozzle.

Like an air-breathing jet engine, the gross thrust of a rocket is equal to the rate of change of momentum across the nozzle of the engine plus the external pressure force on the engine. Since the working fluid of the rocket is contained within the vehicle, there is no ram drag on the rocket engine, and the thrust defined above is also the net thrust of the engine. The equation used to determine the thrust in terms of enthalpy is the same as that for a turbojet or ramjet engine*:

$$F_n = (W_8/g)C_{v8} \sqrt{2gJ(h_{T5} - h_{s9})} + A_9(P_{s9} - P_0)$$

Unfortunately, the determination of the variables in this equation is not as simple for the rocket as it is for air-breathing jet engines. Dissociation of the coolant will be encountered for most of the nuclear rocket propellants currently under consideration (such as hydrogen, ammonia, water, and various hydrocarbons) at the temperatures generally considered for a rocket. The enthalpy of the gas in the nozzle of the rocket then is a function of the percent dissociation of the gas as well as the nozzle inlet temperature and pressure and nozzle pressure ratio; for this reason the total enthalpy at the nozzle inlet (h_{T5}) rather than the total enthalpy at the nozzle throat (h_{T8}) is used in the thrust equation. In addition to the complication introduced

* The total enthalpy at any point in the nozzle is a constant for an air-breathing jet engine, therefore $h_{T5} = h_{T8}$ for a nuclear turbojet or ramjet engine.

into the calculation of nozzle enthalpy by the dissociation of the coolant, a further problem may be encountered if the temperatures are sufficiently high to ionize the coolant. Some work has been done on obtaining properties of ionized gas, but more extensive information would be required in order to calculate performance of a nuclear rocket in which the propellant was ionized.

For purposes of demonstrating the type of calculation which must be performed to determine the enthalpy of the gas in the rocket engine nozzle, a typical expansion process with dissociation of hydrogen gas (H_2) into monatomic hydrogen (H) will be considered. Tables exist which present the enthalpy and entropy of dissociated hydrogen gas as a function of static temperature and static pressure.* There are two expansion processes which can be calculated for the gas in the nozzle: equilibrium and frozen state. In the equilibrium expansion process the percent dissociation of the gas is reduced as a function of the thermal equilibrium; in the frozen state expansion process there is no reduction in the percent dissociation of the gas, i.e., %H remains constant through the process.

From the tables of properties of dissociated hydrogen, the following graphical relationships can be drawn:

Figure 1

Figure 2

* General Electric Co., Flight Propulsion Division, E. A. Mickel and H. N. Powell, Thermodynamic Properties of Combustion Gases, System H_2 to Air, June 1957.

Figure 3

Figure 4

Figure 5

Figure 6

These plots are required in the solution of both expansion processes. The entropy function, ϕ , shown in Figure 5 can be calculated at each point from the equation, $\phi = s + (R)\ln(P_s)$, and the value of T_s corresponding to P_s and $\%H$ can be read from Figure 2.

If the reactor calculations are carried out on the basis of total conditions, the first step in either calculation is to find the static temperature and pressure at the nozzle inlet. The equations for performing this calculation are:

$$T_{s5} = T_{T5} / \left\{ 1 + \left[(\gamma_5 - 1) / 2 \right] M_5^2 \right\}$$

$$P_{s5} = P_{T5} / \left\{ 1 + \left[(\gamma_5 - 1) / 2 \right] M_5^2 \right\}^{\gamma_5 / (\gamma_5 - 1)}$$

The first equation is a form of the definition of total temperature; the second equation is the relationship between pressure ratio and temperature ratio in an isentropic process such as exists in the nozzle. The Mach number at the nozzle inlet (M_5) can be found as a function of the static temperature and static

pressure, the nozzle inlet area, and the rate of mass flow of the coolant. The ratio of specific heats, γ_5 , is a function of the percent dissociation (β_{H_5}), which in turn depends on the static temperature (T_{s5}) and the static pressure (P_{s5}) at the nozzle inlet. Therefore, the solution of the equations requires an iterative process. If the reactor calculations are carried out on the basis of static conditions, the static temperature and pressure at the nozzle inlet will depend only on the Mach number change from reactor discharge to nozzle inlet, this change being a function of the flow area change. In addition to finding the static conditions at the nozzle inlet, it is assumed that the pressure ratio across the nozzle, P_{T5}/P_{s9} , is known.

The method of calculation of enthalpies in the nozzle for the equilibrium process of expansion requires the following steps:

1. $s_5 = f(T_{s5} \text{ \& } P_{s5})$ from Figure 1
2. $\beta_{H_5} = f(T_{s5} \text{ \& } P_{s5})$ from Figure 2
3. $h_{s5} = f(T_{s5} \text{ \& } \beta_{H_5})$ from Figure 6
4. $h_{T5} = h_{s5} \left\{ 1 + \left[(\gamma_5 - 1)/2 \right] M_5^2 \right\}$
5. $s_9 = s_5$
6. $T_{s9} = f(s_9 \text{ \& } P_{s9})$ from Figure 1
7. $\beta_{H_9} = f(T_{s9} \text{ \& } P_{s9})$ from figure 2
8. $h_{s9} = f(T_{s9} \text{ \& } \beta_{H_9})$ from Figure 6

The enthalpies for the frozen state expansion process can be found by a similar procedure as follows:

1. $s_5 = f(T_{s5} \text{ \& } P_{s5})$ from Figure 1
2. $\beta_{H_5} = f(T_{s5} \text{ \& } P_{s5})$ from Figure 2
3. $h_{s5} = f(T_{s5} \text{ \& } \beta_{H_5})$ from Figure 6

4. $h_{T5} = h_{s5} \left\{ 1 + \left[(\gamma_5 - 1)/2 \right] M_5^2 \right\}$
5. $R_5 = f(\%H_5)$ from Figure 3
6. $s_9 = s_5$, $\%H_9 = \%H_5$, and $R_9 = R_5$
7. $\phi_9 = s_9 + (R_9) \ln(P_{s9})$
8. $T_{s9} = f(\phi_9 \text{ \& } \%H_9)$ from Figure 5
9. $h_{s9} = f(T_{s9} \text{ \& } \%H_9)$ from Figure 6

Both the equilibrium and frozen state expansion processes are ideal processes, and the actual nozzle expansion will lie somewhere between the two. Work with conventional fuels in air-breathing engines has shown the performance based on an equilibrium expansion process from nozzle inlet to throat and a frozen state expansion from nozzle throat to discharge to be approximately correct. At the present time no work has been done to determine whether the performance of a nuclear rocket based on this combination of expansion processes is accurate, nor to determine precisely what the actual expansion process will be. Experimental data will be required in this field to calculate with any certainty the thrust of a nuclear rocket.

Assuming that the thermodynamic conditions in the nozzle can be calculated from some combination of the two expansion processes described above, it is possible to determine the nozzle discharge Mach number:

$$M_9 = \sqrt{\left[2/(\gamma_9 - 1) \right] \left[(P_{T5}/P_{s9})^{(\gamma_9 - 1)/\gamma_9} - 1 \right]}$$

For any realistic rocket design, the value of M_9 will be greater than 1.0 since a rocket nozzle is converging-diverging. If the calculated value of M_9 is less than 1.0, the assumed value of nozzle pressure ratio is too low.

The nozzle geometry can be determined by calculating the nozzle throat and discharge areas from the following equations, assuming that the coolant flow, W_8 , is steady and known:

$$A_8 = \sqrt{R_8 T_{s8} / \gamma_{8g}} (W_8 / P_{s8} C_{F8})$$

$$A_9 = \sqrt{R_9 T_{s9} / \gamma_{9g}} (W_8 / P_{s9} M_9 C_{F8})$$

The static temperature and pressure at the nozzle throat (T_{s8} and P_{s8}) can be calculated in the same manner as the static temperature and pressure at the nozzle inlet (T_{s5} and P_{s5}) realizing that the throat Mach number, $M_8 = 1.0$.

It is important to remember that the acceleration due to gravity, g , which appears in these equations cannot be considered a constant if the nuclear rocket under discussion is intended for flight outside the earth's atmosphere. In this case, the trajectory of the vehicle must be factored into the calculation, and the appropriate value of g used at each point. The flow coefficient of the nozzle, C_{F8} , must be determined experimentally for a particular nozzle design used in a nuclear rocket. This coefficient depends on deviations from the theoretical coolant mass flow caused by friction with the nozzle walls, dissociation and recombination of the gases, and a finite boundary layer.

If the geometry of the rocket engine nozzle is known, an iteration through the calculation procedure will be required to determine the nozzle pressure ratio at which the engine is operating.

The trajectory of the vehicle assumed for the nozzle geometry calculation also will establish the ambient pressure, P_0 , and the value of g to be used at each point in the thrust calculation.

There remains one variable in the equation for net thrust which has not been discussed as it applies to the nuclear rocket: the nozzle velocity coefficient,

C_{v8} . This coefficient reflects the losses in the nozzle due to deviations from the isentropic value of exhaust velocity of the coolant caused by radial motion of the expanding gases, thermal radiation to space, and shock waves in the supersonic flow region. Losses in velocity due to radial motion of the gases are small, but in the case of the rocket, serious; therefore, the nozzle design for a nuclear rocket is more rigorous than that for a conventional air-breathing engine. The magnitude of the nozzle velocity coefficient must be determined experimentally for the nozzle design and coolant characteristics under consideration for a particular rocket design.

Using the information calculated and assumed in the above procedure the net thrust of the rocket engine can be calculated at each point in the assumed trajectory. The trajectory is a function of the thrust of the nuclear rocket and the gross mass of the vehicle. Since the gross mass ~~diminishes~~ as propellant is expended, the required net thrust decreases also; therefore, it is necessary to perform the calculation in increments along the trajectory and to iterate for the solution at each step.

Another complication in calculating the performance of a nuclear rocket was assumed out of existence in the foregoing example, i.e., the possibility of non-steady flow of the coolant. As the rocket progresses along its trajectory, the mass of coolant in the tank decreases. In order to maintain steady flow of the coolant through the reactor and nozzle, the pump or valve at the storage tank discharge must control the release of coolant and adjust its pressure level. It may or may not be desirable to have a steady flow process in the rocket engine, depending on the mission of the vehicle. In any case, the above relationships are not applicable to non-steady flow of the coolant; the

exact solution to obtain the thermodynamic properties of the coolant during operation of the rocket will require differential equations. As a first approximation, it can be assumed that steady flow exists instantaneously at any point in the rocket trajectory. By imposing a schedule of rate of mass flow of the coolant as a function of time, the thrust of the rocket at each point then can be calculated using the steady flow relationships as before. The accuracy of the thrust determined by using this method is not known at the present time. An exact calculation using differential equations for a typical nuclear rocket engine would have to be performed and the results compared to the approximate solution before limits on the accuracy of the approximate solution could be established.

Whether the coolant flow is steady or non-steady, unless some external pressurization equipment is used, the pressure within the storage tank will decrease as coolant is expended. If the valve or pump cannot compensate for this pressure decrease, the approximate solution described above can be used to determine the thrust with the change in pressure as a function of time of operation factored into the calculation as a variable. Again, the accuracy of the performance is questionable until such time as the performance of a typical nuclear rocket is found using exact differential equations or from test data.

There are other problems involved in the design of nuclear rocket engines which at their present stage of development are rather academic. Questions have arisen concerning the possibility of heat transfer to and from the storage tank in space because of ambient temperature changes from the dark side to the sun side of the vehicle. Any heat transfer across the boundaries of the system (consisting of the vehicle and its component parts) will change the internal

thermodynamic relationships, hence the performance of the rocket engine. If the mission is such that the vehicle is not to be destroyed during reactor operation or immediately after shutdown, the problem of cooling the reactor after shutdown must be considered. This problem is more severe in a nuclear rocket operating in space than in any other vehicle powered by a heat transfer reactor. Thermal radiation in space is no more adequate for aftercooling the reactor than it is in the earth's atmosphere. Therefore, sufficient propellant must be carried to take care of these cooling requirements. If the vehicle has a particular objective in space there are two extreme possibilities of handling the aftercooling problem: 1) the reactor can be shut down at some intermediate point in the trajectory and the vehicle allowed to coast to its objective with ever decreasing thrust generated by the cooling ~~air~~^{flow} expanding through the nozzle at decreasing temperature, and 2) the reactor can maintain full power until the vehicle reaches its objective, then the decreasing thrust developed by the cooling flow can be dissipated in different directions with a net vector thrust of zero. The first method would not be entirely feasible because the time required to reach the objective would be too long. The second method could be handled by opening ports in the forward part of the rocket and expelling part of the cooling ~~air~~^{flow} through these to produce a reverse thrust equal and opposite to that generated by the exhaust nozzle expansion. The probable method of handling this problem will be some combination of these two extremes.