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Finite Area Combustor Theoretical Rocket Performance

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Sanford Gordon
Sanford Gordon & Associates
Cleveland, Ohio

and

Bonnie J. McBride
Lewis Research Center
Cleveland, Ohio

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NASA

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Sanford Gordon
Sanford Gordon & Associates
Cleveland, Ohio 44121

Bonnie J. McBride
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

ABSTRACT

Previous to this report, the computer program of NASA SP-273 and NASA TM-86885 was capable of calculating theoretical rocket performance based only on the assumption of an infinite area combustion chamber (IAC). An option has been added to this program which now also permits the calculation of rocket performance based on the assumption of a finite area combustion chamber (FAC). In the FAC model, the combustion process in the cylindrical chamber is assumed to be adiabatic, but nonisentropic. This results in a stagnation pressure drop from the injector face to the end of the chamber and a lower calculated performance for the FAC model than the IAC model.

INTRODUCTION

The calculation of theoretical rocket performance involves a number of assumptions. For the same propellant and operating conditions, theoretical performance can vary depending on which assumptions are used. Rocket performance calculated by the computer program of references 1 and 2 assumes adiabatic combustion in an infinite area combustion chamber (IAC) followed by isentropic expansion in the nozzle. In order to have a more realistic model, this supplement to references 1 and 2 presents an additional option for calculating rocket performance based on the assumption of adiabatic combustion in a finite area combustor (FAC) followed by isentropic expansion. Two input options are available for the FAC problem. Input option 1 is an assigned contraction ratio, while input option 2 is for an assigned mass flow per unit combustor area.

The addition of this new FAC option required changes in only two of the subroutines of the reference 1 program; namely, ROCKET and RKTOUT. A short description is given herein of the equations and iteration procedures. Three sample cases are given to facilitate the discussion of input options, output, and analysis of the effect on performance for the assumptions of FAC and IAC.

FINITE AREA COMBUSTION

Combustion in a rocket chamber is a nonisentropic, irreversible process. During the burning process, part of the energy released is used to raise the entropy, an undesirable form of energy inasmuch as it is unavailable to do work. This energy utilization loss is reflected in a total pressure drop as the gases are being accelerated from the beginning of the combustion process (at or near the injector face) to the end of the chamber. The combustion process may still be considered to be adiabatic; however, due to heat not being added at constant pressure during combustion, "the energy available for producing nozzle exit velocities is less than exists under ideal conditions of negligible chamber velocity." (ref. 3). Calculated rocket performance will therefore be less for the model of FAC than for the more commonly used ideal model of IAC.

A sketch of a rocket is given in figure 1. The positions in figure 1 are numbered in the same order as they appear in the performance output tables. The entrance to the finite chamber will be referred to as the injector face and will be indicated by 'inj' or '1' as subscript. The end of the finite chamber (nozzle entrance) will be indicated with the subscript '4' or 'c'. The infinite area position is indicated by subscript 'inf' or '2' while the throat is indicated by subscript 't' or '3'.

Equations

Unless otherwise stated, the International System of Units (SI Units) is used. The relationship of forces between points 1 and 4 for the nonisentropic process of combustion in a finite area is given by the following equation (ref. 3, p. 81)

$$\left(P + \frac{\dot{m}\,u}{A}\right)_1 = \left(P + \frac{\dot{m}\,u}{A}\right)_4 \tag{1}$$

where P is pressure, A is the combustor area, $\dot{m}$ is the mass flow rate, and u is velocity. Equation (1) may be written as

$$(P + \rho u^2)_1 = (P + \rho u^2)_4 \tag{2}$$

by using the continuity relationship

$$\dot{m} = \rho A u \tag{3}$$

where ρ is density.

When velocity at the injector face is negligible, equations (1) and (2) reduce to

$$P_1 = P_{inj} = \left(P + \frac{\dot{m}\, u}{A}\right)_4 = (P + \rho u^2)_4 \tag{4}$$

Iteration Procedure

An iteration procedure is required to satisfy Eq. 4. Two input options are available for FAC. In option 1, the contraction ratio $\frac{A_c}{A_t}$ is assigned. In option 2, the mass flow rate per unit combustor area $\frac{\dot{m}}{A_c}$ is assigned. The iteration procedure for option 1 is simpler and therefore will be described first. All of the first four points shown in figure 1 are involved in the iteration procedure. Thermodynamic parameters at point 1 are obtained by a combustion calculation (HP problem in reference 1). Starting with an estimated value for P_2, calculations are then made for points 2, 3, and 4 (the assigned contraction ratio) as would usually be done for infinite area combustion, throat, and an assigned area ratio as described in reference 1. A check is made to see if equation 4 is satisfied to within the following tolerance

$$\frac{|P_{inj} - (P + \rho u^2)_4|}{P_{inj}} \leq 2 \times 10^{-5} \tag{5}$$

If Eq. 5 is satisfied, then the calculations for the finite area combustor are complete for points 1, 2, 3 and 4. Calculations are then continued if other values of pressure ratio and/or area ratio have been specified in the input dataset. If Eq. 5 is not satisfied, then an improved estimate for P_2 is obtained as described in a later section and the procedure for points 2, 3 and 4 is repeated until Eq. 5 is satisfied.

A similar procedure is used for option 2 as was described for option 1. However, the contraction ratio is not known for option 2. Therefore, the iteration procedure involves starting with an estimated value for $\frac{A_c}{A_t}$ as well as for P_2 and then obtaining improved estimates for both P_2 and $\frac{A_c}{A_t}$. Not surprisingly, more iterations are required for option 2 than for option 1 which requires improved estimates for P_2 only. As in the case of option 1, iteration is complete when Eq. 5 is satisfied.

Initial Estimates

A curve is given in figure 3-18 of reference 3 which relates $\frac{P_2}{P_{inj}}$ with $\frac{A_c}{A_t}$ for an assumed value of $\gamma = 1.2$. The following empirical equation was derived by fitting three selected points read from the curve:

$$P_2 = P_{inj}\left[\frac{1.0257 - 1.2318\frac{A_c}{A_t}}{1 - 1.26505\frac{A_c}{A_t}}\right] \tag{6}$$

Eq. 6 is used only to obtain an initial estimate for P_2.

For option 1, the assigned value of $\frac{A_c}{A_t}$ is used in Eq. 6. For option 2, an initial estimate of $\frac{A_c}{A_t}$ is required (see Input Option Parameters section). This initial estimate is used in Eq. 6 to obtain a value for P_2, which is then used in the following equation to obtain an improved initial estimate for $\frac{A_c}{A_t}$

$$\frac{A_c}{A_t} = \frac{P_2}{2350\frac{\dot{m}}{A_c}} \tag{7}$$

Eq. 7 was derived by starting with the relationship for characteristic velocity $c^* = P_2\frac{A_t}{\dot{m}}$, multiplying both sides by A_c and using an arbitrary value of $c^* = 2350\ m/s$. Somewhat better initial estimates for both P_2 and $\frac{A_c}{A_t}$ are obtained by repeating several times the sequence of substituting values of $\frac{A_c}{A_t}$ from Eq. 7 into Eq. 6 and values of P_2 from Eq. 6 into Eq. 7. If the input value of $\frac{\dot{m}}{A_c}$ is so large that Eq. 7 calculates a value less than 1, the program will stop the calculations and print out the error message "INPUT VALUE OF MDOT/A = (value) IS TOO LARGE. GIVES CR ESTIMATE LESS THAN 1".

Improved Estimates

For option 1, an improved estimate for P_2 is obtained by assuming that the ratio of the assigned value of P_{inj} to the current value of P_{inj} (obtained by means of Eq. 4) is equal to the ratio of the final value of P_2 to the current value of P_2. This assumption leads to the following equation

$$P_{2,new} = P_2\frac{P_{inj,a}}{P_{inj}} \tag{8}$$

The use of Eq. 8 often gives such an excellent improved estimate for P_2 that it need be used only once to obtain convergence (Eq. 5).

For option 2, an improved estimate for $\frac{A_c}{A_t}$ is required in addition to the one for P_2 and is obtained from the following equation

$$\frac{A_c}{A_t} = \frac{\frac{\dot{m}}{A_t}}{\frac{\dot{m}}{A_c}} \tag{9}$$

Inasmuch as both P_2 and $\frac{A_c}{A_t}$ are changing, the iteration procedure is longer for option 2 than for option 1. For option 2, as well as for option 1, convergence is complete when Eq. 5 is satisfied.

Input Option Parameters

Two options are available for obtaining finite combustor area performance. In addition to the usual required input parameters described in ref. 1 for namelist &RKTINP, several additional parameters are required. For option 1 these are: FAC = T and ACAT = some assigned value for $\frac{A_c}{A_t}$. For option 2, the additional parameters are FAC = T and MA = some assigned value for $\frac{\dot{m}}{A_c}$. Option 2 also requires an initial estimate of $\frac{A_c}{A_t}$. A default value of ACAT = 2 is provided in the program for this initial estimate. However, if desired, a different initial estimate for ACAT may be included in the &RKTINP data. Thus, for option 2, a value of MA is required in &RKTINP, whereas an estimated value for ACAT is optional.

In FAC, the PCP values $\frac{P_{inf}}{P_e}$ are relative to the injector face pressure, whereas in IAC, the PCP values $\frac{P_{inj}}{P_e}$ are relative to the infinite area chamber pressure. Due to this definition of PCP, the assigned values of PCP must be larger than $\frac{P_{inj}}{P_{inf}}$. Otherwise, this will give values of P_e larger than P_{inf}, which is an impossible condition. For example, in table III, the value of $\frac{P_{inj}}{P_{inf}} = 1.0848$. If a value of PCP less than this had been assigned, 1.05 for example, this would have given a value of $P_e = \frac{P_{inj}}{PCP} = \frac{5331700}{1.05} = 5077810\ Pa$ which is more than the value of $P_{inf} = 4914900\ Pa$, an impossible condition. If impossible values of PCP are inadvertantly included in the input data set, these values will automatically be omitted by the program and the following error message printed: PRESSURE RATIO OF (value) GIVES PE GREATER THAN PINF. OMIT THIS POINT.

SAMPLE PROBLEMS

Three sample problems were selected, one for IAC (case 1) and two for FAC (cases 2 and 3), to illustrate some input and output features and to provide performance data for a comparison of results. The input datasets for these problems are given in table I and the output is given in tables II to IV. All sample problems are for the same propellant, o/f and chamber pressure. The propellant is H_2(l) at 20.17K and O_2(l) at 90.18K, o/f = 5.55157, and chamber pressure is 5331721 $\frac{N}{m^2}$ (52.62 atm). A number of assigned pressure ratios, PCP, and supersonic area ratios, SUPAR ($\frac{A_e}{A_t}$) are common to all problems. The PCP values selected are 10, 100, and 1000. The SUPAR values are 25, 50 and 75. In addition, the FAC case 2 has as assigned contraction ratio $\frac{A_c}{A_t}$ = 1.58 while the FAC case 3 has an assigned $\frac{\dot{m}}{A_c}$ = 1332.0. The value of $\frac{\dot{m}}{A_c}$ was calculated from the results of case 2 as follows: from table III, in the column for $\frac{A_c}{A_t}$ = 1.58, ρ = 2.0353 and $u = I_{sp}$ = 654.5. The product, ρu = 1332, is equal to $\frac{\dot{m}}{A_c}$ from the continuity relationship (Eq. 3). Case 3, therefore, should reproduce the case 2 contraction ratio of 1.58, which indeed it does. Cases 2 and 3 both have FAC = T. For comparison purposes, the IAC problem includes an assigned subsonic area ratio SUBAR = 1.58. In the FAC cases, the output column for the contraction ratio appears before the assigned pressure ratios, while in IAC, the SUBAR column appears after the assigned pressure ratio columns.

Output Format

The output format previously used for IAC has been somewhat revised to accomodate FAC. These revisions are as follows:

1. The first line of the output headings are the same for both cases and now read as follows: THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM COMPOSITION DURING EXPANSION. The second line for IAC reads FROM AN INFINITE AREA COMBUSTOR, while for FAC it reads FROM A FINITE AREA COMBUSTOR.

2. The line following the heading which gives chamber pressure in units of psia has been changed from PC to PINJ for FAC and PINF for IAC.

3. An additional line has been added for FAC which gives either MDOT/AC = (value) if the input data set contains an assigned value for $\frac{\dot{m}}{A_c}$ or AC/AT = (value) if input contains an assigned value for contraction ratio $\frac{A_c}{A_t}$.

4. An additional row of output has been added for FAC; namely, PINJ/P (ratio of pressure at the injector face to exit pressure $\frac{P_{inj}}{P_e}$).

5. The next row gives the ratio of pressure at infinite chamber area to exit pressure. The label PC/P formerly used for IAC for this row has now been changed to PINF/P.

6. The first four columns for FAC are INJECTOR, INF CHAM, THROAT, and CN RATIO for conditions at the injector face, infinite area chamber, throat and contraction ratio. The columns for conditions at the injector face and contraction ratio are two additional columns which have been added for FAC and do not appear for IAC.

7. When more than 13 columns of data are required, the first two columns of data are repeated on the second sheet of output data for IAC as before, while the first three columns of data are repeated for FAC.

8. For IAC, the option remains, as before, of calculating rocket performance based on the assumption of either equilibrium composition, frozen composition or both during expansion. For FAC, only the equilibrium option is permitted.

9. An option is provided to print intermediate output pertaining to the convergence process for $\frac{A_c}{A_t}$ or $\frac{\dot{m}}{A_c}$. This output is obtained by setting IDBUGF = 1 in namelist &RKTINP.

EFFECT ON PERFORMANCE

Table II presents rocket performance data for the infinite area combustor (case 1) while tables III and IV present similar data for the finite area combustor cases 2 and 3. As expected, the results in tables III and IV are identical (see discussion in SAMPLE PROBLEMS). Table V summarizes and compares some of the data in tables II and III. It may be noted in table V that for the same pressure ratios, the area ratios and specific impulse for the case of finite area combustor are less than for case of infinite area combustor. This is due to a loss in total pressure during the non-isentropic combustion from the injector face to the end of the finite combustor.

The term $1 - \frac{I_{ini}^2}{I_{inf}^2}$ represents the energy utilization loss due to this non-isentropic combustion. The energy utilization loss for this particular example (contraction ratio equal 1.58) is about 3.12% at a pressure ratio of 10 and about 0.62% at a pressure ratio of 1000. There are two general trends in energy utilization losses. The first trend, which was just illustrated, is that for the same contraction ratio, energy utilization losses decrease with increasing pressure ratios $\frac{P_{inf}}{P_e}$. The second trend, for which data are not given in this report, is that for the same pressure ratio $\frac{P_{inf}}{P_e}$, energy losses decrease with increasing contraction ratios.

The previous numerical comparisons of table V data are for the same pressure ratios for IAC and FAC. However, the area ratios are not the same. When the comparison in energy utilization loss is for the assumption of the same area ratios, the losses are negligible. For example, as may be seen in table V, for area ratios of 25, 50 and 75, the energy utilization loss is only 0.05% or less.

CONCLUDING REMARKS

Previous to this report, the computer program of ref. 1 permitted calculation of theoretical rocket performance based on combustion in an infinite area combustor. An option has now been added to this program that permits performance calculations based on the assumption of a finite area combustor. Calculations were made for a typical example ($H_2 - O_2$ propellant, contraction ratio of 1.58) based on the two assumptions of finite and infinite area combustion chambers in order to assess the size of energy utilization losses due to the nonisentropic combustion process in the finite area combustor. The comparison of an energy utilization loss term involving specific impulse was made at several assigned pressure ratios and several assigned area ratios. The comparison showed energy utilization losses of 0.6% to 3.0% for assigned pressure ratios of 1000 to 10 respectively, whereas for assigned area ratios of 25 to 75, the energy utilization losses were trivial (0.05% or less).

Further information on the code can be obtained from the authors. Contact COSMIC, The University of Georgia, Athens, Ga. 30602, concerning the availability of this program.

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APPENDIX – SYMBOLS

A	area, m^2
ACAT	symbol in program for contraction ratio
$\frac{A_c}{A_t}$	contraction ratio (ratio of finite chamber area to throat area)
$\frac{A_e}{A_t}$	ratio of nozzle exit area to throat area
c^*	characteristic velocity, $\frac{m}{sec}$
FAC	finite area combustor
HP	assigned enthalpy and pressure problem (combustion at constant pressure)
IAC	infinite area combustor
I_{sp}	specific impulse with exit and ambient pressures equal, $\frac{N}{kg/sec}$ or $\frac{m}{sec}$
MA	symbol in program for ratio of mass flow rate to chamber area, $\frac{\dot{m}}{A_c}$, $\frac{kg}{sec}m^2$
$\dot{m}$	mass flow rate, $\frac{kg}{sec}$
P	pressure, $\frac{N}{m^2}$
PCP	symbol in program for ratio of chamber pressure to exit pressure (For FAC, PCP = $\frac{P_{inj}}{P_e}$. For IAC, PCP = $\frac{P_{inf}}{P_e}$)
SUPAR	symbol in program for supersonic area ratio
u	velocity, $\frac{m}{sec}$
γ	ratio of specific heats
ρ	density, $\frac{kg}{m^3}$

Subscripts

a	assigned
c	combustor
e	exit
f	finite
i	infinite or ideal
inf	infinite
inj	injector
o/f	oxidant-to-fuel mass ratio
t	throat

REFERENCES

1. Gordon, Sanford; and McBride, Bonnie J.: Computer Program for Calculation of Complex Chemical Equilibrium Compositions, Rocket Performance, Incident and Reflected Shocks, and Chapman-Jouguet Detonations. NASA SP-273, 1971, and SP-273, Interim Revision, 1976.

2. Gordon, Sanford; McBride, Bonnie J.; and Zeleznik, Frank J.: Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications. Supplement 1 – Transport Properties. NASA TM 86885, 1984.

3. Sutton, George P.: Rocket Propulsion Elements. Second ed., John Wiley & Sons, Inc., 1956.

TABLE I. - INPUT FOR SAMPLE CASES

Case 1 Input. - Infinite Area Combustor

```
REACTANTS
H 2.                                        100.     -2154.  L   20.17 F
O 2.                                        100.     -3102.  L   90.18 O

NAMELISTS
 &INPT2 KASE=1,RKT=T,P=52.62,OF=T,MIX=5.55157,
  SIUNIT=T     &END
 &RKTINP SUBAR=1.58,
  PCP=10,100,1000,SUPAR=25,50,75     &END
```

Case 2 Input. - Finite Area Combustor, Option 1

```
REACTANTS
H 2.                                        100.     -2154.  L   20.17 F
O 2.                                        100.     -3102.  L   90.18 O

NAMELISTS
 &INPT2 KASE=2,RKT=T,P=52.62,OF=T,MIX=5.55157,
  SIUNIT=T     &END
 &RKTINP FAC=T,ACAT=1.58,
  PCP=10,100,1000,SUPAR=25,50,75     &END
```

Case 3 Input. - Finite Area Combustor, Option 2

```
REACTANTS
H 2.                                        100.     -2154.  L   20.17 F
O 2.                                        100.     -3102.  L   90.18 O

NAMELISTS
 &INPT2 KASE=3,RKT=T,P=52.62,OF=T,MIX=5.55157,
  SIUNIT=T     &END
 &RKTINP FAC=T,MA=1332.0,
  PCP=10,100,1000,SUPAR=25,50,75     &END
```

TABLE II. - THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM COMPOSITION DURING EXPANSION

FROM INFINITE AREA COMBUSTOR

PINF = 773.3 PSIA
CASE NO. 1

	CHEMICAL FORMULA	WT FRACTION (SEE NOTE)	ENERGY KJ/KG-MOL	STATE	TEMP DEG K
FUEL	H 2.00000	1.000000	-9012.332	L	20.17
OXIDANT	O 2.00000	1.000000	-12978.762	L	90.18

O/F= 5.5516 PERCENT FUEL= 15.2635 EQUIVALENCE RATIO= 1.4297 PHI= 1.4297

	CHAMBER	THROAT	EXIT	EXIT	EXIT	EXIT	EXIT	EXIT	EXIT
PINF/P	1.0000	1.7397	10.000	100.00	1000.00	1.1020	262.17	659.57	1131.38
P, MPA	5.3317	3.0647	0.53317	0.05332	0.00533	4.8380	0.02034	0.00808	0.00471
T, DEG K	3395.7	3196.6	2572.9	1756.6	1112.1	3360.4	1461.6	1213.6	1083.2
RHO, KG/CU M	2.3995 0	1.4802 0	3.2707-1	4.8208-2	7.6150-3	2.2043 0	2.2101-2	1.0580-2	6.9103-3
H, KJ/KG	-1026.10	-2214.14	-5444.06	-8577.70	-10632.1	-1240.69	-9552.21	-10327.6	-10717.4
U, KJ/KG	-3248.13	-4284.56	-7074.23	-9683.68	-11332.3	-3435.47	-10472.4	-11091.6	-11399.4
G, KJ/KG	-64354.4	-61828.8	-53427.2	-41336.8	-31373.0	-63910.6	-36810.7	-32960.5	-30919.4
S, KJ/(KG)(K)	18.6496	18.6496	18.6496	18.6496	18.6496	18.6496	18.6496	18.6496	18.6496
M, MOL WT	12.706	12.837	13.123	13.205	13.207	12.730	13.207	13.207	13.207
(DLV/DLP)T	-1.02061	-1.01508	-1.00326	-1.00005	-1.00000	-1.01960	-1.00000	-1.00000	-1.00000
(DLV/DLT)P	1.3750	1.2906	1.0761	1.0017	1.0000	1.3602	1.0001	1.0000	1.0000
CP, KJ/(KG)(K)	8.3873	7.5061	4.8310	3.4070	2.9632	8.2381	3.2096	3.0396	2.9405
GAMMA (S)	1.1453	1.1476	1.1745	1.2276	1.2698	1.1456	1.2441	1.2612	1.2724
SON VEL,M/SEC	1595.3	1541.5	1383.7	1165.2	942.9	1585.6	1070.0	981.6	931.5
MACH NUMBER	0.000	1.000	2.148	3.335	4.649	0.413	3.859	4.394	4.726

PERFORMANCE PARAMETERS

AE/AT	1.0000	2.3469	12.179	68.360	1.5800	25.000	50.000	75.000
CSTAR, M/SEC	2337	2337	2337	2337	2337	2337	2337	2337
CF	0.660	1.272	1.663	1.876	0.280	1.767	1.846	1.884
IVAC, M/SEC	2884.6	3520.9	4170.9	4542.9	4005.3	4352.3	4490.3	4557.5
ISP, M/SEC	1541.5	2972.5	3886.3	4383.2	655.1	4129.4	4313.1	4402.6

MOLE FRACTIONS

H	0.03442	0.02728	0.00811	0.00019	0.00000	0.03316	0.00001	0.00000	0.00000
HO2	0.00002	0.00001	0.00000	0.00000	0.00000	0.00002	0.00000	0.00000	0.00000
H2	0.29457	0.29401	0.29697	0.30043	0.30055	0.29441	0.30054	0.30055	0.30055
H2O	0.63378	0.65244	0.69045	0.69935	0.69945	0.63724	0.69945	0.69945	0.69945
H2O2	0.00001	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
O	0.00225	0.00131	0.00008	0.00000	0.00000	0.00206	0.00000	0.00000	0.00000
OH	0.03306	0.02379	0.00433	0.00003	0.00000	0.03135	0.00000	0.00000	0.00000
O2	0.00190	0.00115	0.00007	0.00000	0.00000	0.00176	0.00000	0.00000	0.00000

ADDITIONAL PRODUCTS WHICH WERE CONSIDERED BUT WHOSE MOLE FRACTIONS WERE LESS THAN 0.50000E-05 FOR ALL ASSIGNED CONDITIONS

O3 H2O(S) H2O(L)

NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS

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TABLE III. - THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM COMPOSITION DURING EXPANSION

FROM FINITE AREA COMBUSTOR

PINJ = 773.3 PSIA
AC/AT = 1.5800
CASE NO. 2

	CHEMICAL FORMULA	WT FRACTION (SEE NOTE)	ENERGY KJ/KG-MOL	STATE	TEMP DEG K
FUEL	H 2.00000	1.000000	-9012.332	L	20.17
OXIDANT	O 2.00000	1.000000	-12978.762	L	90.18

O/F= 5.5516 PERCENT FUEL= 15.2635 EQUIVALENCE RATIO= 1.4297 PHI= 1.4297

	INJECTOR	INF CHAM	THROAT	CN RATIO	EXIT	EXIT	EXIT	EXIT	EXIT	EXIT
PINJ/P	1.0000	1.0848	1.8868	1.1955	10.000	100.000	1000.00	283.87	714.18	1225.05
PINF/P	0.92182	1.0000	1.7393	1.1020	9.2182	92.182	921.82	261.68	658.35	1129.28
P, MPA	5.3317	4.9149	2.8258	4.4600	0.53317	0.05332	0.00533	0.01878	0.00747	0.00435
T, DEG K	3395.7	3387.6	3190.3	3352.6	2601.0	1784.2	1132.3	1463.1	1214.9	1084.4
RHO, KG/CU M	2.3995 0	2.2155 0	1.3666 0	2.0353 0	3.2325-1	4.7461-2	7.4796-3	2.0391-2	9.7609-3	6.3749-3
H, KJ/KG	-1026.10	-1026.10	-2211.87	-1240.26	-5306.21	-8483.37	-10572.3	-9547.52	-10323.7	-10713.9
U, KJ/KG	-3248.13	-3244.56	-4279.62	-3431.63	-6955.64	-9606.76	-11285.1	-10468.6	-11088.5	-11396.6
G, KJ/KG	-64354.4	-64383.5	-61880.1	-63943.6	-53952.3	-41852.2	-31748.9	-36911.2	-33045.2	-30995.8
S, KJ/(KG)(K)	18.6496	18.7029	18.7029	18.7029	18.7029	18.7029	18.7029	18.7029	18.7029	18.7029
M, MOL WT	12.706	12.696	12.828	12.720	13.111	13.205	13.207	13.207	13.207	13.207
(DLV/DLP)T	-1.02061	-1.02105	-1.01545	-1.02003	-1.00372	-1.00007	-1.00000	-1.00000	-1.00000	-1.00000
(DLV/DLT)P	1.3750	1.3837	1.2982	1.3687	1.0860	1.0021	1.0000	1.0001	1.0000	1.0000
CP, KJ/(KG)(K)	8.3873	8.5031	7.6085	8.3518	4.9770	3.4295	2.9787	3.2107	3.0406	2.9415
GAMMA (S)	1.1453	1.1447	1.1469	1.1449	1.1717	1.2260	1.2680	1.2440	1.2611	1.2723
SON VEL,M/SEC	1595.3	1593.6	1540.0	1584.0	1390.2	1173.6	950.7	1070.4	982.1	932.0
MACH NUMBER	0.000	0.000	1.000	0.413	2.105	3.291	4.596	3.857	4.391	4.723
PERFORMANCE PARAMETERS										
AE/AT			1.0000	1.5800	2.2253	11.482	64.394	25.000	50.000	75.000
CSTAR, M/SEC			2335	2335	2335	2335	2335	2335	2335	2335
CF			0.659	0.280	1.253	1.654	1.871	1.768	1.846	1.885
IVAC, M/SEC			2882.7	4002.8	3489.5	4152.8	4532.6	4351.4	4489.6	4556.9
ISP, M/SEC			1540.0	654.5	2925.8	3861.9	4369.5	4128.3	4312.2	4401.8
MOLE FRACTIONS										
H	0.03442	0.03515	0.02793	0.03387	0.00910	0.00024	0.00000	0.00001	0.00000	0.00000
HO2	0.00002	0.00002	0.00001	0.00002	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
H2	0.29457	0.29444	0.29387	0.29428	0.29661	0.30041	0.30055	0.30054	0.30055	0.30055
H2O	0.63378	0.63254	0.65140	0.63603	0.68910	0.69932	0.69945	0.69945	0.69945	0.69945
H2O2	0.00001	0.00001	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
O	0.00225	0.00233	0.00136	0.00214	0.00010	0.00000	0.00000	0.00000	0.00000	0.00000
OH	0.03306	0.03354	0.02423	0.03183	0.00501	0.00004	0.00000	0.00000	0.00000	0.00000
O2	0.00190	0.00197	0.00120	0.00182	0.00009	0.00000	0.00000	0.00000	0.00000	0.00000

ADDITIONAL PRODUCTS WHICH WERE CONSIDERED BUT WHOSE MOLE FRACTIONS WERE LESS THAN 0.50000E-05 FOR ALL ASSIGNED CONDITIONS

O3 H2O(S) H2O(L)

NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS

TABLE IV. - THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM COMPOSITION DURING EXPANSION

FROM FINITE AREA COMBUSTOR

PINJ = 773.3 PSIA
MDOT/AC = 1332.000 (KG/S)/M**2
CASE NO. 3

	CHEMICAL FORMULA	WT FRACTION (SEE NOTE)	ENERGY KJ/KG-MOL	STATE	TEMP DEG K
FUEL	H 2.00000	1.000000	-9012.332	L	20.17
OXIDANT	O 2.00000	1.000000	-12978.762	L	90.18

O/F= 5.5516 PERCENT FUEL= 15.2635 EQUIVALENCE RATIO= 1.4297 PHI= 1.4297

	INJECTOR	INF CHAM	THROAT	CN RATIO	EXIT	EXIT	EXIT	EXIT	EXIT	EXIT
PINJ/P	1.0000	1.0848	1.8868	1.1955	10.0000	100.000	1000.00	283.87	714.18	1225.05
PINF/P	0.92182	1.0000	1.7393	1.1020	9.2182	92.182	921.82	261.68	658.35	1129.28
P, MPA	5.3317	4.9149	2.8258	4.4600	0.53317	0.05332	0.00533	0.01878	0.00747	0.00435
T, DEG K	3395.7	3387.6	3190.3	3352.6	2601.0	1784.2	1132.3	1463.1	1214.9	1084.4
RHO, KG/CU M	2.3995 0	2.2155 0	1.3666 0	2.0352 0	3.2325-1	4.7461-2	7.4796-3	2.0391-2	9.7609-3	6.3749-3
H, KJ/KG	-1026.10	-1026.10	-2211.87	-1240.27	-5306.21	-8483.37	-10572.3	-9547.52	-10323.7	-10713.9
U, KJ/KG	-3248.13	-3244.56	-4279.62	-3431.63	-6955.64	-9606.76	-11285.1	-10468.6	-11088.5	-11396.6
G, KJ/KG	-64354.4	-64383.5	-61880.1	-63943.6	-53952.3	-41852.2	-31748.9	-36911.2	-33045.2	-30995.8
S, KJ/(KG)(K)	18.6496	18.7029	18.7029	18.7029	18.7029	18.7029	18.7029	18.7029	18.7029	18.7029
M, MOL WT	12.706	12.696	12.828	12.720	13.111	13.205	13.207	13.207	13.207	13.207
(DLV/DLP)T	-1.02061	-1.02105	-1.01545	-1.02003	-1.00372	-1.00007	-1.00000	-1.00000	-1.00000	-1.00000
(DLV/DLT)P	1.3750	1.3837	1.2982	1.3687	1.0860	1.0021	1.0000	1.0001	1.0000	1.0000
CP, KJ/(KG)(K)	8.3873	8.5031	7.6085	8.3518	4.9770	3.4295	2.9787	3.2107	3.0406	2.9415
GAMMA (S)	1.1453	1.1447	1.1469	1.1449	1.1717	1.2260	1.2680	1.2440	1.2611	1.2723
SON VEL,M/SEC	1595.3	1593.6	1540.0	1584.0	1390.2	1173.6	950.7	1070.4	982.1	932.0
MACH NUMBER	0.000	0.000	1.000	0.413	2.105	3.291	4.596	3.857	4.391	4.723

PERFORMANCE PARAMETERS

AE/AT			1.0000	1.5800	2.2253	11.482	64.394	25.000	50.000	75.000
CSTAR, M/SEC			2335	2335	2335	2335	2335	2335	2335	2335
CF			0.659	0.280	1.253	1.654	1.871	1.768	1.846	1.885
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ISP, M/SEC			1540.0	654.5	2925.8	3861.9	4369.5	4128.3	4312.2	4401.8

MOLE FRACTIONS

H	0.03442	0.03515	0.02793	0.03387	0.00910	0.00024	0.00000	0.00001	0.00000	0.00000
HO2	0.00002	0.00002	0.00001	0.00002	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
H2	0.29457	0.29444	0.29387	0.29428	0.29661	0.30041	0.30055	0.30054	0.30055	0.30055
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H2O2	0.00001	0.00001	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000	0.00000
O	0.00225	0.00233	0.00136	0.00214	0.00010	0.00000	0.00000	0.00000	0.00000	0.00000
OH	0.03306	0.03354	0.02423	0.03183	0.00501	0.00004	0.00000	0.00000	0.00000	0.00000
O2	0.00190	0.00197	0.00120	0.00182	0.00009	0.00000	0.00000	0.00000	0.00000	0.00000

ADDITIONAL PRODUCTS WHICH WERE CONSIDERED BUT WHOSE MOLE FRACTIONS WERE LESS THAN 0.50000E-05 FOR ALL ASSIGNED CONDITIONS

O3 H2O(S) H2O(L)

NOTE. WEIGHT FRACTION OF FUEL IN TOTAL FUELS AND OF OXIDANT IN TOTAL OXIDANTS

TABLE V. COMPARISON OF SPECIFIC IMPULSE OBTAINED UNDER ASSUMPTION OF EXPANSION FROM FINITE AND INFINITE COMBUSTION CHAMBERS

Infinite chamber area			Finite chamber area			Energy Utilization loss due to finite chamber
$\frac{P_{inj}}{P_e}$	I_{sp} $(\frac{m}{sec})$	$\frac{A_e}{A_t}$	$\frac{P_{inj}}{P_e}$	I_{sp} $(\frac{m}{sec})$	$\frac{A_e}{A_t}$	$1 - \frac{I_f^2}{I_i^2}$
10.00	2972.5	2.3469	10.00	2925.8	2.2253	0.0312
100.00	3886.3	12.179	100.00	3861.9	11.482	0.0125
1000.00	4383.2	68.360	1000.00	4369.5	64.394	0.0062
262.17	4129.4	25.	283.25	4128.3	25.	0.0005
659.57	4313.1	50.	714.18	4312.2	50.	0.0004
1131.38	4402.6	75.	1225.05	4401.8	75.	0.0004

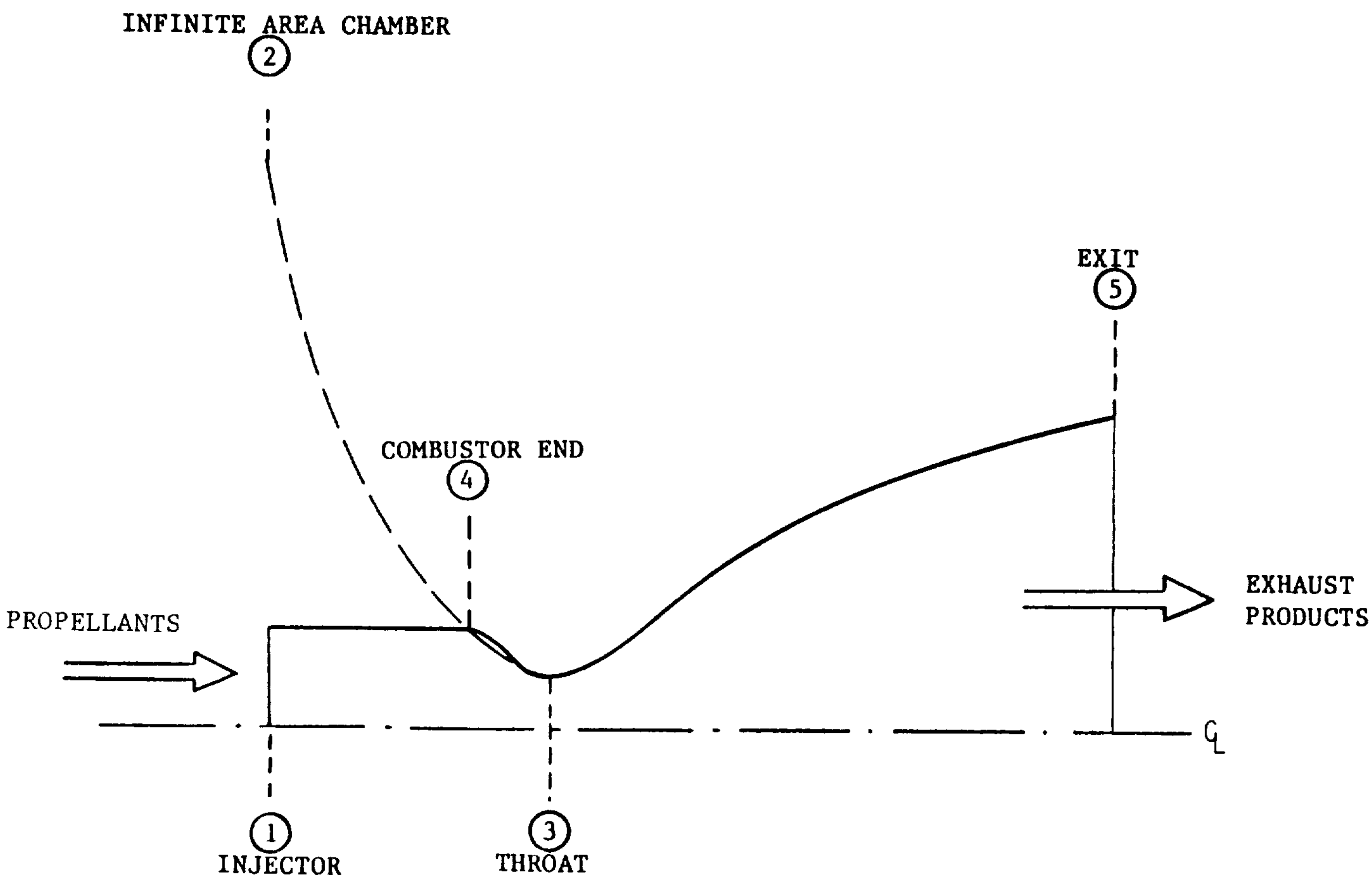

FIGURE 1. - THRUST CHAMBER SCHEMATIC WITH POSITIONS LABELED AS THEY APPEAR IN PROGRAM OUTPUT.

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Sanford Gordon, Sanford Gordon & Associates, Cleveland, Ohio 44121, subcontractor to Sverdrup Technology, Inc. (Subcontract No. 5312-80, Kenneth Davidian, Monitor). Bonnie J. McBride, NASA Lewis Research Center.

16. Abstract

Previous to this report, the computer program of NASA SP-273 and NASA TM-86885 was capable of calculating theoretical rocket performance based only on the assumption of an infinite area combustion chamber (IAC). An option has been added to this program which now also permits the calculation of rocket performance based on the assumption of a finite area combustion chamber (FAC). In the FAC model, the combustion process in the cylindrical chamber is assumed to be adiabatic, but nonisentropic. This results in a stagnation pressure drop from the injector face to the end of the chamber and a lower calculated performance for the FAC model than the IAC model.

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