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THE DAEDUT/PMLTNO HYBRID ROCKET MOTOR FACILITIES

by

H.F.R. Schöyer
P.A.O.G. Korting

DELFTRIJSWIJK - THE NETHERLANDS

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Summary

This report describes the hybrid rocket motor and test facilities, which are installed at the test location of the Prins Maurits Laboratory of the Organization for Applied Scientific Research (PMLTNO) at Rijswijk (ZH).

The Department of Aerospace Engineering of Delft University of Technology (DAEDUT) and PMLTNO jointly own and operate this motor and its test facilities for basic research and educational purposes. An introduction to hybrid rocket motors in general and these specific facilities is followed by a description and discussion of the major components of the system. Operational procedures are briefly discussed.

The third section of this report discusses some aspects of hybrid combustion modelling in order to introduce the reader to some fundamental problems, encountered in this field.

Some of the more important references on hybrid rocket motor technology are listed in the references. For a much more complete list of references, the reader is referred to Schmucker's book\(^1\).
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Nomenclature

$A_p$  port area
$a$  coefficient in regression rate expression
$D_0$  initial port diameter
$G$  mass flow per unit port area
$L$  grain length
$m$  mass flow
$n$  pressure exponent in regression rate expression
$p$  pressure
$r$  regression rate
$S_b$  burning surface area
$T$  temperature

$\alpha$  mass flow exponent in regression rate expression
$\rho$  density
$(f)$  fuel, flame zone
$(o)$  main gas flow
$(ox)$  oxidizer
$(s)$  burning surface
1. Introduction

Rocket motors seem to have originated in India or China about the eleventh century. We may safely assume that these precursors of our modern rockets consisted of cartridges, filled with a gunpowder-like substance and that they were stabilized by a long rod. These rockets were used for fireworks and for military applications. During the nineteenth century, Congreve, in England developed his "life saving rocket" to launch cables to wrecked vessels. Still, the solid propellant rocket was far from being the sophisticated vehicle of nowadays. Towards the end of the nineteenth century, Tsiolkovsky in Russia, developed his ideas about liquid propellant rocket motors. In 1926 Robert Goddard in the U.S.A. successfully launched a liquid propellant rocket. World War II gave a large impetus to the development of rockets and rocket propelled missiles. The V-2 is well-known as a sophisticated missile employing liquid propellants. At the same time the solid propellant rocket matured, and at the end of World War II mankind possessed the know-how to build reliable solid- and liquid-propellant rockets.

The liquid propellant rocket usually stores the liquid oxidizer and fuel in separate tanks. These propellants then are pumped into the combustion chamber, where they react and produce high temperature combustion products. To this end complicated turbopumps and injectors are required, and as the hot combustion products are in direct contact with the combustion-chamber and nozzle walls, a cooling system is mandatory. All this makes the liquid propellant rocket motor quite a complex piece of machinery. On the other hand, if cryogenic propellants are used, the performance is very high. Thrust modulation and thrust vector control are relatively simple for liquid rocket motors, thus allowing for very accurate trajectories.

The solid rocket motor of nowadays employs propellants that insulate the walls from the hot combustion-products. The propellants usually are a mixture of an anorganic oxidizer and a plastic or a gel of
cellulosenitrate and glyceroltrinitrate. Once ignited there is hardly any way to stop the motor from burning while the possibilities of thrust magnitude control are very limited. Thrust vector control is very well possible. The inherent simplicity of the solid rocket motor makes it relatively cheap, while its storability is especially very attractive for military purposes.

Liquid and solid propellant rockets are used for many modern applications, ranging from small anti-tank missiles to large ICBM's, ranging from small sounding rockets to huge satellite launchers and launch vehicles for interplanetary spacecraft. The human requirements for rockets can easily be met by either solid or liquid rockets or by a combination of both. This certainly explains that a third type, the hybrid rocket motor has received little attention in the past and has not been developed to the same extent as its liquid or solid counterparts.

The hybrid rocket motor employs propellants which are in two different states of aggregation. The usual concepts are to use a solid fuel (usually a polymer) and a liquid or gaseous oxidizer (often O₂, H₂O₂ or N₂O₄ + HNO₃). Figure 1.1 is a diagram of a hybrid rocket motor. A pump, or a pressure feed system (1) feeds the liquid or gaseous propellant (2) (usually the oxidizer) through a control valve (3) into the combustion chamber (4). The combustion chamber contains the solid component (5) (usually the fuel). Both components react (combustion) and produce high-temperature, high-pressure combustion gases. To initiate the combustion, sometimes a separate ignition system is required. This is not shown in Fig. 1.1. The combustion products then flow towards the nozzle throat (6) where they attain sonic velocity and then expand in the diverging section of the nozzle (7), reaching velocities well in excess of 2 km/s. This change in momentum from virtually nil to more than 2 km/s produces the thrust. To improve the combustion sometimes a secondary combustion chamber (8) is added to the motor.
Fig. 1.1: Diagram of a hybrid rocket motor

1. Pump or pressure feed system
2. Liquid or gaseous propellant (usually oxidizer)
3. Control valve
4. Combustion chamber
5. Solid propellant (usually fuel)
6. Nozzle throat
7. Diverging nozzle section
8. Secondary combustion chamber
The hybrid rocket motor combines some of the advantages of the other two types of rocket motors: it is fairly simple, the fuel isolates the chamber walls from the hot gases, thrust magnitude control is well possible like stopping the motor at any moment. In addition it has excellent restart capabilities. On the other hand, it is a little more complicated than the solid propellant rocket motor while its performance cannot match the performance of the liquid propellant rocket motor, especially not if the latter one employs cryogenic propellants. In comparison to the two other types of rockets, the hybrid motor is extremely safe. Most liquid propellants are highly dangerous in direct human contacts, while solid propellants in fact are explosives requiring very careful handling. The level of "unsafety" of a hybrid rocket motor is comparable to that of industrial welding apparatuses, i.e. one deals with high pressure gases and inflammable material. Another distinct advantage, which is of particular interest for scientific research, is that several candidate fuels are transparant, allowing for visual observation of the combustion process within the rocket motor. Long burning-times can be achieved (up to several minutes), and all this makes the hybrid rocket motor a very attractive laboratory tool and suitable for education and instruction of students in rocket propulsion.

A detailed discussion of hybrid rocket motors in general is given by Schmucker\(^1\), who also extensively discusses hybrid rocket propulsion and the regression of the fuel in dependence on grain geometry and pressure.

When prof. Harry Wolff of Technion, Haifa, spent a sabbatical period at the Department of Aerospace Engineering of Delft University of Technology many of these aspects have been discussed, while also prof. Wolff's experience with a laboratory scale hybrid rocket motor at Technion, led to the conclusion that such a device would also be very attractive for the Department of Aerospace Engineering. As the university did not have facilities to operate a hybrid rocket motor, while on
the other hand, the nearby Prins Maurits Laboratory of the Organization for Applied Scientific Research, TNO, was also very interested in the possibilities of the hybrid rocket motor, it was decided to cooperate in this project, and to install a hybrid rocket motor at the test locations of the Prins Maurits Laboratory. The motor has been designed by one of the authors of this report while being temporarily employed by the Deutsche Forschungs- und Versuchsanstalt für Luft- und Raumfahrt, DFVLR. The motor is a gift from DFVLR to the Department of Aerospace Engineering of Delft University of Technology.

A thrust stand has been manufactured by the Department of Aerospace Engineering. The thrust stand and hybrid rocket motor, together with its gas supply system and control system have been installed at the test location of the Prins Maurits Laboratory TNO under a cooperative programme between the University and TNO.
2. The DAEDUT/PMLTNO Hybrid Rocket Motor Facility

2.1 Introduction

Hybrid rocket motors have some distinct advantages as compared to other chemical rocket motors. As a result they are ideally suited for educational purposes and research in the area of combustion and rocket propulsion. These advantages may be summarized:

i. The hybrid rocket motor is the safest type of chemical rocket motors, due to the use of industrial fluids as an oxidizer and commercial plastics as a fuel. Its ability to operate at low chamber pressures also contributes to its inherent safety.

ii. Although the motor is fairly simple, thrust variation is easily possible.

iii. The motor and the propellants are relatively cheap and do not require extensive safety precautions, though one should be aware that one deals with inflammable materials and gases at elevated pressures.

iv. Several candidate fuels, such as plexiglas, polystyrene and polycarbonate are clearly transparent, allowing for visual observations of the combustion and flow within the motor.

The awareness of these advantages has contributed to the decision of the Department of Aerospace Engineering of Delft University of Technology (DAEDUT) and the Prins Maurits Laboratory of the Organization for Applied Scientific Research, TNO (PMLTNO) to extend their already existing cooperation into the area of hybrid rocket motor technology and related areas. To this end, a motor has been installed at the PMLTNO test location at Rijswijk (ZH).

Gaseous oxygen serves as an oxidizer while several polymers can be used as a fuel. The main emphasis, at present, lies on plexiglas, this being a clearly transparent fuel. The combination Oxygen/Plexiglas not being hypergolic, requires an ignition system. To ignite the motor, hydrogen
gas is supplied to the motor for a short time, in combination with oxygen. This mixture has a large ignitable mixture ratio and is for this reason considered as being safer than for example the combination butane/oxygen. Ignition takes place by means of a spark plug. The heat, generated by the combustion of the hydrogen/oxygen mixture, gasifies the surface of the fuel, and the fuel gases react with the remainder of the oxygen. Once this gasification has taken place, the hydrogen supply is stopped and the combustion will continue as long as oxygen is supplied to the fuel grain. The thrust of the motor can be varied by varying the oxygen supply, allowing for a simple thrust modulation mechanism. After shut-down of the oxygen supply, the combustion ceases and the motor is purged with nitrogen gas to clear it from combustion products and fuel gases. The design of the motor has been discussed extensively by Korting\(^2\). A detailed description of the complete system is given in the following sections.

2.2 Description of the Hybrid Rocket Motor Facility

The hybrid rocket motor facility consists of the following components:
i. a hybrid rocket motor
ii. a thrust stand
iii. a gas supply system
iv. a control system
v. a measuring and registration system.
These components are discussed below.

2.2.1 The Hybrid Rocket Motor

Figure 2.1 is an exploded view of the hybrid rocket motor, showing all its components. The motor consists of the following parts (the numbers behind the components refer to Fig. 2.1):
- an injection chamber (1) with needle values for hydrogen (2) and oxygen (3) supply
- a spark plug (4) to ignite the motor
Fig. 2.1 Exploded View of the Hybrid Rocket Motor

1. Injection Chamber
2. Needle Valve for Hydrogen Supply
3. Needle Valve for Oxygen Supply
4. Spark Plug
5. Connection for a Pressure Transducer
6. Fuel Grain (Shown here is a Perspex Fuel Grain)
7. Secondary Combustion Chamber
8. Nozzle
9. Connection for a Pressure Transducer
10. Rods and Bolts for Mounting the Fuel Grain.
- a connection for a pressure transducer (5) for the measurement of the injection pressure
- a fuel grain (6). Fig. 2.1 shows a plexiglas fuel grain
- a secondary combustion chamber (7)
- a nozzle (8)
- a connection for a pressure transducer (9) for the measurement of the pressure in the secondary combustion chamber
- three rods and bolts (10) to hold the fuel grain between the injection chamber and the secondary combustion chamber.

The needle valves, which are mounted on the injection chamber allow for a manuel adjustment of the mass flow of the hydrogen and oxygen on entering the combustion chamber. After that the hydrogen and oxygen supply has been started, the spark plug serves to ignite the gas mixture. By a pressure transducer on the injection chamber, the injection pressure can be measured. The fuel grain is cylin­drical with a cylin­drical conduit. During operation of the motor the bore of the conduit increases. For the present motor, the fuel grain also serves as a mechanical wall of the combustion chamber. This implies that not all fuel can be burnt. In the case of a burn-through the pressure will drop and the combustion will continue at a lower pressure level. Upon noticing such a burn-through the operator will shut-off the oxygen supply. In Fig. 2.1 a plexiglas (polymethylmethacrylate) grain is shown, but many other polymers are feasible. The secondary combustion chamber allows for the complete combustion of unburnt, or partially burnt fuel gases with the remainder of the oxidizer. It is fitted with an interchangeable nozzle as to allow for various test conditions. Both, the secondary combustion chamber and the nozzle have been manufactured from copper as this is an excellent heat sink. With the relatively short burning times of this motor, a cooling system was not considered necessary. In order to keep the inner walls of the secondary combustion chamber and the nozzle well below their melting temperature, a material with a large heat capacity and large heat conductivity is necessary. Such requirements are easily met by copper.

Table 2.1 summarizes some characteristics of the hybrid rocket motor.
Table 2.1 Some Characteristics of the DAEDUT/PMLTNO Hybrid Rocket Motor

<table>
<thead>
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<th>Characteristic</th>
<th>Specification</th>
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<tr>
<td>Oxidizer</td>
<td>Gaseous oxygen</td>
</tr>
<tr>
<td>Mass flow</td>
<td>Continuously adjustable up to 200 g/s</td>
</tr>
<tr>
<td>Injection pressure</td>
<td>Continuously adjustable from 0,4 MPa up to 10 MPa</td>
</tr>
<tr>
<td>Fuel</td>
<td>Any suitable polymer or other combustible solid</td>
</tr>
<tr>
<td>Dimensions of the fuel grain</td>
<td>Maximum diameter 0,09 m</td>
</tr>
<tr>
<td>Ignition</td>
<td>Maximum length 0,35 m</td>
</tr>
<tr>
<td>Spark plug in combination with hydrogen and oxygen gas</td>
<td></td>
</tr>
<tr>
<td>Ignition duration</td>
<td>0,1 s to 0,2 s</td>
</tr>
<tr>
<td>Combustion/Propulsion</td>
<td>Maximum pressure 5 MPa</td>
</tr>
<tr>
<td>Maximum attainable temperature (O₂/PMMA)</td>
<td>3500 K</td>
</tr>
<tr>
<td>Maximum thrust (O₂/PMMA)</td>
<td>400 N</td>
</tr>
<tr>
<td>Maximum burning time (at maximum oxygen mass flow, chamber pressure 2 MPa and PMMA fuel)</td>
<td>30 s</td>
</tr>
<tr>
<td>Combustion products (O₂/PMMA)</td>
<td>H₂O, CO₂, CO, H₂, O₂, H, OH, O, CH₂, CH₄.</td>
</tr>
</tbody>
</table>

It is seen from this table that the motor allows for a very large range of operational conditions. This feature is of special interest as it allows for experimental research in a large parameter range.

2.2.2 The Thrust Stand

Figure 2.2 is an exploded view of the thrust stand. The thrust stand is a horizontal installation. This implies that the reaction forces, due to the varying weight of the motor, and the thrust are orthogonal and hence do not affect each other. As there is only one force link, forces
Fig. 2.2 Components of the Thrust Stand

1. Frame, for Mounting the Thrust Stand on a Foundation
2. Upper Beam on which the Motor is Mounted
3. Force Link
4. Elastic Hinges
5. Adjustable Bolt, to Adjust Force Link Prestress
due to thrust misalignment or thrust vector control cannot be measured. A detailed description of the thrust stand is given by Korting (2).

To measure the thrust, the hybrid rocket motor is mounted on the upper beam (2) of the thrust stand, which on its turn is connected to the frame (1) by means of two elastic hinges (4) that can only transmit forces in vertical, but not in horizontal direction. The horizontal force (the thrust) is transmitted to the frame (1) via a piezo-electric force link (3). By means of an adjustable bolt (5) the force link can be pre-stressed. The thrust stand allows for fuel grains of different lengths, up to a maximum length of 0,35 m. Figure 2.3 shows in detail how the force link connects the upperpart of the thrust stand to the frame. Figure 2.4 shows the thrust stand with the hybrid rocket motor. The installation is capable of measuring thrusts up to 2,5 kN.

2.2.3 The Gas Supply System

The gas supply system has to provide the hybrid motor with oxygen gas during motor operation. In addition a small supply of hydrogen gas is necessary for ignition, while after the completion of a test-run nitrogen gas has to be supplied to purge the motor from oxygen gas and the remainder of combustion gases and fuel vapours. The gas supply system therefore consists of

- an oxygen supply system
- a hydrogen supply system
- a nitrogen supply system.

Figure 2.5 is a diagram of the gas supply system. For the hybrid rocket motor, normal industrial gases are used which are supplied in industrial gas bottles. The mass flow of gaseous oxygen and gaseous hydrogen that may be required during a test-run, however, is much larger than can be provided by the bore of normal industrial gas bottles. To this end an array of buffer vessels with an increased bore has been installed. The array of the hydrogen buffer consists of three bottles, with a capacity of 4 l each and the oxygen buffer consists of two oxygen bottles with a capacity of 50 l each and one oxygen bottle with a capacity of 10 l.
Fig. 2.3 Detail of the Force Link, mounted on the Thrust Stand.
Fig. 2.4 The Hybrid Rocket Motor Mounted on the Thrust Stand.
Fig. 2.5 Diagram of the gas supply system and hybrid rocket motor.
Before a test-run these bottles are filled from the supply to a required pressure level. After that the bottles have been filled the valves to the supply bottles are closed. The hydrogen and oxygen buffer are indicated diagrammatically in Fig. 2.5 by GH₂ and GO₂, respectively. The required nitrogen gas flow being much smaller, allows to obtain the nitrogen directly from industrial gas bottles. The nitrogen storage is indicated diagrammatically in Fig. 2.5 by GN₂.

Between the buffer bottles and the feedlines to the hybrid motor adjustable pressure reducers have been installed. These allow for an adjustment of the secondary pressure in the gas feed lines. The pressure reduction valve in the oxygen line can be adjusted continuously from the control room, as to allow for a varying oxygen supply during a test run. This feature allows for thrust variation during motor operation. The secondary O₂ and H₂ pressure, i.e. the pressure downstream of the pressure reduction values is measured continuously, and displayed in the control room. Electromagnetic valves in the lines can be opened or closed from the control room. This allows (by closing the H₂ and O₂ valves) to interrupt a test-run instantaneously in the case of anomalies and (by opening the N₂ valve) to purge the system with nitrogen gas. The needle valves allow for an additional adjustment of the mass flows. These needle valves are pre-set before a test-run and cannot be adjusted during a testrun as these are mounted on the rocket motor. Finally, check-valves prevent a backflow of the gases from the rocket motor into the feedlines. The gas supply system has been designed by PMLTNO and DAEDUT in cooperation with Hoek-Loos (Schiedam). The latter has also supplied and installed the gas supply system.

2.2.4 The Control System

The hybrid rocket motor system is operated from the control room. The sequence of events that is necessary to start, operate and stop the motor is diagrammatically shown in Fig.2.6. As all times involved are rather short, this sequence of events is automatically controlled from the control panel, which is shown in Fig. 2.7. The upperpart of this
$t_1$, $t_2$, and $t_3$ are pre-set and adjustable (accuracy better than 0.1 s)

$0 < t < t_1$ ignition phase
$t_1 < t < t_2$ operational phase
$t_2 < t < t_3$ purging with nitrogen

Fig. 2.6 Sequence of events for the hybrid rocket motor system.
Fig. 2.7 The Registration and Control Facilities for the Hybrid Rocket Motor.

The control panel is in the upperright corner. Below the control panel an instrumentation tape recorder. To the left amplifiers, UV oscillograph and ultrasonic registration equipment. Top left, TV screen from closed TV-circuit.
panel, visually displays the occurrence of all events. The duration of each event (see Fig. 2.6), ignition phase \( t_1 \), operational phase \( t_2 \) and purging with nitrogen \( t_3 - t_2 \) are adjusted by means of timers. There are two digital pressure indicators which show the secondary oxygen and hydrogen pressure. In addition, the safety valves for the hydrogen, oxygen and nitrogen supply are controlled from this panel. To warn people the vicinity of and on the test location of the occurrence of a test-run, sirens are also initiated from this control panel.

After the timers have been set to their required values, the safety valves are opened and a test-run may be started by turning the starter key and subsequently depressing the starter button. This starter button has to be kept depressed all the time during a test run. As long as the starter button is depressed, the sequence of events, as depicted in Fig. 2.6 will continue. If depressing the starter button is stopped, before the sequence of events has been completed, the testrun is interrupted as the oxygen and/or the hydrogen supply are interrupted like the nitrogen supply.

In addition, there is an emergency stop. If the emergency stop is depressed, the sequence of events is also interrupted: the oxygen and/or hydrogen supply are closed and a supply of nitrogen is initiated. The supply of nitrogen continues until the "end-of-emergency" button is depressed.

Lights on the upper part of the control panel indicate which phase in the sequence of events is taking place.

2.2.5 The Measurement and Registration System

Before, during, and after a test-run many variables have to be measured in order to obtain information about the performance of the rocket motor. At present, the system allows for pressure and force measurements during a testrun. It is foreseen that these facilities will be extended to temperature measurements and the measurement of the oxygen mass flow. In addition, many other variables are measured, before and after a test-run:
i primary and secondary oxygen pressure
ii primary and secondary hydrogen pressure
iii ambient temperature and pressure (only before a test run)
iv nozzle-throat and -exit diameter
v mass and dimensions of the fuel grain.

During motor operation the thrust, the injection chamber pressure and the pressure in the secondary combustion chamber are measured. The electrical signals from the pressure transducers and the force link are recorded on an instrumentation taperecorder and on a UV-oscillograph. The timebasis is provided by the UV-oscillograph and/or the taperecorder. It is possible also to record temperatures and oxygen mass flows after that the required instrumentation has been installed. The UV-oscillograph records allow for a quick, first look, analysis of the test-run. The recorded analog signal allows for digitizing the signal and subsequent computer processing. Also play-back at different speeds on the UV-oscillograph is possible as to give interesting features of the pressure- or thrust-history a second look. The registration equipment is shown in Fig. 2.7. The tape recorder is situated under the control panel; the UV-oscillograph is located to the left of the tape-recorder.
3. Some General Aspects of the Hybrid Combustion Process

3.1 Introduction

Hybrid combustion is characterized by the fuel and oxidizer being in different states of aggregation. In hybrid rocket motors, the fuel usually is a solid, while the oxidizer is a fluid. On entering the combustion chamber the oxidizer is either atomized or is gaseous. A second important observation is that hybrid combustion takes place in a (turbulent) boundary layer, adjacent to the solid fuel surface\(^3\). Heat transfer from the flame and the hot gases causes gasification of the fuel surface, which on its turn leads to a continuous injection of fuel gases into the boundary layer. The concentration of fuel gases decreases with increasing distances from the fuel surface, while the opposite is true for the concentration of oxidizer. The combustion takes place in a narrow region within the boundary layer, being indicated as the flame zone or the reaction region. The location of the flame zone within the boundary layer depends on the respective concentrations of the oxidizer and fuel gases. The flame zone is characterized by a discontinuity in the temperature profile, which is depicted in Fig. 3.1. This figure is based on an experimental simulation of hybrid combustion in a small-scale wind tunnel\(^4\). In many hybrid rocket motors, the oxidizer flow will already possess a large degree of turbulence on entering the fuel grain. In such a case, the flame zone may be expected to broaden.

The fuel grain usually has a cylindrical conduit. This makes it plausible to describe the hybrid combustion process by the theory of turbulent pipe flows with heat transfer, chemical reactions and mass addition. Experiments with the system PMMA/O\(_2\) support this. The most important observations for this combination are:

1. Schlieren and shadow photographs indicate the presence of a turbulent boundary layer over the full of the fuel conduit\(^4,5\).
Fig. 3.1 The hybrid combustion process (ref. 4).
The flame height is at about 10 per cent to 20 per cent of the momentum boundary layer thickness, depending on operational conditions(6).

The flame temperature is about 2600 K. The theoretical flame temperature has a maximum of about 3600 K at a stoichiometric mixture ratio, $\phi = 1.92$. The observed mixture ratio in the flame zone agrees to $\phi \approx 1.5$. This fuel rich mixture ratio may explain the lower flame temperature(7).

The PMMA surface temperature is about 600 K(7).

3.2 Fuel Regression Rate

A very important parameter for the thrust of a rocket motor is the mass flow. For a hybrid rocket motor the mass flow consists of the oxidizer and fuel mass flow. The fuel mass flow, $m_f$, is determined by the area of the burning surface, $S_b$, the density of the fuel, $\rho_f$, and the fuel regression rate, $r$. This regression rate is the distance that the fuel surface regresses per unit time. The fuel mass flow may be expressed as:

$$m_f = \rho_f \cdot S_b \cdot r$$  \hspace{1cm} (3.1)

In addition, the ratio of oxidizer and fuel mass flow determines the composition and temperature of the combustion products, and hence again affects the performance. It is evident that a good knowledge of the regression rate in dependence on other variables is mandatory for the design, performance prediction, and the understanding of hybrid rocket motors.

Based upon experimental observations, it has been established that the fuel regression rate is at least governed by the following parameters: the total mass flow per unit port area, and the chamber pressure.
In addition, the dimensions of the grain and the temperature of the fuel and the oxidizer affect the regression rate. The effects of accelerations normal to the burning surface are minute.

The mass flow per unit port area, $G$.

The mass flow per unit port area is defined as

$$ G = \frac{(m_{ox} + m_f)}{A_p} $$

(3.2a)

In this expression, $m_{ox}$ and $m_f$ represent the oxidizer and fuel mass flow, while $A_p$ stands for the cross-sectional area of the conduit. It turns out that the regression rate, $r$, is primarily governed by the mass flow, $G$. From Fig. 3.2 it is evident that $r$ increases monotonically with $G$.

Fig. 3.2 Regression rate versus the total mass flux, $G$, and oxidizer mass flow, $G_{ox}$, for polyethylene liquid-oxygen (ref. 8).
However, as it is very difficult to determine $G$ accurately during actual experiments, $G$ is often replaced by $G_{ox}$, i.e.

$$G_{ox} = \frac{m_{ox}}{A_p}$$

(3.2b)

which can be determined much more accurately. From Fig. 3.2 it is evident that this is also a very useful relationship.

The chamber pressure

The effect of the chamber pressure on the regression rate depends strongly on the fuel. At low pressures, i.e. below 1.5 MPa, the regression rates of most pure plastic fuels, are hardly affected by the pressure. On the other hand, if metal powder is mixed through the fuel, there is a strong dependency on the pressure, especially at low values for $G_{ox}$. At higher $G_{ox}$ values there is hardly a noticeable effect. This is illustrated in Fig. 3.3.

The effect of grain dimensions

In the previous sections, an overall picture has been drawn. It is evident, that locally, the mass flow, $G$, or $G_{ox}$, varies along the grain. It is not surprising therefore, that the dimensions of the fuel grain affect the hybrid combustion process, and hence the regression rate. Woolridge and Muzzy(9) have measured the burning rates for some PMMA grains with $O_2$ as oxidizer. They kept the length to initial part diameter, $L/D_0 = 10$, and found that regression rates increase with decreasing dimensions. This is depicted in Fig. 3.4. It is recalled that, while $L/D_0$ remains constant, for the same value of the oxidizer mass flow, $G_{ox}$, the total oxidizer mass flow, $m_{ox}$, has to vary with the square of the grain port diameter. It is seen from this figure that decreasing the dimensions by a factor four increases the regression rate with a factor 1.5.
Fig. 3.3 Regression rate versus oxidizer mass flow for some combustion pressures (ref. 9).

Initial port diameter, $D = 0.1$ m, grain length, $L = 1$ m.

Fuel: 80% PMMA, 20% Aluminium.

Oxidizer: Oxygen.
Fig. 3.4 The regression rate versus oxidizer mass flow for some different motor sizes, and $L/D_o = 10$ (ref. 9).
The effect of the initial temperature

The possible variations for the initial temperature of the oxidizer and fuel, are limited in comparison with the flame temperature of 2500 - 3000 K. This may explain why only limited effects of the initial temperature on the regression rate have been observed\(^{(8)}\).

The effect of accelerations

Accelerations, normal to the burning surface may affect the boundary layer structure. Groothoff\(^{(10)}\) has investigated the changes in burning rate by rapidly rotating the fuel grain during motor operation. These effects, however, were minute.

It may be concluded from the above that the main governing parameter for the regression rate is the mass flow per unit port area, \(G\), or the related \(G_{\text{ox}}\). If metal powders have been added to a plastic fuel pressure effects will be noticeable at low mass flows per unit port area. In addition the grain dimensions affect the regression rate. Other effects usually remain of secondary importance. Based upon these observations, an empirical, overall relationship for the regression rate has been formulated:

\[ r = a \cdot p^n \cdot G_{\text{ox}}^\alpha \quad (3.3) \]

The coefficient \(a\) depends upon the grain dimensions, and the particular fuel and oxidizer. The exponent \(n\) will only be noticeably different from zero for very specific fuels; hybrid combinations usually exhibit a very weak pressure dependency, i.e. \(n \approx 0\); in that case, Eq. (3.3) may be written as:

\[ r = a \cdot G_{\text{ox}}^\alpha \quad (3.3a) \]

The exponent \(\alpha\) is mainly governed by the particular combination of fuel
Fig. 3.5 Experimentally determined regression rates (ref. 1).

1. PMMA/O₂ at \( p_c = 0.1 \) MPa
2. PMMA/O₂ at \( p_c = 0.1 \) MPa
3. PMMA/O₂
4. polyethylene/H₂O₂
5. rubber/N₂O₄
6. rubber/N₂O₄ + metal
7. polyethylene/O₂ (liquid)
8. p-toluidin + p-Aminophenol/HNO₃
9. p-toluidin + PVC/HNO₃
10. tagaform/HNO₃
11. lithium aluminium hydride/H₂O₂
and oxidizer. This is illustrated by Fig. 3.5, which has been compiled by Schmucker\(^{(1)}\). Comparing p-toluidin with p-Aminophenol and p-toluidin + PVC and HNO\(_3\) for example shows the difference in the exponent \(\alpha\).

### 3.3 Combustion Modelling

As hybrid rocket motors have not found practical applications, until now, it is not surprising that research efforts have remained limited. As a result, also combustion modelling studies remained limited in number. The primary purpose of such studies is to adequately understand and describe hybrid combustion processes, with the ultimate goal of making accurate performance predictions. A review of some combustion models was published by Green\(^{(11)}\), while at about the same time Marxman and his co-workers\(^{(7)}\) published a combustion model based on turbulent boundary layer combustion. This work was continued and extended during the sixties. It is assumed that the hybrid combustion process is primarily governed by mixing and subsequent reactions in a turbulent boundary layer, adjacent to the fuel surface. The chemical reaction rate is assumed to be so fast, that as soon as oxidizer and fuel gases come into contact with each other, combustion takes place.

The time necessary for transport of oxidizer and fuel gases, which time depends upon the turbulence level and diffusion, therefore determines the speed of the combustion reaction. For this reason most early models aimed at seeking relations between the regression rate and typical parameters of the turbulent boundary layer, in combination with the thermo-chemical characteristics of the reactions. Marxman and co-workers\(^{(7)}\) assumed a thin flame- or reaction-zone within the turbulent boundary layer. The maximum temperature is reached in this flame-zone. This model is supported by experimental evidence\(^{(4)}\). Rate controlling mechanisms are heat transfer towards the fuel surface and diffusion of the gaseous species within the turbulent boundary layer. Both, convective and radiative heat transfer are considered; conductive heat transfer is neglected. Whether radiative or convective heat transfer dominates depends upon the specific fuel/oxidizer combination. The amount of gas
that stems from the fuel surface (blowing) strongly affects the convective heat transfer. In case of severe blowing, the convective heat transfer is strongly reduced.

To model the convective heat transfer, a modified expression for turbulent pipeflow may be applied, but it is evident that such a relation is but a course and poor description of the actual processes. Part of the heat that is transferred to the solid fuel serves to heat this fuel; the other part is needed for gasification. The more heat is needed to gasify the fuel, the smaller the blowing into the boundary layer. This in its turn allows for more convective heat transfer, more or less balancing the larger heat requirements. Marxman's early model appears to be able to explain in general terms the combustion behavior in hybrid rocket motors. For large mass flows (G) or low pressures the model failed. It was believed that this failure was caused by the elementary treatment of heat transfer in a turbulent, blown boundary layer. The effects of pressure on the combustion are not predicted correctly by the early model. Some effects of pressure on the regression rate may be explained by assuming finite reaction rates, or reaction rates which are of the same order as the transport rates. To this end, the model has been modified\textsuperscript{(12)} to accommodate these effects. Figure 3.6 shows the regression rate, \( r \), versus the mass flow, \( G \), and pressure. It is also indicated in the figure where effects dominate: radiative heat transfer for low values of \( G \), convective heat transfer for intermediate values of \( G \), and chemical kinetics for larger values of \( G \).

### Comparison with experiments

For the combination PMMA/O\textsubscript{2}, Marxman's combustion model shows reasonable agreement with experiments. Still, at low pressures, discrepancies between model and experiment are observed. Some other combinations, like polyethylene and H\textsubscript{2}O\textsubscript{2} also follow the predicted behavior. Doubts have arisen whether the model is also applicable to metal-enriched fuels. Only lithium-enriched fuels show reasonable agreement between theory and experiment.
Fig. 3.6 Regression rate behavior according to combustion modelling\(^{(?), (12)}\).
It is evident from the foregoing that the hybrid combustion process is not understood in detail. Similar observations can be made, however, for many more combustion processes. The present DAEDUT/PMLTNO hybrid rocket motor enables these institutions to initiate experimental and theoretical research to obtain a better understanding and insight in the hybrid combustion process. It is believed that this may also contribute to the understanding of more general combustion phenomena, as many of these phenomena bear a strong relation to hybrid combustion.
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