TU Delft

SOME TYPICAL HYBRID PROPELLANT ROCKET MOTORS

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Preface

Version 1 of this document has earlier been published as Technical Note 10084 (dated June 1993) within the framework of a Lecture Series on Chemical Rocket Propulsion (Lecture Series lr13A) at TU-Delft, Faculty of Aerospace Engineering. This version differs from version 1 in that some corrections have been made to the text. Also, some additional hybrid rocket motors have been included to form a more complete picture of the current status of this type of rocket motors.

I would like to give credit to the many suggestions, which I received from readers for improving and extending version 1 of this document. Also I would like to give credit to ir. A.G.M. Marée for carefully proof-reading this document. For the future, I would like to encourage all readers to provide the author with 'missing' information and/or suggestions for further improvement of this document.

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26 pages in total.

List of acronyms

AMROC	AMerican ROcket Company of Camarillo, California
DLR	Deutsche forschungs- und versuchsanstalt für luft- und Raumfahrt
EPDM	Ethylene Propylene Diene Polymer (type of rubber)
ESA	European Space Agency
GOX	Gaseous OXygen
HRM	Hybrid propellant Rocket Motor
HTP	High Test hydrogen Peroxide
HTPB	Hydroxy-Terminated PolyButadiene
LI	Liquid Injection
LOX	Liquid OXygen
LR	Faculty of Aerospace Engineering, Delft University of Technology
MLV	Medium Launch Vehicle
N_2O	Nitrous oxide
NTO	Nitrogen TetrOxide
O/F	Oxidiser-to-Fuel mass ratio
PML	Prins Maurits Laboratory
PB	PolyButadiene
PE	PolyEthene
STS	Space shuttle Transportation System
SRM	Solid propellant Rocket Motor
TEAl	Triethyl-Aluminium
TVC	Thrust Vector Control
UoS	University of Surrey
USAFA	United States Air Force Academy

Introduction

In this document, some typical Hybrid propellant Rocket Motors (shortly referred to as hybrid rocket motors; HRM's) will be described in order to form an in-house database, which allows for comparative analysis and applications in practical HRM engineering.

To start with, however, a general description is provided of HRM's including a discussion of the various advantages and disadvantages of HRM's. This is to allow the reader to form a basic understanding of the important motor components and how they influence the design and the performances of HRM's.

HRM's use a combination of liquid and solid propellants. Usually, a liquid oxidiser is used in combination with a solid fuel. The liquid oxidiser is mostly stored in a separate tank, whereas the solid fuel is stored in the actual combustor in the form of a shaped solid fuel grain with one or more ports. The shape of the grain is chosen such that the burning area of the grain matches with the oxidiser flow to obtain the proper mixture ratio and to provide insulation to a large part of the combustor case as well. During operation, the solid fuel is gasified, causing fuel regression, after which a combustible mixture is formed with the oxidiser present. The fuel regression rate is mainly determined by the oxidiser mass flux[1] through the engine. The liquid oxidiser is fed to the combustor by a pressure-fed or turbopump-fed feed system. Start-up or ignition of the motor is accomplished hypergolically or by a separate ignition system.

HRM's have been studied in the past, using many combinations of liquid oxidisers and solid fuels [1]. Interest in these studies has mainly been focussed on high specific impulse. Unfortunately, study results showed that most promising propellants (giving specific impulse values of over 500s) are highly toxic and therefore not recommended for use. Allthough these studies also showed that reasonable specific impulse values can be achieved, when using very common non-toxic, non-explosive, and non-corrosive fuels and oxidisers, interest in hybrid propulsion subsided.

With the catastrophic failures of solid rocket booster motors in 1985 and 1986 during launches of the Titan 4 rocket launcher and the STS Challenger, interest in HRM's has been revived, promising higher safety, more reliable and less costly propulsion [1,2].

In terms of specific impulse, HRM's offer specific impulse values, which lie in between the values for solid and liquid propellant rocket motors [2]. Compared to solid propellant rocket motors (SRM's)s, the performance of HRM's is such that high specific impulse can be achieved using non-toxic and non-corrosive fuels. This ensures that no aluminum oxide or other particulate is released into the orbital environment. Furthermore, it precludes the multiphase flow and thermomechanical abrasion concerns of SRM's.

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[1] Mass flux is defined as mass flow per unit of combustor port area (taken perpendicular to the main flow direction). Actually, the fuel regression rate is determined by the total (oxidiser and fuel) mass flux. However, in most practical HRM's, the fuel mass flow is almost negligible compared to the oxidiser mass flow.

Since for HRM's, unlike for SRM's, the fuel and oxidiser are not intimately mixed, like for solid propellants, there is no potential for detonation of the propellants. Also, when leakage occurs, hybrid propellants are not likely to explode. Furthermore, hybrid propellants can be easily manufactured in a light-industrial production facility with simple safety procedures unlike the remote production facilities and extensive safeguards required for SRM propellant production. Furthermore, when using non-toxic, non-explosive, and non-corrosive (hybrid) propellants, launch processing- and payload personnel can work safely near the fully loaded stage.

For HRM's the rate of combustion, i.e. the regression rate, is limited by the flow rate of the oxidiser over the fuel surface. Thus by throttling the oxidiser flow the thrust is modulated. Terminating the oxidiser flow terminates thrust.

The HRM is a robust propulsion system because, it is a simple design that can tolerate minor production flaws in the fuel grain and off-design feed system performance. The hybrid combustion process is insensitive to minor grain de-bonds and cracks because the hybrid combustion actually occurs in a flame zone near the surface. If a crack exists, the oxidiser and fuel mixture ratio will be such that it is fuel rich thus the rate of combustion is very slow reducing the potential of a burn through to the motor case. This inherent combustion process of hybrids permits hybrids to be built rapidly without excessive manufacturing tolerances and with reduced inspections. This translates directly into faster propulsion system production, vehicle integration and launch processing.

The HRM can use an extremely simple, single fluid injector unlike the dual fluid injectors used in liquid rocket engines. Since the hybrid injector only injects one fluid, a simple showerhead type injector can be used; whereas, liquid rocket injectors are complex devices that require hydraulic matching between the two fluid streams thereby complicating the injector development, particularly for throttling applications. In addition, since only one fluid must be injected into the combustion chamber, hybrid systems only require one feed system compared with two for liquid propellant engines. Only one-half of the feed system hardware is required for a hybrid system which translates into higher reliability (i.e. fewer parts) compared with a liquid system.

It are the above mentioned specific design features, which make HRM's very attractive for space applications.

Hereafter, a number of HRM's will now be described in more detail including (depending on whether information is available or not):

- Lay-out
- Motor components like motor case case, fluid injector, feed system, igniter, nozzle, and fuel grain.
- Geometrical data: overall length, maximum diameter and nozzle geometric expansion ratio.

- Propulsive performances:
  - Average thrust over the burn time. In general, it is found that the thrust increases with increasing altitude. The difference between sea level (SL) thrust and vacuum thrust typically is about 10% (depending on the expansion ratio).
  - Specific impulse as a measure for propellant consumption; the higher the specific impulse, the lower the propellant consumption. Like the thrust, the specific impulse increases with increasing altitude.
  - Total impulse, defined as the integration of the thrust over the burn time, as a measure for the change in momentum, which can be accomplished.
  - Burn time or burn duration. The burn time is defined as the action time minus the ignition time and the time required for thrust tail-off.
  - Propellant fraction, i.e. propellant-to-total motor[2] mass ratio. This parameter allows for judging the quality of a HRM. This ratio can amongst others be used in mission calculations using e.g. Tsiolkovsky's relation.
- Oxidiser and Fuel properties as well as their mixture ratio based on mass (O/F).
- Materials.
- Mass characteristics.
- Ballistic performances like the (average) chamber pressure and fuel regression rate.
- Production, integration and launch processing.

For a(n) (further) clarification/explanation of the above mentioned parameters and items, the reader is referred to the general literature on (hybrid propellant) rocket propulsion.

[2] In general, rocket motor mass is fairly equal to stage mass. It differs only in that it does not include a nose section, separation motors, avionics, and maybe some other remaining small items.

1. Early German motor [3]

A typical early HRM is shown schematically in Figure 1.1. It employs a stack of multi-perforated cylindrical disks of coal as the solid fuel, through which flows (gaseous) nitrous oxide. The rapid development of uniform burning throughout the length of the motor is promoted by lining each of the disk perforations with celluloid, with ignition accomplished by a small initiation charge of gunpowder and a booster charge of granulated coal. Thrust levels of approximately 5000 to 10000 N for durations ranging from 40 to 120 s were achieved.

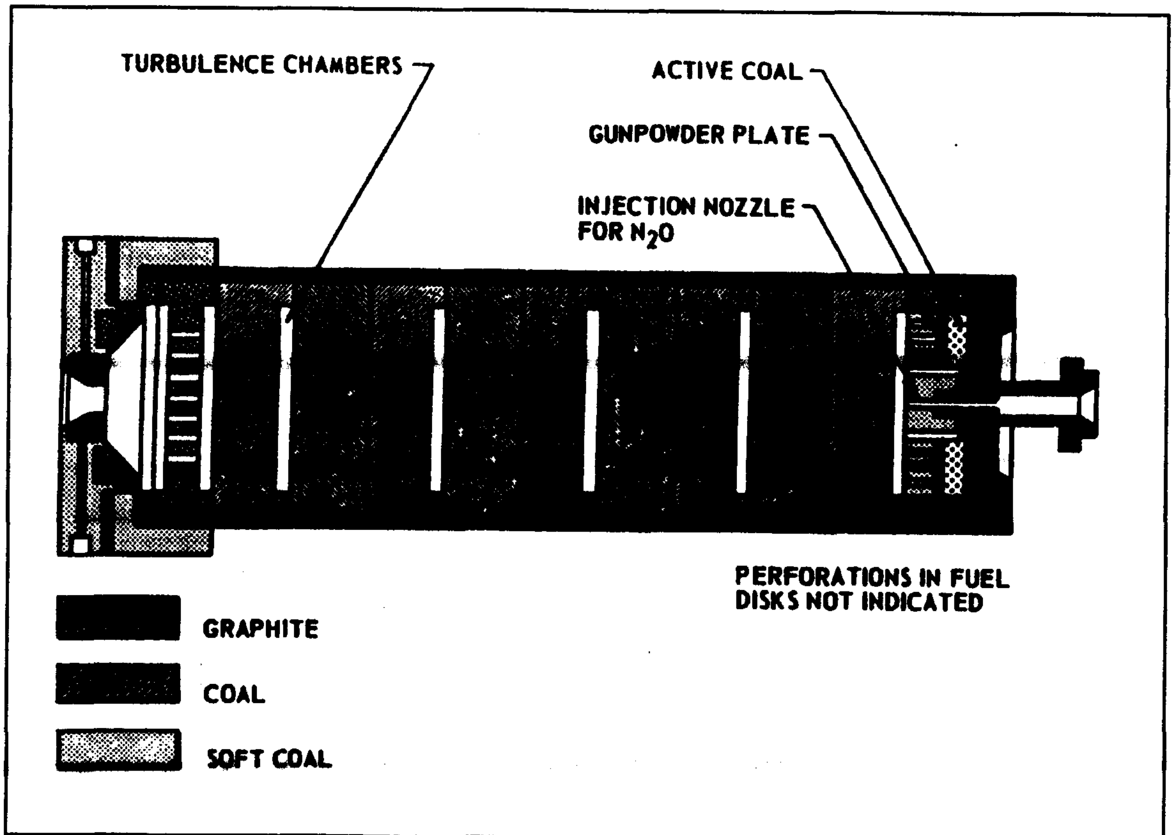


Fig. 1.1: Typical early hybrid rocket motor [3]

2. AMROC H-30 [4,5]

The AMerican ROcket Company (AMROC) of Camarillo, California, is currently in the process of developing a pressure-fed HRM, referred to as as the H-30, for use as upper stage motor. The usable propellant mass of this motor is 497 kg and the total stage mass is 641 kg[3]. The motor delivers an average vacuum thrust of 14.1 kN at a vacuum specific impulse of 287 s. The average burn time of the motor is about 100 seconds with a vacuum total impulse delivered during burning of 1.4 MNs. Some further motor characteristics are given in Table 2.1.

The motor uses a graphite/epoxy composite motor case and high expansion carbon/phenolic nozzle to obtain high propulsive performances. The nozzle is equipped with an liquid injection thrust vector control (LI-TVC) system using the oxidiser drawn from the main oxidiser tank. This system is capable of vectoring the thrust axis a maximum of 2° using four redundant, digitally controlled valves.

Vacuum Total Impulse, (kNs)	1.4×10^3
Average Vacuum Thrust, (kN)	14.1
Vacuum Specific Impulse, (s)	287.3
Burn Time, (s)	100
Chamber Pressure, (MPa)	3.10
Motor Diameter, (m)	0.61
Motor Length, (m)	0.71
Nozzle Expansion Ratio, (-)	75.1:1
Total Motor Mass, (kg)	NA[1]
Total Usable Propellant mass, (kg)	497
Liquid Injection Thrust Vector Control	2 deg. Pitch, Yaw

1) Not Available

Table 2.1: H-30 Hybrid Rocket Motor Characteristics [5]

The motor is loaded with polybutadiene (PB) fuel and utilizes storable nitrous oxide as an oxidiser. This oxidiser can be pre-loaded and maintained in a flight ready state indefinitely. Both the fuel and the oxidiser are classified as non-toxic, non-explosive, and non-corrosive.

[3] This leads to a stage propellant fraction of about 0.775. Note that this should not be confussed with the motor propellant fraction, as besides the motor mass, the stage mass also includes the mass of the skirt extension(s), the destruction system, the electrical and instrumentation system and the separation system (provided that these systems are available).

Nitrous oxide (N_2O) is stored as a liquid in the oxidiser tank and is fed into the motor under its own vapor pressure, thus eliminating the need for turbopump or pressurization system hardware. This simple and reliable pressurization method is based on the thermochemical properties of the liquid and is not dependent on any external feed system for motor operation. The process is identical to the self-pressurizing process which feeds butane in cigarette lighters. Although the vapor pressure decays slightly, and hence also the chamber pressure, as liquid is expelled from the tank, it remains sufficiently high to maintain near optimum motor operation during the entire burn.

A single valve controlling the oxidiser flow is capable of throttling the motor and also allows for precise thrust termination and restart.

3. AMROC H-500 [4,6]

On September 2, 1988, AMROC succesfully completed a full duration, 70 second firing of its 315 kN (70.000 lbf) sea-level thrust H-500. This motor is intended for use on the AMROC proposed Medium Launch Vehicle (MLV). The motor specifications are given in Table 3.1, whereas Fig. 3.1 gives an illustration of the motor.

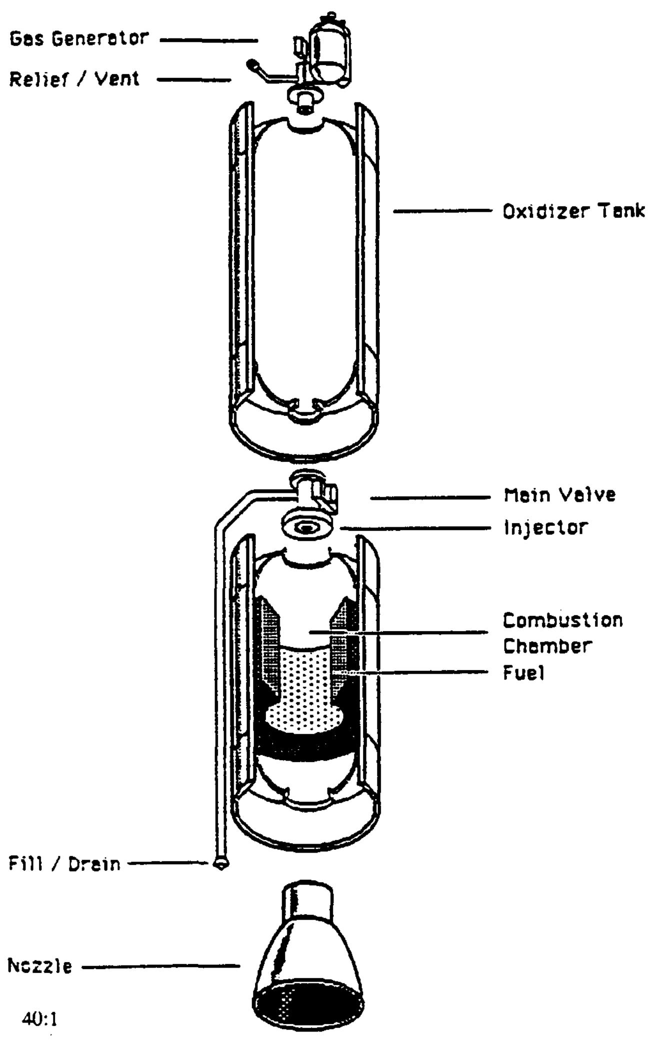


Fig. 3.1: H-500 propulsion system [6]

In the H-500 motor, liquid oxygen (LOX) is fed to a solid PB fuel grain. The fuel grain is an 11-spoke configuration[4] and is embedded in a carbon fibre/epoxy composite motor case. The LOX tank is made of stainless steel with carbon fibre overwrap.

Ignition for this motor is achieved hypergolically by injecting LOX with triethyl-aluminium (TEAl). The motor allows for throttling, TVC and restart. TVC is achieved by LI using 70% hydrogenperoxide. The LI-TVC system consists of 3 injectors flush mounted in the nozzle. The injectors can be operated either singly or in combination to redirect the nozzle's thrust vector upto 6 degrees.

The nozzle is a one-piece, tape-wrapped silica/phenolic and glass/phenolic ablative nozzle. The silica/phenolic portion of the nozzle produces a char layer when exposed to the combustion gases, which is partially consumed during the motor burn. The glass/phenolic portion makes up the outer shell of the nozzle and provides the structural support for the nozzle and the LI-TVC valves.

[4] For information on fuel regression rate, the reader is referred to the entry on the H-1500 motor.

Vacuum Total Impulse, (kNs)	28.0×10^3
Average Vacuum Thrust, (kN)	400.3
Vacuum Specific Impulse, (s)	NA[1]
Burn Time, (s)	70
Chamber Pressure, (MPa)	NA
Motor Diameter, (m)	1.29
Motor Length, (m)	Approx. 15.55
Nozzle Expansion Ratio, (-)	40:1
Approx. Total Motor Mass, (kg)	NA
Total Usable Propellant mass, (kg)	NA
Liquid Injection Thrust Vector Control	6 deg. Pitch, Yaw

1) Not Available

Table 3.1: H-500 Hybrid Rocket Motor Specifications [4]

4. AMROC H-1500 [5,7,15]

Besides the H-30 and H-300 HRM (see earlier in this document), another development of AMROC is the H-1500 HRM. This motor is based on H-500 technology and is intended for use on small launchers, like the Aquila launcher, or as a strap-on booster. Like the H-500, it uses Liquid OXygen (LOX) and PolyButadiene (PB) as the propellants. A first test firing of this motor was conducted on December 18, 1991.

The 8.79 m long H-1500, see Fig. 4.1, is capable of delivering an average vacuum thrust of 7.6 MN for a duration of 95 s, leading to a vacuum total impulse of about 72 MNs. The motor has a vacuum specific impulse of 295.2 s. The total stage mass[5] is 31.2 ton of which approximately 79.7% is propellant (24.9 ton). Some basic features of the H-1500 are outlined in Table 4.1.

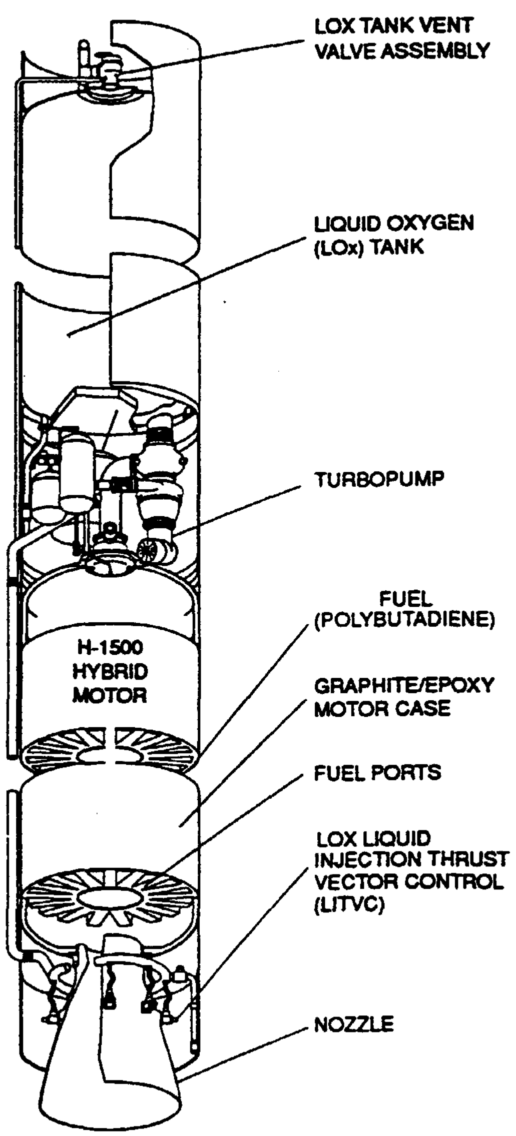


Fig. 4.1: H-1500 propulsion system [5]

The H-1500 is a turbopump-fed motor. To feed the motor, first the LOX tank is pressurized using a pressurization system, then the LOX is drawn into the turbopump, fed into the main LOX valve and injected through the showerhead injector at the head end of the motor. Ignition is achieved hypergolically by injecting LOX with TEAl.

The motor case is a filament wound graphite/epoxy composite case. The Kevlar insulation between the motor case and the PB fuel is wound on the mandrel, then overwrapped with the graphite/epoxy. The PB is cast into an internal grain configuration that consists of 15 spoke-like components made of fuel that are configured like a 'wagon wheel'. The fuel is burned in a triangular port that is located between each spoke. The time and space-averaged fuel regression rate is expressed by:

$$r = a \cdot G^{0.76} \cdot L^{m}$$

where G is the total mass flux, L is the port length, and a and m are as yet unknown constants. This equation is claimed to accurately predict the regression rate of over fourty

[5] No data is available on motor mass.

motors varying in thrust from about 500 N - 1250 kN (port hydraulic diameters of about 0.04 m to 0.35 m) with an average error of less than 4.8%. The motor has both a pre-and aft-combustion chamber incorporated in the thrust chamber. The aft combustion chamber ensures efficient combustion of the gasified fuel, whereas the pre-combustion chamber possibly ensures the proper burning of the motor (flame stabilization).

The nozzle is a one-piece, tape-wrapped silica/phenolic and glass/phenolic ablative nozzle, based on the current H-500 ablative nozzle, and is mounted at the aft end of the thrust chanmber. The structural shell resists the motor thrust loads, as well as the side forces on the nozzle induced by the LI-TVC system operation.

The injector is a centrally located, multiple element, showerhead-type injector. The showerhead injector consists of numerous individual LOX streams that form a cone shaped spray pattern. As such, it provides an even distribution of atomized liquid oxygen to the motor.

The oxidiser tank, which is used as the storage vessel for the LOX used by the hybrid motor and the LI-TVC system, is made of lightweight aluminum that can withstand the tank internal operating pressure of about 0.4 MPa (60 psi) and the external flight loads. The LOX feed system draws the LOX from the LOX tank, and feeds the LOX into the motor injector. It basically consists of a turbopump with a hydrogen peroxide gas generator drive system and a cryogenic main LOX valve. The latter allowing throttling of the motor. A tridyne pressurization system is used to feed the LOX into the hybrid rocket motor. The tridyne system is based on passing a mixture of helium, hydrogen and oxygen over a platinum catalyst bed reacting the hydrogen and oxygen releasing energy which in turn heats the bulk helium. By increasing the bulk temperature, the required helium mass for pressurization can be decreased by as much as fifty percent.

Vacuum Total Impulse, (kNs)	72×10^3
Average Vacuum Thrust, (kN)	758.4
Vacuum Specific Impulse, (s)	295.2
Burn Time, (s)	95.0
Chamber Pressure, (MPa)	3.45
Motor Diameter, (m)	1.83
Motor Length, (m)	8.79
Nozzle Expansion Ratio, (-)	17.7:1
Total Motor Mass, (kg)	NA[1]
Total Usable Propellant Mass, (kg)	24890
Liquid Injection Thrust Vector Control	Pitch, Yaw
Turbine Exhaust Roll Control	Roll

1) Not Available

Table 4.1: H-1500 Hybrid Rocket Motor Specifications [5]

5. ESA 3 kN motor [8]

In 1988, ESA published the results of a design study of a 3 kN hybrid kick motor for Earth satellites. This motor has a propellant mass fraction of about 95.9% and uses polyethene (PE) as the (solid) fuel and storable nitrogen tetroxide (NTO) as the oxidiser. The fuel is combusted in the hybrid combustor at a pressure of 1.0 MPa and an oxidiser/fuel mixture ratio of 3.6 (based on mass).

The design of the 3 kN pressure-fed hybrid motor incorporates a double-domed cylindrical kevlar motor case, which is lined with ethylene propylene diene polymer (EPDM) rubber as a thermal insulator. The parts of the motor that are directly exposed to the combustion gases are protected by a carbon-carbon wall.

The nozzle is made of carbon-carbon and is of a radiation cooled, optimum contoured design with an expansion ratio of about 215.8, a throat radius of 0.0452 m, a length of 0.984 m and an exit divergence angle of about 8.4 degrees.

The grain is of a tubular design having one central hole. The grain length is approximately 1.60 m. The liquid NTO is stored in 6 titanium alloy tanks and is pressure-fed into the motor by a pressure gas. This pressure gas is helium, which is stored in separate tanks, made of composite material with a metal liner, at an initial pressure of 30 MPa.

Injection of the NTO in the combustor is accomplished using a showerhead injector, whereas ignition is accomplished hypergolically through the injection of a small amount of hydrazine together with the NTO.

Table 5.1 gives some typical values for the above described rocket motor.

Vacuum Total Impulse, (kNs)	21×10^3
Average Vacuum Thrust, (N)	3000
Vacuum Specific Impulse, (s)	295
Burn Time, (s)	7000
Chamber Pressure, (MPa)	1.0
Motor Diameter, (m)	1.00
Motor Length, (m)	1.60
Nozzle Expansion Ratio, (-)	190.1:1
Total Motor Mass, (kg)	6994
Total Usable Propellant Mass, (kg)	6710

Table 5.1: ESA 3 kN Hybrid Rocket Motor Specifications [8]

6. DLR HY-157 [9]

Recently, some initial studies have been performed on a hybrid booster motor for Ariane 5 by DLR. This so-called HY-157 motor uses LOX as the oxidiser and PE as the solid fuel. The LOX is stored in a storage tank made of aluminium and is pump-fed to the motor by a turbopump-fed feed system. The turbine of the turbopump is driven by hot gases, which are provided for by a separate hydrazine gasgenerator. Tank pressurization is achieved through cold gas tank pressurization. The PE fuel grain is of a cylindrical design with 12 cylindrical combustion channels, and is embedded in a low alloy carbon steel case. TVC is possible through gimballing of the nozzle.

Table 6.1 gives some typical performance data of the HY-157 hybrid rocket motor.

Vacuum Total Impulse, (kNs)	NA[2]
Sea-level Maximum Thrust, (kN)	5224
Sea-level Specific Impulse[1], (s)	298.2
Burn Time, (s)	NA
Chamber Pressure[1], (MPa)	4.8
Motor Diameter, (m)	5.20
Motor Length, (m)	23.50
Nozzle Expansion Ratio, (-)	NA
Total booster Mass, (ton)	188.0
Total Usable Propellant Mass, (ton)	157.2

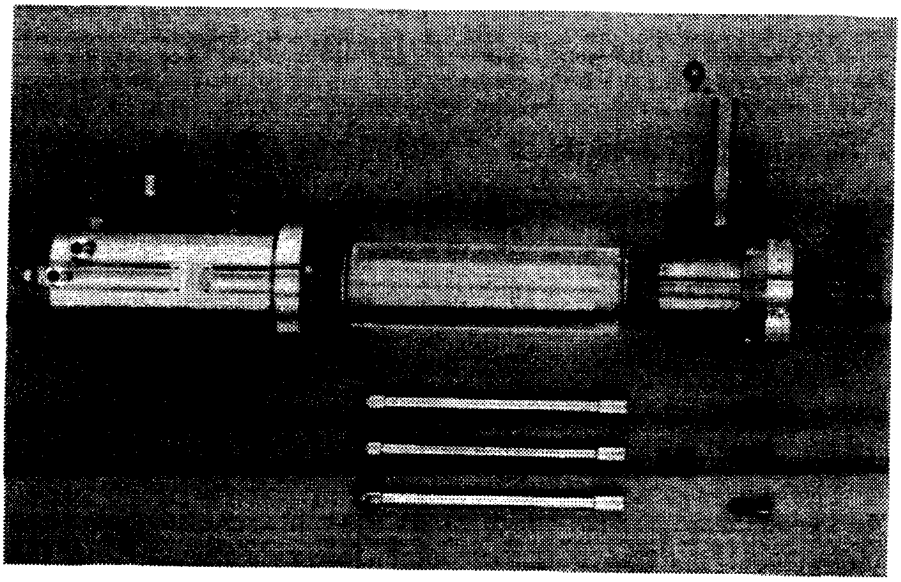
1) Average value
2) Not Available

Table 6.1: DLR HY-157 Hybrid Rocket Booster Specifications [9]

7. TU-Delft/LR - PML 0.1 kN motor [10,11]

Since the early 1980's, several studies have been performed by the Delft University of Technology, Faculty of Aerospace Engineering (LR) together with the Prins Maurits Laboratory (PML) of TNO using a highly modular HRM (burn rate) test motor.

One of the motors tested used Gaseous OXygen (GOX) as the oxidiser and high density PolyEtheen (PE; some kind of plastic) as the solid fuel. This combination is relatively inexpensive[6] and has good commercial availability. In addition, PE is easy to machine.



- an injection chamber (1) with needle valves for hydrogen (2) and oxidizer (3) supply
- a spark plug (4) to ignite the motor
- a connection for a pressure transducer (5) for the measurement of the injection pressure
- the fuel grain (6), either PMMA or PE
- an aft mixing chamber (7)
- a nozzle (8), which is interchangeable so as to allow for various test conditions
- 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 aft mixing chamber

Fig. 7.1: Exploded view of TU-Delft/LR - PML HRM thrust chamber [10]

Figure 7.1 provides an exploded view of the thrust chamber assembly. The motor is capable of providing about 0.1 kN of sea level thrust for 10-30 seconds. Maximum operating pressure of the motor is about 4.0 MPa (40 bar).

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[6] For a solid cylindrical slab of PE with an outer diameter of 90 mm, a price per meter has been quoted of about Hfl. 140,- (1983).

The GOX is stored at high pressure (about 20 MPa) in a stainless steel storage tank and is pressure-fed to the thrust chamber. A pressure regulator provides a constant line pressure. Line pressure is isolated from the pressure in the combustion chamber through the use of a (sonic) throat section. During operation, the oxygen mass flow can be adjusted by adjusting the line pressure. Injection into the combustor port takes place using a single-hole flow piece (injector), mounted in the forward part of the motor, which ensures that the oxidiser enters the fuel port in a direction parallel to the side walls of the fuel port.

The PE grain, which is mounted between the forward end of the motor and an aft combustion chamber, is of a cylindrical design with a single cylindrical combustion port in the centre. The grain length is 0.30 m with an outer diameter of 0.07 m. The fuel burns inside-out. Tests indicate an average (both in time and along the fuel grain) fuel regression rate, which depends on the total mass flux (G) and the pressure (p), according to:

$$r = 0.0635 \cdot G^{0.36} \cdot p^{0.22}$$

where the average regression rate r is in mm/s, G is in kg/(m^{2}s) and p is in MPa. The above equation is valid for 30 kg/(m^{2}s) < G < 140 kg/(m^{2}s) and 0.3 MPa < p < 2.0 MPa. Typical average regression rates are of the order of several one tenths of a millimetre. Typical burn time of the motor is in the range from 15 s - 40 s.

As PE does not react hypergolically with oxygen, the oxygen is mixed for some time (typically 0.2 s) with a small amount of hydrogen (10 g/s - 40 g/s) in the forward part of the motor, the so-called ignition chamber, thereby forming an ignitable mixture. Next, this mixture is ignited by means of a spark plug. The hot gases (mainly water and oxygen) are vented into the combustor port, thereby starting the combustion of the initially solid fuel.

When leaving the combustor port, prior to entering the nozzle, the combustion gases enter a(n) (stainless steel) aft combustion chamber. This is to allow for efficient combustion of the fuel. Hereafter, the combustion gases enter the copper conical convergent-divergent nozzle.

Table 7.1 gives some typical performance data of the above described TU-Delft/LR-PML PE-oxygen motor assuming ideal expansion to sea level pressure.

Sea-Level Total Impulse, (kNs)	1.4[1]
Sea-Level Average Thrust, (N)	87.6[1]
Sea Level Average Specific Impulse, (s)	184.6[1]
Burn Time, (s)	16.4
Average Chamber Pressure, (MPa)	0.91
Average Mixture Ratio, (-)	4.83
Grain Outer Diameter, (m)	0.07
Initial Port diameter, (m)	0.02
Grain Length, (m)	0.30
Total Motor Mass, (kg)	NA[2]
Total Usable Propellant Mass, (kg)	0.794

1) Assuming ideal expansion to sea-level pressure
2) Not Available

Table 7.1: TU-Delft/LR - PML HRM Characteristics [10]

8. USAFA Chiron motor [12,13]

Since 1989, the United States Air Force Acadamy (USAFA) has succesfully developed a HRM using Liquid OXygen (LOX) as oxidiser and Hydroxy-Terminated PolyButadiene (HTPB) as fuel. This motor is more commonly referred to as the Chiron motor (after the sounding rocket for which the motor has been developed). In 1991, a predecessor[7] of the Chiron motor, using gaseous oxygen instead of liquid oxygen, has succesfully flown on board of a small rocket. Total component production cost of the Chiron motor is below 30000 guilders (US$ 20000), mainly, because use has been made of good contacts with amateur rocket builders.

The Chiron HRM is capable of providing 4500 N of thrust for 10-15 seconds. Its length is estimated at about 3.62 m (1.07 m thrust chamber assembly) and its diameter 0.21 m. The thrust chamber incorporates a pre-combustion chamber about 0.05 m long and a 0.1 m long aft-combustion chamber.

Its design is based on a three-port, cylindrical, HTPB fuel grain stored in the motor case. The ports of this grain are pie-shaped, using plexiglass as a grain stiffener in the walls between the ports. The outer diameter of the grain is 0.15 m. The motor case is made of stainless steel with on each end a welded flange of about 1 cm. Insulation of the case walls is provided by unburned HTPB. The regression rate of the solid fuel is found to be about proportional to the square root of the oxidiser mass flux through the motor.

The LOX tank is a welded 0.003 m thick aluminium tank, which holds about 15 kg (30 pounds) of LOX, and is insulated with a water heater tank blanket. This LOX is pressure-fed to the thrust chamber, using a simple regulated blowdown system. Helium (He), stored in a commercially available SCUBA tank at an initial pressure of about 20.5 MPa (3000 psi), is used as the pressure gas. The system is activated by a valve below the He tank. Opening the valve allows the gas to pass to the regulator and then to the LOX tank. The regulator reduces the He pressure to about 6.2 MPa (900 psi). Thrust chamber pressure is about 1.8 MPa (260 psi).

Injection into the combustor port takes place through a simple shower head non-impinging injector with a three clustered orifice pattern designed to inject an equal amount of LOX down in each port. The injector is made of 302 stainless steel, has 27 orifices (9 per port), each with a diameter of about 1 cm. Line pressure is isolated from the combustor ports through the use of a cavitating venturi welded to the injector.

The nozzle is a convergent-divergent grahite bell-type nozzle coated with silicon carbide. Ideal expansion is reported to occur at an altitude of about 5000 m.

[7] This motor has been dubbed H(ybrid)-1 and has flown on a 1.2 m tall, student-developed, rocket. The motor provides about 0.25 kN of thrust and has a burn duration of about 2.3 s.

9. UoS 0.5 kN HRM [14]

Since 1994, several studies have been performed by the University of Surrey (UoS) together with the United States Air Force on a small (0.5 kN) HRM to propell small satellites. This motor uses liquid hydrogen peroxide as the oxidiser and ultra high molecular weight polyethene (PE; some kind of plastic) as the solid fuel. A cut-away drawing of the proof-of-concept hybrid motor attached to its support plate is shown in Figure 9.1.

High Test hydrogen Peroxide (HTP), containing at least 85 mass% of hydrogen peroxide (15% water) has been selected as the oxidiser and PE as the fuel. The reason for selecting this combination is, that HTP (after decomposition) reacts hypergolically with PE, whereas PE offers inexpensive commercial availability and ease of machining.

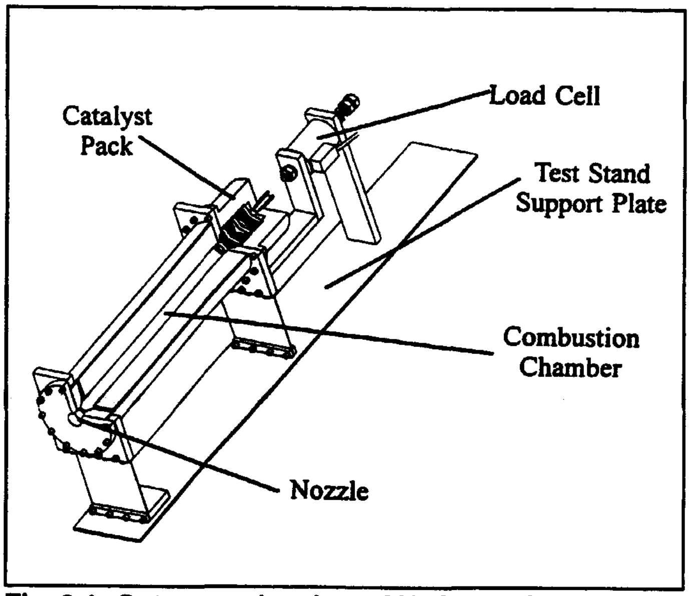

Fig. 9.1: Cut-away drawing of UoS proof-of-concept HRM

The HTP oxidiser is stored at high pressure in a storage tank and is pressure-fed (using Nitrogen gas) to the actual thrust chamber. Firstly, the HTP is injected into a catalyst bed, which consists of a stainless steel housing into which approximately 88 silver-plated nickel gauze disks are compressed. Within this bed, the HTP chemically decomposes into superheated (in excess of 850 K) steam and oxygen. Secondly, The HTP decomposition products are vented into the combustor port causing spontaneous ignition followed by combustion.

The PE fuel grain is placed in a 316 stainless steel seamless tube (the combustor case) with 0.1016 m outer diameter. Nominal operating pressure is 1.724 MPa (about 17 bar). The PE grain is of a cylindrical design with a single cylindrical combustion port in the centre. The grain length is 0.36 m. The fuel is burned inside-out. Initial tests indicated a measured average fuel regression rate of 4.47 x 10^{-4} m/s (about 0.45 mm/s) at a measured oxidiser flow rate of 0.145 kg/s.

After combustion, the combustion gases are exhausted through a graphyte/pyrolytic graphite nozzle, which is fitted into a steel housing. The nozzle expansion ratio is 3.34, giving expansion to sea level pressure at the nominal chamber pressure.

Table 9.1 gives some typical performance data of the UoS 0.5 kN HRM.

Vacuum Total Impulse, (Ns)	NA[1]
Sea-level Maximum Thrust, (N)	NA
Characteristic velocity, (m/s)	1492
Burn Time, (s)	18.5
Chamber Pressure, (MPa)	1.81
Motor Diameter, (m)	0.10
Motor Length, (m)	0.36
Nozzle Expansion Ratio	3.34
Total Motor Mass, (kg)	NA
Total Usable Propellant Mass, (kg)	NA

1) Not Available

Table 9.1: UoS 0.5 kN HRM Characteristics [14]

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