AIAA 97-2802

DEVELOPMENT WORK ON A SMALL EXPERIMENTAL HYBRID ROCKET

Matthias Grosse

D-83435 Bad Reichenhall

33rd AIAA/ASME/SAE/ASEE

Joint Propulsion Conference & Exhibit

July 6 - 9, 1997 / Seattle,WA

DEVELOPMENT WORK ON A SMALL EXPERIMENTAL HYBRID ROCKET

Matthias Grosse*

Bad Reichenhall, Germany

Abstract

This paper presents development work on a small unguided, experimental hybrid rocket with an initial mass of 65 kg (143 lbs). Hydroxyl-terminated polyisoprene and pressure-fed red fuming nitric acid are used as propellants. The HERA (**H**ybride **E**xperimental-**Ra**kete) experimental rocket project was started in order to study characteristics of the above mentioned hybrid propellants. The final goal of this investigation was to develop a rocket with a range of 20 km. More than 40 test firings were performed with different experimental testbeds. These tests were divided into five different series:

- Initial tests
- Internal ballistic tests to evaluate performance and fuel regression rates
- Thrust measurements and hardware tests
- Testing of a robust stationary testbed of the HERA hybrid motor
- Qualification of the entire propulsion system

Nomenclature

a	regression law constant
A_e	nozzle exit area
A_t	nozzle throat area
c^*	characteristic velocity
C_F	thrust coefficient
G_{ox}	oxidizer mass flux
I_s	specific impulse
L^*	characteristic motor length
l_G	grain length
Δm_{fu}	fuel mass consumption
p_c	chamber pressure
r	port radius
r_0	initial port radius
r_1	end port radius
dr/dt	fuel regression rate
t_b	burning time
α	mass flux exponent
β	geometrie exponent
ρ	fuel density
Φ	mixing ratio oxidizer/fuel
η_{c^*}	combustion efficiency
η_{CF}	nozzle efficiency
η_{Is}	thrust efficiency

*Member, AIAA, Development Engineer

Introduction

Besides the established liquid and solid propulsion systems, the hybrid rocket is very interesting for future application. The combination of solid and liquid fuel components promises advantages and improvements concerning reliability, safety, cost effectiveness and high performance combined with non-polluting characteristics.

Project HERA aimed at the devolepment of a small experimental hybrid rocket in a realisistic environment. This procedure has the advantage, that the experimental program was influenced by the development parameters of a flight vehicle. The investigation was conducted and financed between 1989-1992 by the author [1] and was supported by DASA (Bremen/Trauen). Other aspects of the project HERA were the study of specific hybrid internal ballistics, performance characteristics, fuel regression parameters and measurement accuracy.

The design parameters of HERA were a range of 20 km and a lift-off mass of less than 70 kg. Further design features for the experimental rocket were:

- Storable oxidizer
- Uncooled expansion nozzle
- Simple ignition method
- Pressure regulated oxidizer feed unit

The HERA rocket consists of a hybrid rocket motor, stabilizing fins and a parachute system. Construction materials were mainly alumium alloys and carbon composites. The motor nozzle insert is made of graphite and the nozzle structure of nickel-based super-alloy. The oxidizer feed unit consists of the components: oxidizer tank, oxidizer flow controller and regulated pressurization system.

Figure 1: Initial Version of HERA

The initial rocket design (fig. 1), based on theoretical calculations and initial tests, was continually improved in an iterative process, using the test firing data and results of 6-DOF-simulations. These simulations produced launch parameters for maximum range and the resulting impact dispersion, as well as, wind weighting functions and structural loads on the vehicle. Aerodynamic data were obtained with the aid of a semi-empirical computer code. The final rocket design (fig. 2 and tab. 1) should achieve a maximum range of 19 km and a burn-out velocity of Mach 2.2. Final testing could not be carried out, due to budgetary problems.

The initial version of the HERA motor consisted of four chambers with cylindrical grains mounted to a single expansion nozzle. Figure 3 shows the nozzle section made of nickel-based super alloys and the graphite insert.

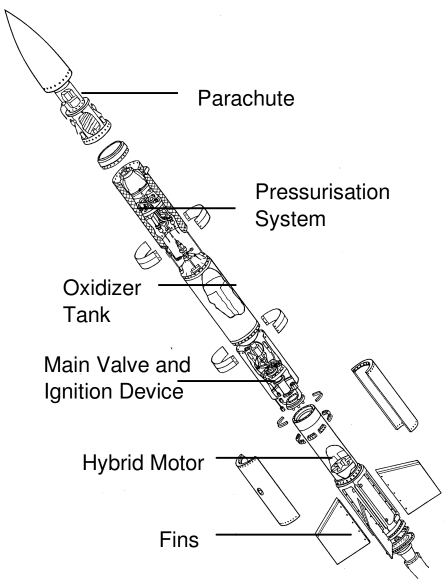


Figure 2: Final Configuration of HERA

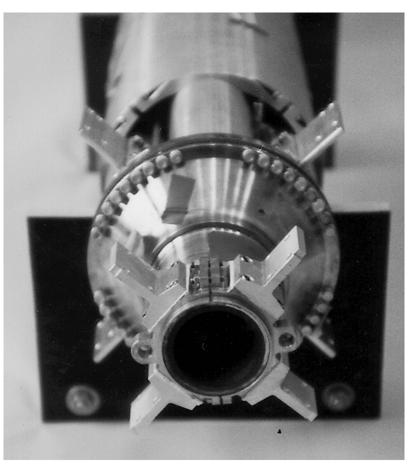


Figure 3: Nozzle Assembly of Rocket

Table 1. Characteristics of Final HERA Version

Launch mass	65.4 kg
Oxidizer mass	26.4 kg
Fuel mass	7.0 kg
Launch thrust	3250 N
Average chamber pressure	32 bar
Burning time	21.6 s
Caliber	0.195 m
Length	2.90 m

For the development of the hybrid propulsion five test series were planned to investigate important aspects of the hybrid rocket design (tab. 2 and 5).

Table 2. Test Series

Test Series	Objectives
Initial Firings	♦ Tests of Motor Components
Internal Ballistic Tests	♦ Combustion Efficiency ♦ Fuel Regression Parameters ♦ Ignition Behavior ♦ Determination of Best Operating Conditions
Thrust Tests	♦ Complete Fuel Utilization ♦ Reproducability ♦ Nozzle Material and Performance
Testbed Firings	♦ Ignition Behavior ♦ Combustion Stability ♦ Qualification of Oxidizer Feed Unit
Qualification Firings	♦ Test of the Entire Propulsion System

Hybrid Propellant Combination Hydroxyl-terminated Polyisoprene /RFNA

The selected solid fuel hydroxyl-terminated polyisoprene (KURARAY LIR-503) is a common industrial product. Diphenylmethandiisocyanate (MDI) was selected as the curative due to its favorable toxic characteristics. The fuel was cast into paper phenolic cartridges. Properties of the fuel are listed in table 3. The selected oxidizer was RFNA (83% HNO_3, 16 % N_2O_4 and 1 % H_2O). The stochiometric mixture ratio with hydroxyl-terminated polyisoprene is 5.107. The maximum theoretical charcteristic velocity is 1540 m/s with a O/F-mixture ratio of 4.3.

Table 3. Properties of Hydroxyl-terminated Polyisoprene (LIR-503)

Reduced Sum Formula (cured state)	$CH_{1,586}N_{0,0022}O_{0,0035}$
Prepolymer:	
Molecular Weight	25000
Funktional Group	OH
Number of OH-groups per Molecule	2.5
Jod Number	368
Viscosity (38°C)	800 poise
Density (20°C)	0.92 g/cm^3

The propellants hydroxyl-terminated polyisoprene and RFNA are not hypergolic, therefore all test firings used a mixture of 3 parts Fe-acetonylacetate and 2 parts 2,6-dimethylaniline as igniter. 15-20 g of the ignition mixture was melted and applied to the grain surface, where it solified very quickly (fig. 4)

.

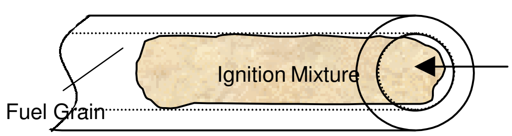


Figure 4: Application of Ignition Mixture

Test Firings and Setups

Initial Tests

Two different motors were built for the testing of motor components, i.e. injectors, nozzle configurations and materials. The first motor used a regenerative cooled steel nozzle coated with zirconium(IV)oxide and a solid cone spray injector (fig. 5). The second motor used an uncooled nozzle structure made of Haynes alloy with a ZrO_2-coated steel nozzle insert. A hollow cone spray injector was used. Both motors used a mixing chamber lined with graphite and a mixing diaphragm. The motor case was made of aluminum. Nine firing tests were performed.

Use of mixing diaphragm and special mixing chamber was given up, due to it complexity. The following motor components and materials were selected for the rocket:

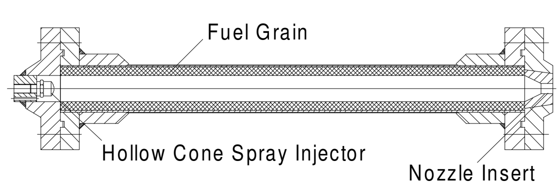
- Hollow cone spray injectors
- High-density, fine-grade graphite nozzle inserts
- Fuel grain design using cylindrical ports and an initial L/D of 18 and initial port radius of 17.5 mm

Figure 5: Initial Test Motor 1

Internal Ballistic Tests

The internal ballistic test serie was used to determine the fuel regression rate as a function of oxidizer mass flux, chamber pressure and grain port radius. Additionally, the characteristic velocity and combustion efficiency were obtained. The data were employed to select the operating conditions of the HERA motor. A simple test motor (fig. 6) was used for evaluating the internal ballistic characteristics of hydroxyl-terminated polyisoprene/RFNA. The test conditions span an interval of oxidizer mass flux from 4-15 g/s*cm², oxidizer mass flow from 95-210 g/s, final port radius ratio 1.1-1.8 and chamber pressure from 8 –18 bar. More than 20 tests were performed. They exhibited very smooth and stable combustion behavior.

Figure 6: Internal Ballistic Test Motor



Thrust Tests

In this series, the grain was fired for the full duration under HERA operating conditions. The thrust, specific impulse, thrust coefficient and efficiency were determined. Furthermore, these tests produced fuel regression rate data. Nine test firings were performed (fig. 7). A modified version of motor 2 from the initial testing was utilized (fig. 8).

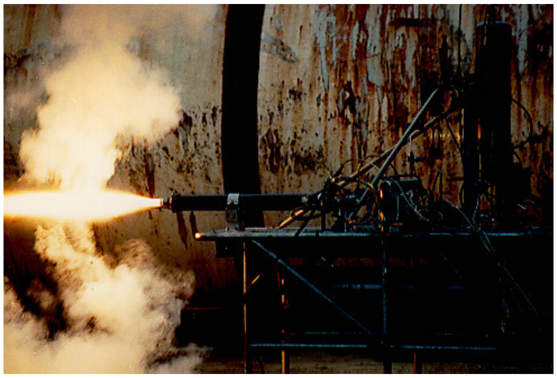


Figure 7: Thrust Test

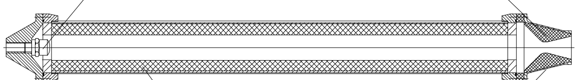


Figure 8: Configuration of Thrust Test Motor

Testbed Firings

The testbed motor was a simplified version of the HERA motor with a maximal burning time of 6 s. The nozzle entrance volume was simulated by an additional chamber (fig. 9). Nozzle and chamber assembly were of mild steel. A 2 mm thick fuel layer was sealed to the chamber wall. The investigation of ignition behavior, combustion stability and performance of the testbed with the rocket oxidizer feed unit was performed with this version. Four test firings were performed (fig. 10).

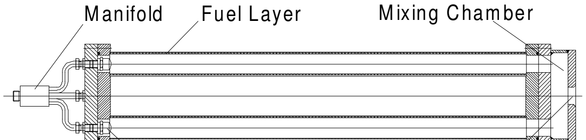


Figure 9: Configuration of Testbed Motor

Figure10: Testbed Firing

Qualification Firings

The objective of this testing was the qualification of the propulsion system for flight testing (tab.4). The test stand was fully instrumented for monitoring of the all firing data (fig. 11).

Figure 11: Preparation of Qualification Test

Table 4: Qualification Parameters of Initial HERA Motor

Chamber Pressure	17.0 bar
Tank Pressure	24.0 bar
Thrust	1780 N
Specific Impulse	2020 Ns/kg
Combustion Efficiency	0.93
Average Mixture Ratio	4.0
Burning Time	37.5 s

Table 5: Test Configuration Matrix

Motor	Spray Injector	Nozzle Throat Area (cm²)	Nozzle Insert Material	Mixing Chamber L^* (m)	Nominal Burning Time (s)	Fuel Grain Dimensions r_1 r_0 l_G (cm)	Nominal Thrust (N)
Initial Test	Full Cone	1.8 – 2.5	Steel+ZrO_2	0.5	50	different	400
Internal Ballistic Test	Hollow Cone*	1.77	Nimonic Alloy	None	32	3.2*1.75*62	400
Thrust Test	Hollow Cone*	1.77-1.92	Graphite	None	32	3.2*1.75*62	400
Testbed	Hollow Cone*	7.55-8.16	Mild Steel	0.2	6	3.2*1.75*72	1700
Initial HERA	Hollow Cone*	7.55	Graphite	0.3	37.5	3.4*1.75*72	1780

* average diameter of droplets: 0.4 mm by an injector pressure difference of 10 bar

Discussion of Results

Selection of Operating Conditions

The optimum chamber pressure and oxidizer mass flux for the investigated motor configuration given through the fuel grain geometry, nozzle throat area and injector parameter were examined. The injected oxidizer mass flow is plotted against chamber pressure for the internal ballistic and thrust test series in figure 12 (averaged values). A nearly linear correlation up to a mass flow of 170 g/s was observed. There was no increase in chamber pressure above 170 g/s. The fuel mass flow was lower than expected. The mixture ratio becomes oxidizer rich and c* declines. The average oxidizer mass flux above 170 g/s was greater than 11.5 g/s*cm².

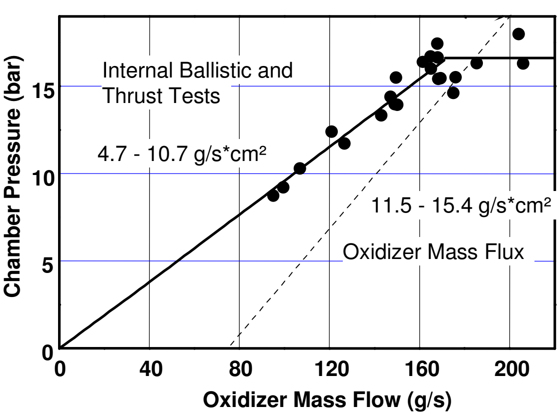


Figure 12: Chamber Pressure versus Oxidizer Mass Flow (single port grain)

Therefore the average oxidizer mass flux of the rocket was set to 9 g/s*cm². The expected average chamber pressure for a single port configuration without mixing devices for 170 g/s is 16.5 bar. Firings with the HERA motor testbed confirmed this expected value.

Ignition

A simple ignition method was used as mentioned above. The grain surface was coated with an hypergolic ignition mixture. It proved to be inadequate in achieving a reliable and predictable ignition and caused several test failures. The mixture was applied only to the first few centimeters of the grain port length for the initial and internal regression test series. There was no pressure overshoot observed, but several tests quenched. Another undesirable effect of this ignition method was an increased fuel regression due to massive thermal ignition peak during the first second of burn (fig. 13). This effect contributed to the regression data errors.

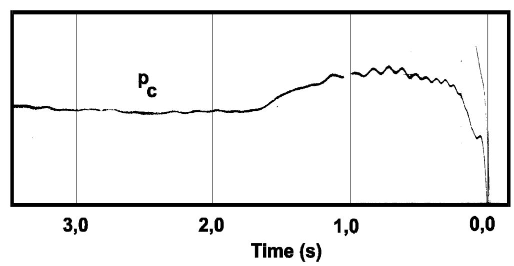


Figure 13: Chamber Pressure versus Time Test (Internal Ballistic Test)

For further testing the ignition mixture was applied to nearly half the length of the port to avoid quenching. Correct ignition was achieved, but ignition pressure varied widely. Extreme ignition pressure peaks destroyed several test motors. An ignition pressure peak higher than 100 bar caused the destruction of the HERA motor in the qualification test series.

Combustion Stability

The initial and internal ballistic tests displayed very smooth combustion (fig.14 a 15).

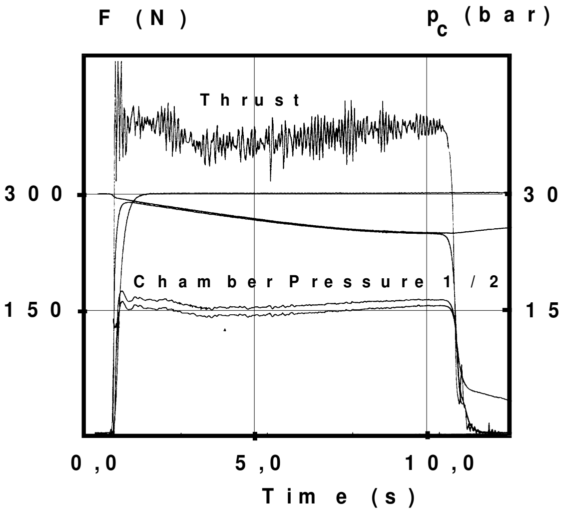


Figure 14: Parameter of a Initial Test versus Time

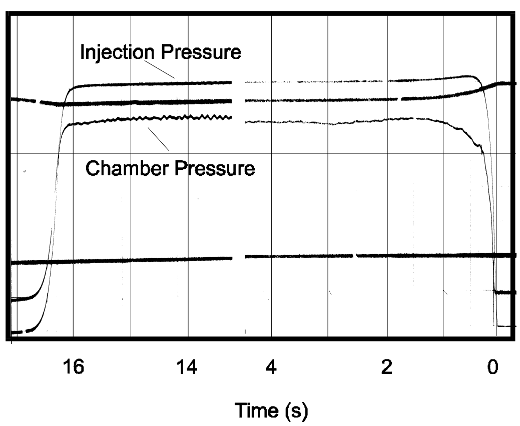


Figure 15: Typical Smooth Internal Ballistic Test

The first thrust test firings showed some rough combustion, believed to have been caused by the mounting of the complete setup on flexible thrust measurement facility. Modifications of the eliminated the rough combustion. The testbed firings showed an unstable operation with very high amplitude and oscillations at 8 Hz. (fig. 16).

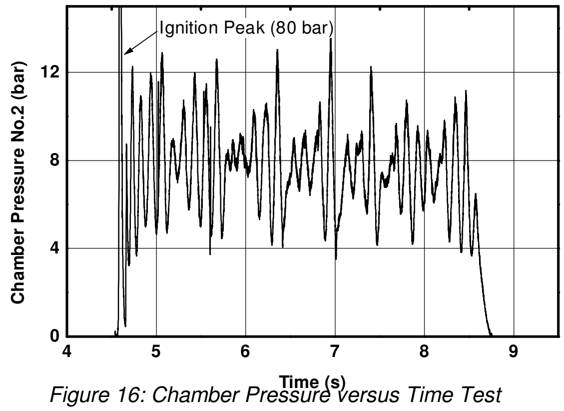


Figure 16: Chamber Pressure versus Time Test

Characteristic Velocity and Combustion Efficiency

The combustion efficiency defined as $\eta_{c*} = c*_{ex} / c*_{th}$ characterize the completeness of combustion. The theoretical characteristic velocity and the experimental results of the internal ballistic and thrust test series are shown in figure 17. The theoretical performance was calculated with computer code [2], which is based on the method of Bathelt and Volk [3]. The ignition mixture values were included in the calculations. A quadratic polynom was fitted to the experimental results ($c*_{ex} = 986.35+194.1\Phi-23.0\Phi^2$). The average combustion efficiency was 0.91. This value was calculated using the mean values of the polynom and the theoretical curve. The maximum $c*_{ex}$ of 1400 m/s was at a mixture ratio of 4.3. The average η_{c*} of the testbed firings was 0.92.

The chamber pressure at the nozzle entrance was used for calculation purposes. A pressure drop between injection head and nozzle entrance for this grain configuration (without mixing diaphragm) was typicaly 0.3 bar. In the initial test firings (with mixing diaphragm and chamber) a value of 1 bar was measured.

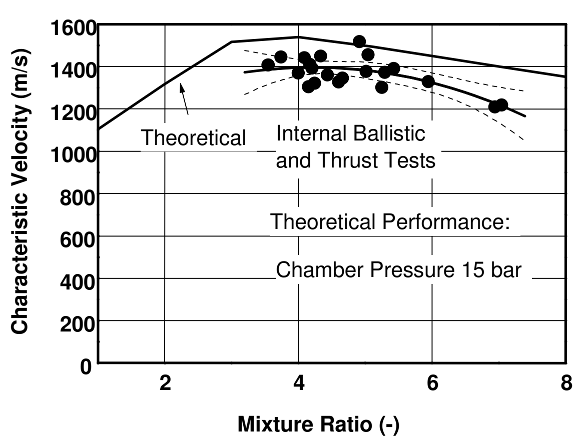


Figure 17: Experimental & Theoretical Characteristic Velocity versus Mixture Ratio

Specific Impulse and Thrust Coefficient

The experimental specific impulse results were fitted by a quadratic polynom. The specific impulse efficiency, defined as $\eta_{Is} = I_{s\,ex} / I_{s\,th}$, was 0.90. A specific impulse of 1910 Ns/kg was achieved with a mixture ratio of 4.3. The nozzle or thrust coefficient effectiveness, defined as $\eta_{CF} = \eta_{Is} / \eta_{c*}$, was 0.99. The theoretical calculations, based on a frozen equilibrium flow assumption, probably under-predicted the nozzle performance (A_e/A_t = 4.49), which would lead to this high value. The thrust coefficients are shown in figure 18.

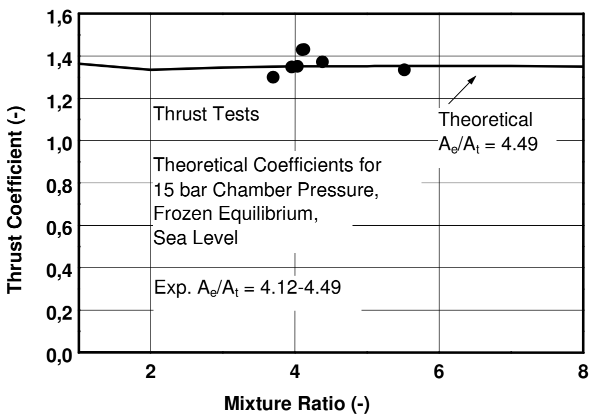


Figure 18: Experimental & Theoretical Thrust Coefficient versus Mixture Ratio

Fuel Regression

The overall average regression rate is based on the assumption that the surface of the fuel grain regresses cylindrically throughout the firing.
It is defined as:

$$dr/dt = (\Delta m_{fu}/\pi\rho l_G + r_0^2)^{0.5} - r_0))/t_b \,.$$

A data analysis program [4] was used to determine the fuel regression rate as a function of the oxidizer mass, chamber pressure and grain port radius. The replacement of the time dependent motor parameters by their time-averaged values for this grain configuration results only in negligible errors in the regression correlation parameters [1], [5], since the final to initial grain port radius ratio was small only 1.8. The fuel regression results gave the best correlation with oxidizer mass flux and port radius as parameters:

$$dr/dt = a\, G_{ox}^{\alpha} r^{\beta} \,.$$

The correlation constants a, α and β are subject to relatively high errors (tab. 6). This results from the limited accuracy of measurements of the parameters involved. The ignition mixture also falsified the average regression rate (fig. 13). The small size of the test motors also leads to additional errors in the measurements [6].

It was observed that a significant change in fuel regression behavior accured at an oxidizer mass flux of 8-10 g/s*cm². Therefore, two regression correlation values as function of the oxidizer mass flux above and below this value are given in figure 19 and table 6 (port radius as parameter). The value of the empirical oxidizer mass flux exponent (0.6 - 0.9) below about 9 g/s*cm² is well known from other authors. Above 9 g/s*cm² the exponent is unusually small. The correlation results display a surprisingly strong dependency on the port radius (tab. 6).

Table 6: Correlation Parameter and Relative Errors

Parameter	a	α	β
Interval-1	0.0057	0.8	0.81
Error	± 135 %	± 22 %	± 50 %
Interval-2	0.0078	0.254	1.10
Error	± 41%	± 56 %	± 28 %

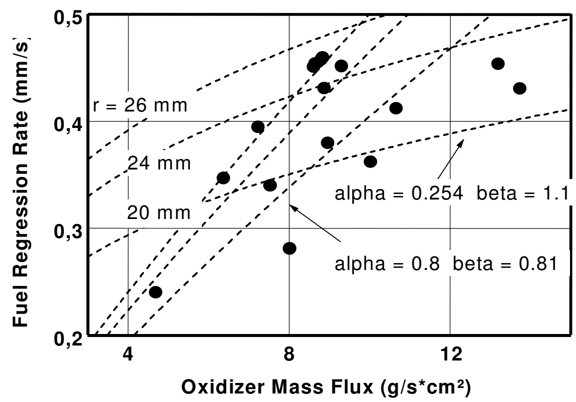


Figure 19: Fuel Regression Rate versus Oxidizer Mass Flux

Another approach to evaluate regression behavior is to analyse tests over a small range of average oxidizer mass flow, chamber pressure and with a constant initial grain port radius. This procedure simulates a one time firing test. The average oxidizer mass flux was above 9 g/s*cm² for these tests. A plot of the final port radius versus burning time for these tests shows a constant fuel regression rate (fig. 20).

The increasing fuel mass flow resulted in a decreasing O/F mixture ratio (fig. 21). This is confirmed by observation of the plume color of long burning tests, which change from transparent to brilliant yellow and a slightly increased chamber pressure (max. 10%). These results support the statement that above ~ 8-10 g/s*cm² the port radius is the determining factor in the fuel regression rate.

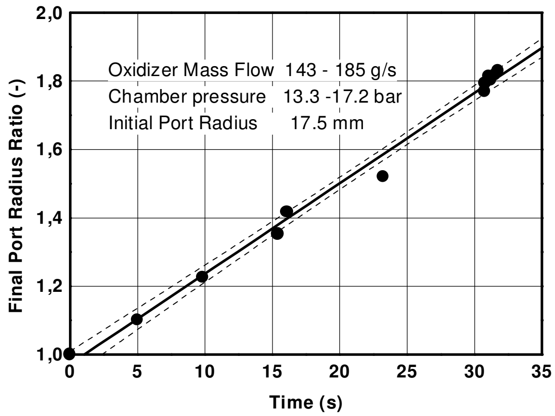


Figure 20: Final Port Radius Ratio versus Time

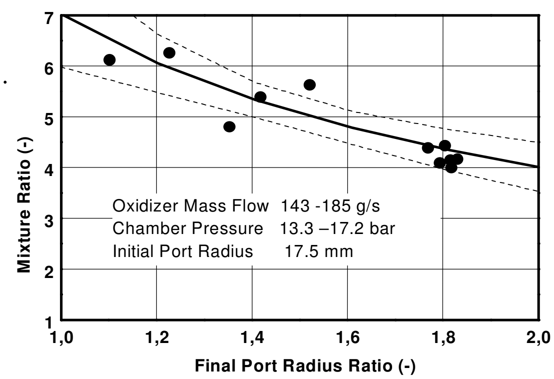


Figure 21: Mixture Ratio versus Final Port Radius Ratio

Error Propagation Calculation

Characteristic velocity, specific impulse, oxidizer mass flux and fuel regression rate are derived values, that cannot be measured directly. Therefore, error propagation calculations are important to determine the accuracy of these parameters. The range and error range of measured data are given in table 7. Using the method of error propagation, the estimated errors of the important motor parameters were calculated (tab. 8).

Table 7: Estimated Errors in Measured Basic Test Motor Parameters

Motor Parameter	Value	2σ-Error (%)
Chamber Pressure	8 - 18 bar	1.0 – 2.5
Oxidizer Mass Flow	95 - 210 g/s	1.4 – 2.8
Burning Time	5 –30 s	0.5 – 3.0
NozzleThroat Area	1.76 – 8.0 cm²	1.2 – 2.7
„Effective“ Port Length*	3.0 cm	5.0
Initial Port Radius	1.75 – 2.5 cm	3.0 – 4.0

* = fictive port length uncertaincy due to variation of length dependent regression

It is interesting to note, that the scatter of the data points in the graphs ($c^* \pm 2.5 - 3.3$ %, $I_s \pm 4.6$ % and dr/dt $\pm 3.1 - 7.9\%$) are on the same order as the estimated errors calculated by the method of error propagation.

Especially the parameters oxidizer mass flux and fuel regression rate, show significant errors, due to conceptional uncertainty in port diameter and active fuel regression time. Furthermore, the length dependent fuel regression gave regression rate errors. (fig. 22).

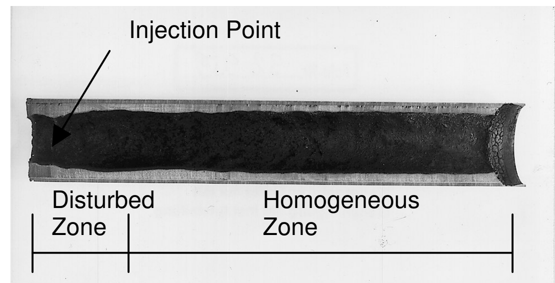


Figure 22: Port Length Dependent Regression

Table 8: Estimated Errors of Derived Motor Parameters

Parameter	Value	2σ-Error (%)
Oxidizer Flux	4-15 g/s*cm²	6.0 – 8.0
Fuel Mass Flow	20- 40 g/s	0.5 – 3.0
Total Mass Flow	110 – 250 g/s	1.3 – 2.7
Mixture Ratio	3.5 –7.0	1.5 – 4.0
Characteristic Velocity	1400 m/s	2.0 – 4.6
Fuel Regression Rate	0.25 – 0.45 mm/s	3.5 – 8.0

Nozzle Inserts

Several materials were tested for their suitability for usage as nozzle inserts. Y_2O_3-stabilized ZrO_2 and Haynes alloy proved to be inadequate due to physical defects or excessive weight. After testing, it was determined that high-density, fine-grade graphite was the most suitable material for use in this application.

Summary

A small experimental hybrid rocket project using hydroxyl-terminated polyisoprene/RFNA as propellants was started with the objective of obtaining a range of 20 km and a start mass of less than 70 kg. The testing of the initial motor configuration showed the following results.

The thrust for the given fuel grain geometry is limited by a maximum initial oxidizer mass flux or flow. The combustion and thrust efficiencies were 0.91 and 0.90 without special mixing diaphragm. The fuel regression rate showed an expected oxidizer mass flux dependency up to a limiting value and afterwards remained nearly constant. The grain port radius exerted an influence on the regression rate. The combustion behavior in the single grain configurations was smooth. Instabilities were only in the quadruple configuration observed. Coating the grain with a hypergolic ignition mixture proved to be an unreliable ignition method.

A design review of the initial motor configuration using the above results and those of 6-DOF flight simulations was performed. The simulations indicated the need for higher thrust levels. Therefore, the following modifications were applied. A new single port grain was designed to achieve higher oxidizer mass flow with limited mass flux. The ignition problems were solved by an external pyrotechnique ignition chamber.

Error analysis showed that the relative errors of the derived motor parameters are relatively large due to the small size of the tested configuration. The results therefore may not be applicable to high-performance hybrid motors.

Acknowledgments

The author is grateful to those who contributed much to success of HERA: Gunther Matthäus, Dieter Korte, Dieter Knaak, Horst Frommelt (DASA, Bremen), Sepp Zeug (Sepp Zeug KG, Böblingen), Robert Schmucker (Schmucker Technologien, Munich) and my parents. I wish to thank Gregor Grigoriou (IBG, Munich) and Ralph Cook (BUCK, Bad Reichenhall) for their help in the preparation of this paper and the BUCK Werke for the financial support for the presentation.

References

[1] M. Grosse
„Brennversuche mit dem hybriden Treibstoffsystem Hydroxyl-terminated Polyisoprene/RFNA“
Interner Bericht IRS 90-ST
Institut für Raumfahrtsysteme
Universität Stuttgart, 1990

[2] H. Bathelt and F. Volk
„The ICT-Thermodynamik Code“
Fraunhofer Institut für Chemische Technologie (ICT)
Pfinztal, Germany

[3] H. Bathelt and F. Volk
„Application of the Viral Equation of State in Calculating Interior Ballistics Quantities“
Propellants and Explosives 1, 7-14 (1976)

[4] ORIGIN 4.1
Microcal Software, Inc.
Northampton, MA 01060 (1996)

[5] R.H. Schmucker
„Hybridraketenantriebe - Eine Einführung in theoretische u. technische Probleme“
Wilhelm Goldmann Verlag München, 1972

[6] B. Greiner und R.A. Frederick, Jr.
„Labscale Hybrid Uncertainty Analysis“
AIAA/SAE/ASME/ASEE 31th Joint propulsion Conference and Exhibit
July 10-12, 1995, Paper No. 95-3085
San Diego, CA