Defence Science Journal, Vol 46, No 5, November 1996, pp. 337-345

Regression Rate Study in HTPB/GOX Hybrid Rocket Motors

Philmon George and S.Krishnan
Indian Institute of Technology, Madras-600 036

Lalitha Ramachandran
Liquid Propulsion System Centre, Thiruvananthapuram-695 547

and

P.M.Varkey and M.Raveendran
Vikram Sarabhai Space Centre, Thiruvananthapuram-695 022

ABSTRACT

The theoretical and experimental studies on hybrid rocket motor combustion research are briefly reviewed and the need for a clear understanding of hybrid rocket fuel regression rate mechanism is brought out. A test facility established at the Indian Institute of Technology, Madras, for hybrid rocket motor research study is described.The results of an experimental study on hydroxyl terminated polybutadiene and gaseous oxygen hybrid rocket motor are presented. Fuel grains with ammonium perchlorate "additive" have shown enhanced oxidizer mass flux dependence. Smaller grains have higher regression rates than those of the larger ones.

NOMENCLATURE

A_t	Nozzle throat area
c^*	Characteristic velocity
C_d	Coefficient of discharge
D	Grain port diameter
G	Total mass flux
G_0	Oxidizer mass flux
L	Grain length
$\dot{m}$	Mass flow rate
p	Pressure
$\dot{r}$	Regression rate
R	Gas constant of oxygen
T	Oxygen temperature
Δt	Time increment
ϕ	Oxidizer-fuel mixture ratio
ρ	Fuel density
γ	Specific heat ratio of oxygen

Subscript

c	Aft combustion-chamber
f	Fuel
i	Igniter
o	Oxidizer
s	Sonic nozzle

1. INTRODUCTION

Rocket propulsion system in which one of the two propellant components (oxidizer and fuel) is in liquid phase and the other is in solid phase is called hybrid rocket motor. The combination of solid fuel and liquid oxidizer is the most common one. Even though the idea of hybrid rocket engine was conceived as early as 1937, it was not till the 1960's that they were built successfully with good combustion efficiency. One of the reasons for

Received 02 September 1996

this had been the insufficient understanding of the combustion mechanism in the engine. During this period almost every aerospace propulsion company in USA and the five European countries vij. Germany, France, Sweden, Italy, and the Netherlands, experimented with various types of hybrids. After a lull of about two decades in hybrid rocket research, the growing interest in recent years towards lower developmental and operational costs without much losses in specific impulse and density, specific impulse values, safer operational characteristics, and better environment-friendly exhaust, has motivated many research activities in hybrid rockets for launch vehicle and other applications.[1,2]

The dependence of solid fuel regression rate on various operating conditions is the most important research aspect in the study of hybrid rocket. In USA during the 1960's, Chemical Systems Division of United Technologies Centre and Lockheed Propulsion Company have conducted fundamental investigations on the combustion behaviour and internal ballistics in hybrid rocket motors. Marxman and co-workers[3,4] of the United Technologies Centre, developed a turbulent boundary layer regression rate model under the assumption that the regression rate of solid fuel was controlled by the heat transfer to the fuel from the flame zone existing within the turbulent boundary layer. The important results of this study are that the regression rate of hybrid fuel is 1) a strong function of total mass flux ($\dot{r} \propto G^{0.8}$), 2) a weak function of the properties of fuel and oxidizer (heat of reaction and heat of gasification), and 3) independence of pressure over the range of practical interest of rocket engines — the combustion being diffusion limited. Smoot and Price[5-7] of Lockheed Propulsion Company made significant contributions to the data on hybrid fuel combustion and demonstrated the dependence of regression rate on oxidizer mass flux as well as pressure. They conducted experiments in a two dimensional burner using fuel blocks of width 25 mm and length up to 380 mm; Butyl rubber, polyurethane, and butyl rubber with lithium hydride were the fuels used and the two oxidizers used were the mixtures of fluorine-oxygen and fluorine-nitrogen. The measurements using the first oxidizer at different pressures (2.4-12 bar) and oxidizer mass fluxes ($10 \leq G_o \leq 120$ kg/m^{2}s) showed that 1) at low oxidizer mass fluxes the *average* (spatial and temporal) regression rate is G_o dependent ($\dot{r} \propto G_o^{0.8}$) but pressure independent; 2) at high oxidizer mass fluxes it is G_o-independent but pressure dependent (r increases with P); and 3) at intermediate oxidizer mass fluxes, however, the regression rate is dependent on oxidizer mass flux as well as pressure. They attributed the pressure dependence to the rate limiting chemical kinetic processes, possibly heterogeneous in nature. Their theoretical results considering condensed phase reactions for fuels containing lithium hydride compared well with their experimental results, only at low oxidizer mass fluxes. Muzzy,[8] after discounting the possibility of rate limiting heterogeneous chemical kinetic processes at gas-solid interface, considered the influence of chemical kinetic processes in gas phase that cannot be neglected for low pressure and/or high total mass flux conditions and showed that the regression rate is not a strong function of total mass flux and is dependent on pressure ($\dot{r} \propto G^{0.4} . P^{0.5}$); Further, the effects of fuel grain and oxidizer temperatures on the fuel regression rate were experimentally found to be negligible. Krier and Kerzner[9] applied laminar boundary layer theory to obtain the regression rates of hybrid fuels. This theory leads to a total mass flux index of 0.5 instead of 0.8 of turbulent boundary layer theory[3,4]. Rastogi and co-workers[10,11] showed that the combustion process in hybrid motor could be divided into two regions: the first one, at the entrance, governed by the fuel ablation and heterogeneous combustion at interface; and the second one, downstream, characterized by homogeneous gas phase reaction within the turbulent boundary layer. Paul *et al*[12] re-examined the turbulent boundary layer regression rate model of Marxman and co-workers and, after accounting for the density variation across the boundary layer, improved it by the better estimation of the "blocking effects"— the effects of transpiration on skin friction factor and heat transfer coefficient.

The renewed interest in recent years in hybrid rocket propulsion may be gauged by the spurt in the number of papers in the last few Joint Propulsion Conferences of AIAA. Strand, *et al*[13] studied the solid fuel regression rate in a hybrid combustion model of rigour higher than that of the turbulent boundary layer regression rate model of 1960's to confirm the turbulent boundary layer heat and mass transfer, as the rate limiting process for hybrid fuel decomposition and combustion for rocket operating pressures. Lewin *et al.*[14] through their experimental study in a hydroxyl terminated polybutadiene (HTPB)/gaseous oxygen

(GOX) hybrid motor showed that the calculated *instantaneous* regression rate was G_o dependent and pressure independent over a G_o range of 11-400 kg/m^{2}s and a pressure range of 8-23 bar. This observation matches with that of Smoot and Price only for their low oxidizer mass fluxes ($G_o \leq 15$ kg/m^{2}s). Kuo and co-workers[15] from their HTPB/GOX two dimensional hybrid motor experimental study ($G_o \leq 338$ kg/m^{2}s; $13 \leq p \leq 90$ bar) concluded that solid fuel surface temperature was 950-1000 K and the *average* regression rate increased continuously in the axial direction, possibly due to the increased mass flux of the core flow. Their *instantaneous* regression rate data of solid fuel over different portions of fuel slab through real-time X-ray radiography and ultrasonic pulse-echo techniques are the valuable inputs for hybrid rocket combustor modelling and design.

Thus, we see that the regression rate of solid fuel in hybrid rocket, the most important design parameter, is dependent on many operating variables such as mass flux, pressure, fuel port geometry, and fuel composition but mostly independent of initial temperatures of fuel and oxidizer. Among these operating variables, mass flux and pressure are the major ones that determine the regression rate and the widely used regression rate correlation is, $\dot{r} = aG^n p^m$. The values of n and m significantly change depending on the model conditions viz. 1) turbulent boundary layer heat transfer with diffusion controlled flame ($n = 0.8$ and $m = 0$)[3,4]; 2) turbulent boundary layer heat transfer with gas phase chemical kinetics controlled flame ($n = 0.4$ and $m = 0.5$)[8]; or 3) laminar boundary layer heat transfer with diffusion controlled flame ($n = 0.5$ and $m = 0$)[9]. Smoot and Price through their experimental studies found the average regression rate to be 1) mass flux dependent and pressure independent ($n = 0.8$ and $m = 0$) for low mass flux conditions; 2) mass flux independent and pressure dependent ($n = 0$ and $m > 0$) for high mass flux conditions; and 3) mass flux as well as pressure dependent ($n > 0$ and $m > 0$) for intermediate mass flux conditions. While the results of the low mass flux conditions can be explained by the first model conditions and the results of the intermediate mass flux conditions can be similarly explained by the second model conditions; the results of high mass flux conditions, however, cannot be explained by any of the above. Lewin *et al*[14], found that for mass flux and pressure conditions much higher than those of Smoot and Price[5-7], their calculated instantaneous regression rate to be mass flux dependent and pressure independent ($n = 0.55$-0.65 and $m = 0$) and this can be explained by turbulent boundary layer heat transfer with diffusion controlled flame model. They, however, noticed that the value of n is significantly different from 0.8. Thus, notwithstanding the recognition of mass flux and pressure as the most important variables affecting fuel regression rate, a clear understanding of the boundary layer combustion processes in hybrid rocket is still elusive.

The low regression rate of solid fuel is the basic problem which degrades the performance of hybrid rocket. Addressing this issue, Korting *et al*[16], conducted experimental study with polymetylmethacrylate and polyethylene as fuels, and oxygen and oxygen-nitrogen mixtures as oxidizers. One of their important observations is that a rearward facing step has a noticeable effect on combustion behaviour —increasing the *mean* (spatial) regression rate and changing the profile of the burned fuel grain. Through another study, Strand *et al*,[17], supported the inclusion of particulate additives (aluminium and/or coal) in solid fuel as an approach to enhance fuel regression rate with decreased mass flux dependency. They argued that the heat transfer by radiation from the particulate additives was a contributor for the enhancement. Lewin *et al*,[14] by their experimental study in a HTPB/GOX hybrid motor showed that fuel grains of shorter length (89 mm) had a higher regression rate than that of longer length (203 mm).

Considering the background detailed above, in order to aid in the development of large scale hybrid rocket motors, an experimental research programme has been initiated by the Indian Institute of Technology (Madras) [IIT(M)] with the objectives of enhancing our understanding of the boundary layer combustion processes in hybrid motors and developing solid fuel grains with enhanced regression rate.

2. EXPERIMENTAL SET-UP & PROCEDURE

A high pressure hybrid rocket motor test facility has been designed and assembled at IIT(M), (Fig. 1). The ability to carry out tests under high motor operating oxidizer mass fluxes up to 600 kg/m^{2}s is the speciality of the facility. The fuel and oxidizer used in this study are HTPB and GOX. The oxygen supply is from a bank of cylinders kept at a maximum pressure of 150 bar. A ball valve is used to initiate and terminate the flow of oxygen and a sonic nozzle in the line maintains steady mass flow rate. Nitrogen is used as

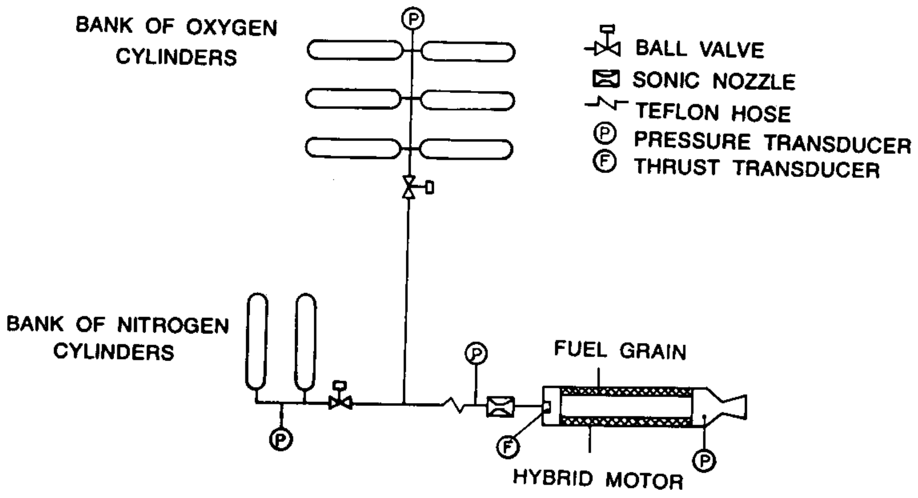

Figure 1. Schematic diagram of hybrid rocket test facility.

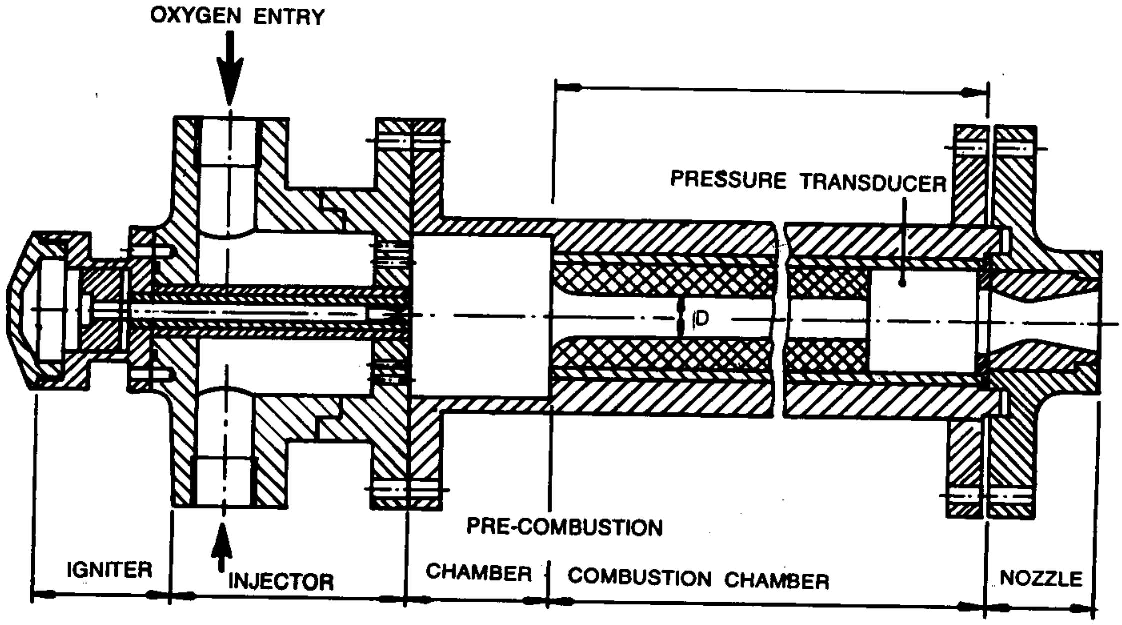

Figure 2. Hybrid rocket motor assembly.

purge gas to terminate combustion after the desired burning time. The pressure maintained upstream of sonic nozzle is always higher than two times the maximum combustion chamber pressure, so that the oxidizer mass flow rate is constant during the test. Oxygen supply to motor injector is through two 20 mm flexible stainless steel wire braided teflon hoses. A high pressure (40 bar) axisymmetric subscale proof motor assembly has been designed and fabricated for this investigation, (Fig. 2). This assembly consists of an injector, a pre-combustion chamber, a combustion chamber with grain and aft chamber, and a nozzle assembly with interchangeable nozzles. The diameter and length of the pre-combustion chamber are 75 and 50 mm respectively. The hybrid combustion chamber is a steel case having an outer diameter of 88.9 mm and inner diameter of 58.4 mm; Combustion chambers of two different lengths (380 and 580 mm) are used. A pyrogen type igniter initiates combustion.

The solid fuel used in this study is HTPB. A small amount of carbon black powder (*C*) is mixed with the fuel. In order to improve the mechanical property of the grain, trimethylol propane (TMP) is also added, and toluene di-isocyanate (TDI) is used for curing. The preparation of fuel grain is as follows:- The pre-mix containing 800 g of HTPB, 8 g of *C*, and 40 g of TMP is thoroughly mixed for 10 min. Then on adding 131.6 g of TDI, the blend is mixed for further 10 min. Thus, the percentage composition of the fuel is HTPB:C:TMP:TDI =81.68:0.82:4.08:13.42. The mixture is now poured into the mould of cylindrical fuel grain and cured, first for three days at room temperature and then for four days at 60 °C. The fuel grains of outer diameter 46.4 mm are in two sizes; length/port-diameter = 250 mm/12 mm and 400 mm/20 mm. Two fuel compositions were studied in the present investigation; the first one had the above base composition and the second one had 7.55 per cent of fine (10 μ m) ammonium perchlorate (AP) of the total *mix* — retaining the above base composition in the balance blend. Before bonding the cured grain to the combustion chamber using epoxy, the chamber inner wall is lined with two layers of 2 mm thick "rocasin" insulator. For initial tests, however, silica phenolic liners were used; Because of handling problems, these have since been satisfactorily replaced with rocasin ones.

Each test run includes the following steps. By suitable setting of the ball valve opening, constant supply pressure of GOX is maintained upstream of the sonic nozzle. After the flow becomes steady, the igniter train is initiated. After the desired burn time, oxygen supply is cut off and nitrogen purge is opened to extinguish combustion. The test measurements are pressures at upstream and downstream of sonic nozzle, aft combustion-chamber, combustion chamber of the pyrogen igniter, and thrust. All the signals from strain gauge type pressure and thrust transducers are recorded in an IBM-PC(AT) using a data acquisition system. The other measurements are fuel density, ambient temperature, initial and final nozzle throat diameters, and initial and final fuel grain mass. For each test, the desired flow rate can be obtained by choosing the appropriate sonic nozzle throat and its upstream pressure. The oxidizer flow rate is determined using the sonic nozzle throat upstream pressure, oxygen temperature (atmospheric temperature), and the calibrated coefficient of discharge of the sonic nozzle.

3. REGRESSION RATE CODE

A code was developed for hybrid rocket motor in order to calculate the fuel regression rate and other performance parameters as functions of time. With respect to time, the measured values of sonic nozzle upstream pressure (p_s), igniter pressure (p_i), aft combustion-chamber pressure (p_c), and thrust are used for the analysis. At the initial time step, the oxygen mass flow rate through sonic no

$$\dot{m}_o = c_{d_s} p_s A_{t_s} / (\sqrt{RT}/\Gamma) \tag{1}$$

where $\Gamma = \sqrt{\gamma}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}$

The igniter mass flow rate,

$$\dot{m}_i = c_{d_i} p_i A_{t_i} / c_i^* \tag{2}$$

The total mass flow rate through hybrid motor nozzle,

$$\dot{m}_t = p_c A_t / c_{eff}^* \tag{3}$$

where C_{eff}, the effective characteristic velocity of *total* combustion products of igniter and hybrid combustion chamber, is calculated as,[18]

$$c_{eff}^* = \frac{c_i^* \dot{m}_i + \eta\, c_c^* \dot{m}_o \left(\frac{\varphi+1}{\varphi}\right)}{\dot{m}_i + \dot{m}_o \left(\frac{\varphi+1}{\varphi}\right)} \tag{4}$$

The fuel mass flow rate,

$$\dot{m}_f = \dot{m}_t - (\dot{m}_o + \dot{m}_i) \tag{5}$$

C_c^* a theoretical quantity — a strong function of ϕ but weak function of pressure, — is calculated using a standard complex chemical equilibrium calculation procedure.[19] Using Eqns (3)-(5), starting from a trial value of η, $\dot{m}_f$ is obtained iteratively imposing convergence on ϕ . Then the specific impulse is calculated by dividing the measured thrust by $\dot{m}_f$ obtained for the converged ϕ. Assuming that the fuel burns uniformly along the axis, the fuel regression rate,

$$\dot{r} = \frac{\dot{m}_f}{\pi D L \rho} \tag{6}$$

The total mass flux and the oxidizer mass flux are calculated by dividing the mass flow rates by the corresponding port area. Under quasi-steady state assumption, the port diameter at the next time step,

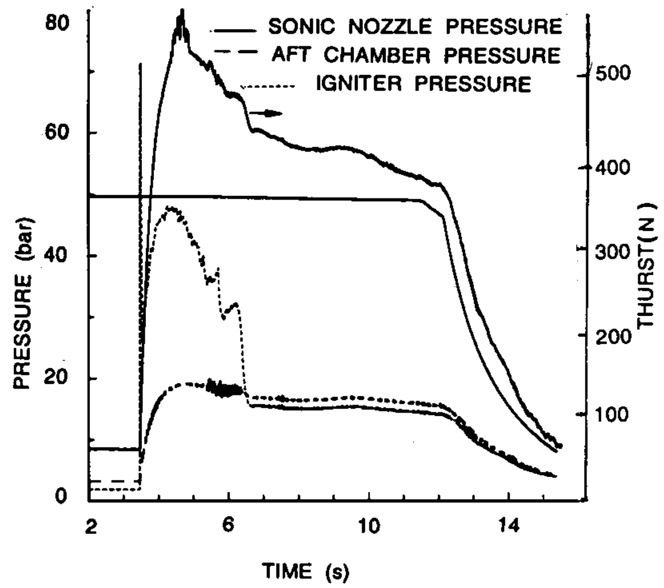

Figure 3. Pressure and thrust-time traces for Test No. 3.

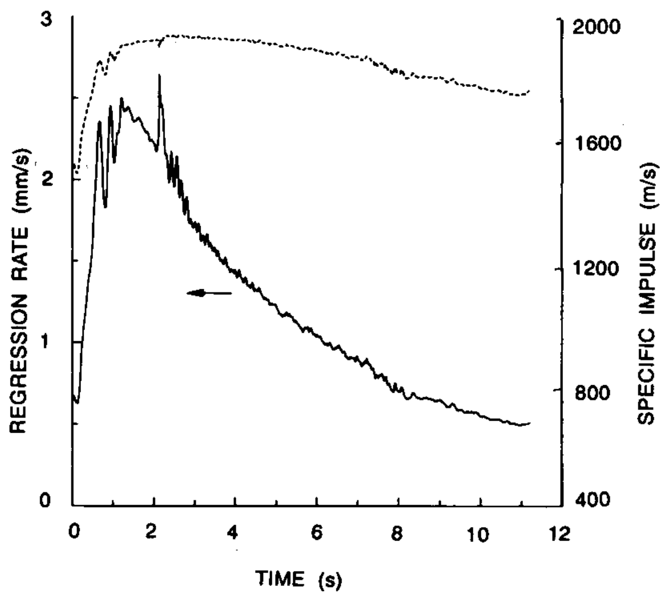

Figure 4. Calculated regression rate and specific impulse values for Test No. 2.

$$D_i = D_{i-1} + 2\,\dot{r}\,\Delta t \qquad (7)$$

Thus the calculations are carried out incrementally for the duration of test run in order to estimate the total fuel mass consumed and to compare it with the actual one. If they are not equal within the allowable error, assuming then a new trial value of η, calculations are repeated till convergence on total fuel mass consumed is reached.

4. RESULTS & DISCUSSION

The test conditions of six successful firings of the hybrid motor out of the eight conducted in this study are summarised in Table 1. The pressure ranges shown in Table 1 are with respect to the maximum chamber pressure attained during ignition and the minimum pressure before oxygen cutoff. The high pressure drop in the Test No. 2 is due to the graphite nozzle throat erosion. As excessive throat erosion was experienced during Test Nos 1 and 2, the graphite nozzles have since been replaced satisfactorily with heavy copper-nozzles. In all the tests, the oxidizer mass flow rates were maintained constant and the initial and final oxidizer mass fluxes, found by dividing the mass flow rates by initial and final grain port areas, are given in the third column. The average fuel regression rates were calculated, noting the initial and final masses as well as dimensions of the fuel grain. These two agree within a maximum difference of six per cent. The regression rate values calculated by the mass measurement method is always less than that through the other method; The values through the mass measurement method are given in the last column of Table 1.

The fuel grains were visually examined after the firing. The port diameter measurements are given in Table 2; As the grains of Test Nos 3 and 4 could not be retrieved without damage, their measurements are not given in the table. It appears that shorter grain (250 mm) burns faster at the fore end; However, this inference should be taken with caution as only one set of data is available for longer grain.

Table 1. Hybrid motor test conditions

Test No.	Pressure (bar)	Oxidizer mass flux (Kg/m^2s)	Initial port dia. (mm)	Grain length (mm)	Test duration (s)	Average regression rate (mm/s)
2[a]	20.3-10.4	530.9-23.9	12	250	11.78	1.23
3[a]	19.0-15.8	478.5-43.0	20	400	8.54	1.38
4[a]	24.3-18.2	478.5-62.3	20	400	5.56	1.79
5	16.1-12.2	286.4-71.6	20	400	7.29	1.38
6	24.5-18.1	467.3-28.3	12	250	10.58	1.37
8	22.0-16.7	397.8-30.0	12	250	7.68	1.67

[a] Fuel grain contains 7.55 per cent of AP fine (10 μm).

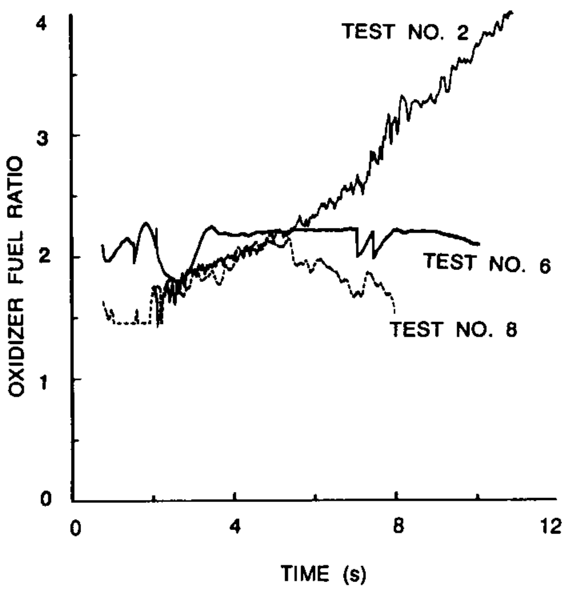

Figure 5. Oxidizer-fuel ratios for Test Nos 2, 6, and 8.

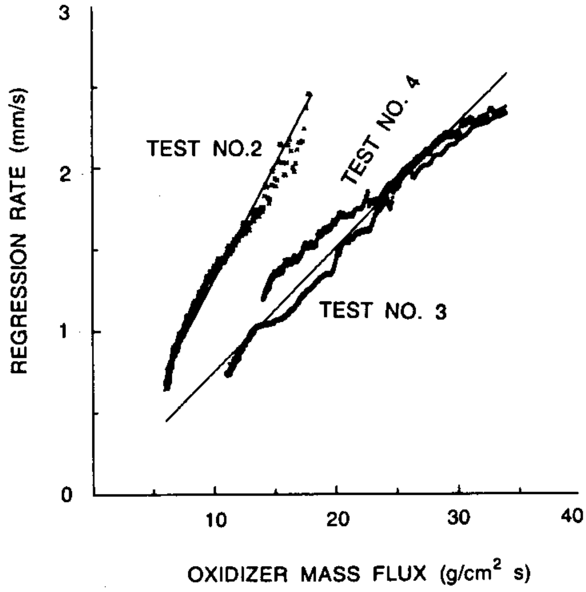

Figure 6. Regression rates for Test Nos 2, 3, and 4.

Table 2. Grain port measurements

Test No.	Injector end dia. (mm)	Nozzle end dia. (mm)	Length/port initial dimensions (mm/mm)
2	41.33	40.70	250/12
5	39.79	42.18	400/20
6	43.31	41.63	250/12
8	40.61	37.90	250/12

As a typical example, the combined firing-record of Test No. 3 is given in Fig. 3. The igniter operates for a duration of about 3s but the maximum chamber-pressure is reached well ahead of 1.5s. The equilibrium chamber operation sets in after the duration of the igniter operation. During the igniter operation, evidently due to the significant igniter mass flow contribution to the total flow, the chamber pressure is about 2 bar higher than the equilibrium chamber-pressure. As the present igniter is found to be rather "strong", future tests are planned with an igniter burn duration of around 1s. The chamber pressure oscillations seen, (Fig. 3), appear to have been caused by igniter pressure peaks in the corresponding period. However, even during equilibrium chamber operation for certain other tests, low frequency (30 Hz) low amplitude (10 per cent of average pressure) oscillations were noticed. With a view to suppressing these oscillations, sonic nozzle was shifted just upstream of injector[15]; This, however, did not improve the situation.

As a representative case, the calculated instantaneous regression rate and the specific impulse values from experimental results of Test No. 2 are shown in Fig. 4. After reaching a peak value during ignition, the regression rate is continuously decreasing due to the decrease in oxidizer mass flux. The large initial fluctuations are due to the pressure peaks experienced during ignition transient. The above calculated specific impulse value works out to be about 90 per cent of the theoretical value. The variations of mixture ratio with time for Test Nos 2, 6 and 8 are shown in Fig. 5. For a cylindrical grain port receiving constant oxidizer mass flow, theoretically it can be shown that for the regression rate law $\dot{r} = a\ G_o^n$ the oxidizer-fuel mixture ratio ϕ will increase with time for $n > 0.5$ and decrease for $n < 0.5$. The variations on ϕ (Fig. 5) indicate that the fuel grain with AP has a higher n value than the one without AP.

In order to find the effect of grain dimensions, the calculated regression rates with respect to oxidizer mass flux for Tests Nos 2, 3 and 4 are shown in Fig. 6. The obtained data point out that a smaller grain has higher regression rate than the larger one and this agrees with the results of Lewin *et al*[14]. The regression rate fits for different grain dimensions are:

$\dot{r} = 0.117\ G_o^{1.05}$ mm/s $\qquad G_o = [g/(cm^2 s)]$
(small grain, 12 mm/250 mm)

and

$\dot{r} = 0.074\ G_o^{1.006}$ mm/s G_o = [g/(cm^{2}s)]
(large grain, 20 mm/400 mm).

CONCLUSION

1. A high pressure hybrid rocket motor test facility has been designed and assembled at IIT(M) with the assistance of Indian Space Research Organisation. The facility has been successfully used for HTPB/GOX hybrid rocket study.

2. Initial test results with fuel grains with different compositions indicate that the fuel grains with AP have stronger dependence on oxidizer mass flux.

3. Grains of smaller dimensions have higher regression rates than those of larger ones.

ACKNOWLEDGEMENT

The study reported forms a part of the research sponsored by the Indian Space Research Organisation through the ISRO-IIT(M) Space Technology Cell at the Indian Institute of Technology, Madras.

REFRENCES

1. Dornheim, M.A. AMROC Hybrid motor tests aimed at 1995 flight. Aviation Week & Space Technology. 1993. 51.

2. Anon. Hybrid rocket propulsion. Report of an AIAA Workshop. 25-26 July 1995, Washington DC, USA, 1995.

3. Marxman, G.A.; Wooldridge, C.E. & Muzzy, R.J. Fundamentals of hybrid boundary layer combustion. Progress in Astronautics and Aeronautics, Vol 15; Heterogeneous Combustion, edited by Wolfhard, H.G.; Glassman, I. & Green, Jr. L. Academic Press, New York, 1964, pp. 485-522.

4. Marxman, G.A. Boundary layer combustion in propulsion. Proceedings of the Eleventh Symposium (International) on Combustion, 1967, The Combustion Institute. Pittsburg, Pennsylvania, 1967. pp. 269-89.

5. Smoot, L.D. & Price, C.F. Regression rates of nonmetallized hybrid fuel systems. *AIAA J*, 1965, 3 (8), 1408-13.

6. Smoot, L.D. & Price, C.F. Regression rates of metallized hybrid fuel systems. *AIAA J*, 1966, **4** (5), 910-15.

7. Smoot, L.D. & Price, C.F. Pressure dependence of hybrid fuel regression rates. *AIAA J*, 1967, **5** (1), 102-06.

8. Muzzy, R.J. Applied hybrid combustion theory. Paper presented at the AIAA/SAE 8th Joint Propulsion Conference, 29 Nov.-1 Dec 1972, New Orleans, Louisiana, 1972. AIAA Paper No. 72-1143.

9. Krier, H. & Kerzner, H. Analysis of chemically reacting laminar boundary layer during hybrid combustion. *AIAA J*, 1973, **11** (8), 1691-98.

10. Rastogi, R.P.; Kishore, K. & Chaturvedi, B.K. Heterogeneous combustion studies on polystyrene and oxygen styrene copolymer. *AIAA J*, 1974, **12** (9), 1187-92.

11. Rastogi, R.P. & Desh Deepak. Pressure dependence of hybrid fuel burning rate. *AIAA J*, 1976, **14** (7), 988-90.

12. Paul, P.J.; Mukunda, H.S. & Jain, V.K. Regression rates in boundary layer combustion. Proceedings of the Nineteenth Symposium (International) on Combustion, 1982. The Combustion Institute, Pittsburg, Pennsylvania, 1982. pp. 717-29.

13. Strand, L.D.; Ray, R.L.; Anderson, F.A. & Cohen, N.S. Hybrid rocket fuel combustion and regression rate study. Paper presented at the AIAA/SAE/ASME/ASEE 28th Joint Propulsion Conference, 6-8 July 1992, Nashville, TN, 1992. AIAA Paper No. 92-3302.

14. Lewin, A. *et al.* Experimental determination of performance parameters for a polybutadiene/ oxygen hybrid rocket. Paper presented at the AIAA/SAE/ASME/ASEE 28th Joint Propulsion Conference, 6-8 July 1992, Nashville, TN, 1992. AIAA Paper No. 92-3590.

15. Chiaverini, M.J. *et al.* Fuel decomposition and boundary-layer combustion processes of hybrid rocket motor. Paper presented at the AIAA/SAE/ASME/ASEE 31st Joint Propulsion Conference, 10-12 July 1995, San Diego, CA, 1995. AIAA Paper No. 95-2686.

16. Korting, P.A.O.G.; Schoyer, H.F.R. & Timnat, Y.M. Advanced hybrid rocket motor experiments. *Acta Astronaut,* 1987, **15** (2), 91-104

17. Strand, L.D.; Ray, R.L. & Cohen, N.S. Hybrid rocket combustion study. Paper presented at the AIAA/SAE/ASME/ASEE 29th Joint Propulsion Conference, 28-30 June 1993, Monterey, CA, 1993. AIAA Paper No. 93-2412.

18. Adams, D.M. Igniter performance in solid-propellant rocket motors. *J. Spacecraft,* 1967, **4** (8), 1024-29.

19. Gordan, S. & McBride, B.J. Computer program for calculation of complex chemical equilibrium compositions, rocket performance, incident and reflected shocks, and Chapman-Jouguet detonations. NASA, Washington, D.C., 1971. NASA-SP-273.