IRS-93-P7

NASA Grant No. NAGW 1736

Development of a High Specific 1.5 - 5 kW Thermal Arcjet

Interim Report No. 2

Report Period : November 1992 - April 1993

Autors:

M. Riehle, B. Glocker, M. Auweter - Kurtz, H. Kurtz

Submitted to:

NASA-Lewis Research Center
Cleveland, Ohio

Institut für Raumfahrtsysteme
Universität Stuttgart
Pfaffenwaldring 31
D-7000 Stuttgart 80
May 1993

Prof. Dr.-Ing. habil. M. Auweter-Kurtz

Contents

Development of a High Specific 1.5 - 5 kW Thermal Arcjet

M. Riehle, B. Glocker, H.L. Kurtz, M. Auweter-Kurtz
University of Stuttgart
Institut für Raumfahrtsysteme

The work described in this interim report is supported by NASA grant NAGW-1736.

Abstract

A research and development project on the experimental study of a 1.5 - 5 kW thermal arcjet thruster was started in 1992 at the IRS. Two radiation cooled thrusters were designed, constructed and adapted to the test facilities, one at each end of the intended power range. These thrusters are currently subjected to an intensive test program with main emphasis on the exploration of thruster performance and thruster behaviour at high specific enthalpy and thus high specific impulse. Propelled by simulated hydrazine and ammonia, the thruster's electrode configuration such as constrictor diameter and cathode gap was varied in order to investigate their influence and to optimize these parameters. In addition, test runs with pure hydrogen were performed for both thrusters.

Introduction

The research and development activities in the field of thermal arcjets have recently been focused on lower power levels of about 1-5 kW, which have better chances for application as station keeping devices on satellites in the near future. In the U.S. a radiation cooled 1 kW-class thruster has been flight qualified, and other projects world-wide will follow the same objectives, e.g. the ATOS project (Arcjet Thruster on Oscar Satellite) currently under development at the IRS. It will be installed on the amateur radio satellite AMSAT P3-D to be launched with the first ARIANE 5 rocket in 1995 *[1]*.

At the IRS a research program has been started for detailed examination of thruster behaviour and performance over a broad operating range. The test program which has been recently started is intended to establish the operational characteristics of low power arcjets with different electrode configurations by varying the cathode gap and the constrictor diameter with different propellants. Combined with performance calculations, the knowledge gained will service the geometrical optimization at a desired power level.

Furthermore, the influence of the cooling concept on the arcjet performance will be studied by switching from radiation cooled to regeneratively cooled structures, especially with new designs for the anode nozzle. The final intention will be the development of a regeneratively cooled thruster with optimized geometries.

The whole project can be split into two separate development branches which will now be described independently.

1. 1.5 kW Power Level

1.1. Thruster Design Description

A modified version of the IRS ARTUS thruster was used for the tests on the 1.5 kW level (Fig. 1.1). The laboratory version of this thermal arcjet (dubbed ARTUS for Arcjet Thruster, University of Stuttgart) *[1]* is derived from the laboratory version of the arcjet developed at the NASA Lewis Research Center in Cleveland *[2]*.

The long cylindrical housing of the ARTUS thruster allows an easy disassembly of the thruster to exchange components like the nozzle or the injector disk. A flange type nozzle was chosen to allow the use of different nozzle inserts without lengthy adaptations of tapered fits.

Compared to the NASA design two major changes were incorporated. First, the length of the thruster was reduced, because the propellant feed line could be connected directly to the thruster housing, the feed line did not have to be isolated electrically from the anode potential.

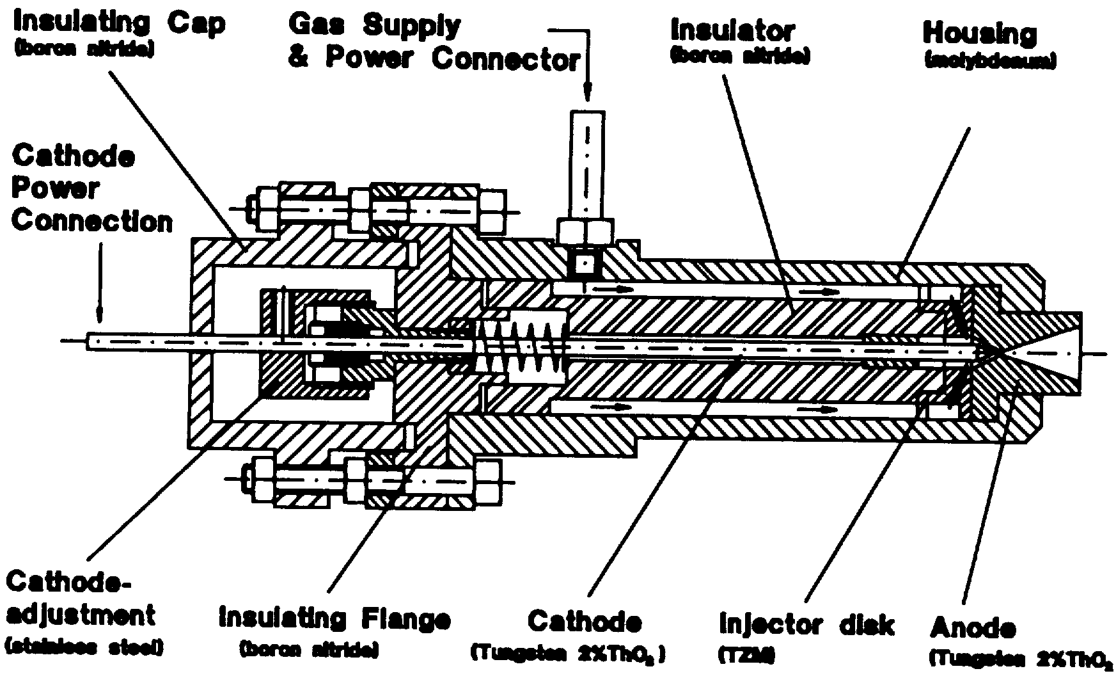

Fig 1.1: Scheme of 1.5 kW thruster

The second - and more important - modification was the mechanism to adjust the gap between anode nozzle and cathode tip. The ARTUS design uses a micrometer type assembly. With this modified design the cathode gap can be easily adjusted with an accuracy of 0.02 mm. Furthermore, it can be rechecked for changes after each experiment, where the criterion of the contact between cathode and anode is defined as a cathode gap of "0.00", which is measured with an ohmmeter. Eccentricities between anode and cathode sometimes allow the cathode to be moved an additional distance up to 0.07 mm inwards. However, no precise reproducible criterion for the gap could be established using only mechanical contact between anode and cathode. Sealing along the cathode feed through is provided by a gland seal squeezed by a gland stud.

Due to the high temperatures expected at the nozzle throat and in order to reduce erosion near the throat edges, thoriated tungsten (2% ThO_2) was used as material for the anode nozzle insert. The nozzle was specified with a 0.60 mm throat diameter, a 0.25 mm throat length, and an exit plane diameter of 12.00 mm (expansion ratio 1:400). These specifications were kept within 0.02 mm. The same tungsten material was used for the cathode. The cathode diameter is 3 mm, the cathode tip half angle was 20^o - 30^o.

TZM (a molybdenum based alloy) was chosen for the thruster housing. TZM withstands high temperatures (2850 K), and it can be machined easier than tungsten. The injector disk is made also from TZM. Its two injection boreholes provide the usual tangential propellant injection as well as a small forward component obtained by a 20^o forward tilt.

The main insulator, the rear flange, and the insulation cap, the components providing electrical insulation, are manufactured from boron nitrite. The remaining parts, like the propellant feed line and cathode adjustment mechanism, are manufactured from stainless steel, because their temperature loads were estimated to remain below 920 K. The spiral spring that presses the main insulator and the nozzle against the forward housing flange to provide the sealing between nozzle and housing is manufactured from a high temperature Inconel alloy. All seals were punched from 0.35 mm thick graphite foil.

Four different nozzle types were intended to be investigated with all three propellants. Table 1.1 shows the geometries of the inserts.

ARTUS-4	Constrictor Diameter [mm]	Constrictor length [mm]	Area Ratio	Nozzle End [mm]
Nozzle 1	0.5	0.377	576	12
Nozzle 2	0.6	0.6	400	12
Nozzle 3	0.8	1.04	225	12
Nozzle 4	1.0	1.49	144	12

Table 1.1 : Nozzle Geometries for 1.5 kW Thruster.

These different nozzle inserts combined with the variation of cathode gaps from 0.8 to 1.3 mm were used to investigate the influence of electrode configuration on thruster behavior.

1.2. Experimental Facilities

Performance investigations were conducted in a 2.0 m long stainless steel vacuum chamber with 1.0 m diameter. The chamber is equipped with a three stage roots pump group with a pumping capacity of 1300 m^3/h. These pumps are capable of sustaining a vacuum of 2.0 Pa at a mass flow of 50 mg/s simulated hydrazine.

The PC-type data collection and their reduction for the performance measurements was the same as described in *[3]*.

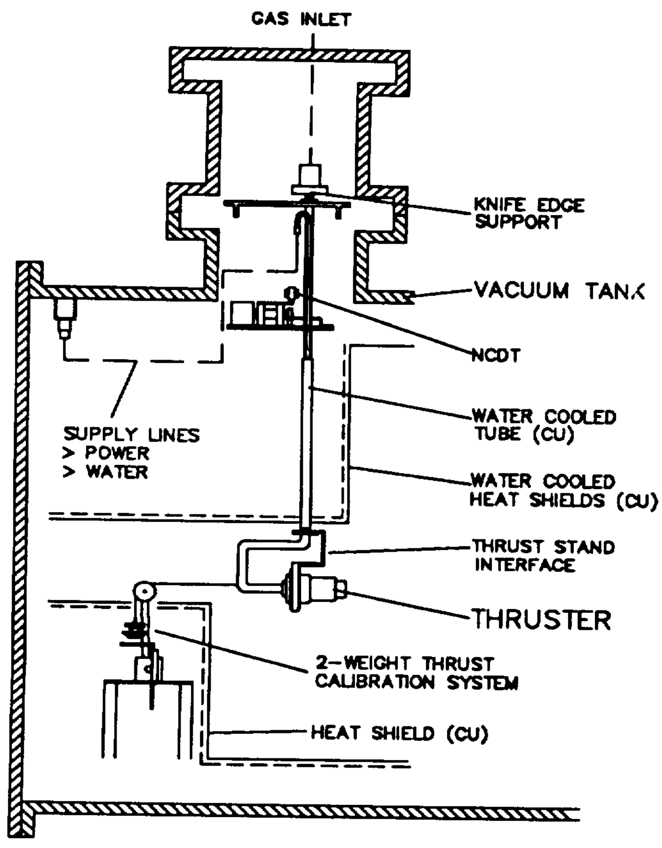

Fig. 1.2: Pendulum Type Thrust Balance and test facilities for 1.5 kW Arcjet.

The measured performance parameter include :

- arc voltage & current
- thrust
- mass flow
- ambient tank pressure
- feed line pressure (to monitor changes in thruster behavior)
- anode temperature

The thrust in this facility is measured with a pendulum type thrust balance, the pendulum displacement is determined with a non-contacting linear displacement transducer (NCDT). Fig. 1.2 shows its schematic. The pendulum arm is coaxial to allow the feeding of the propellant gas and the current. It is water-cooled to prevent thermal effects such as expansion. This pendulum could measure the thrust with an uncertainty of about 0.002 N after being calibrated with two weights of about 0.1 N each.

The propellant mass flow rates were measured and controlled by commercial mass flow controllers (MFCs), based on the thermal conductivity of the propellant gases. Due to problems with these controllers, discussed later on, the mass flow rates are determined additionally with a high precision weight balance.

For temperature measurements at the surface of the hot anode a linear pyrometer is used. The emissivity of the tungsten anode was determined to be 0.4 at a wavelength of 961.3 nm.

1.3. Operating Experience and Results

Until now experiments with nozzle 1,2 and 3 were conducted. The different constrictors were investigated with the three propellants at different cathode gaps. The thruster was ignited at a current level of 12 to 15 A and operated until thermal equilibrium was reached. Then a new current level was set and again kept constant until equilibrium. The described points are the points at thermal equilibrium. After measuring different operation points from 1-2 kW at the high mass flow rate, the current was set such that 1.5 kW el. power were measured. Having recorded these operational data, the mass flow rate was lowered to the next intended level and the current level was adjusted such that the power level remained at 1.5 kW. Having achieved equilibrium, the data were recorded, and the next mass flow rate was set. Then, at a low mass flow rate, some operation points at different power levels were adjusted and recorded.

It was recognized that the anode temperature changes very fast after changing the current and it rapidly achieves the level of equilibrium with a difference of 20 K. After that the temperature changes more slowly to that level. The temperature difference between the nozzle end and a point near the housing was found to be only about 5-10 K, which is nearly not measurable

1.3.1. Problems with the Mass Flow Rates Adjustments

At the beginning of the investigations with this thruster the test runs were conducted using only the mass flow controllers to measure the mass flow rates. During the first tests runs some problems with the reproducibility of the received data were recognized at some repeated test points. The discrepancy was not systematical, it depended on the operation point of the thruster such as the line feed pressure. Thus one reason can be found in the calibration of the flow controllers, which is done by the supplier using a different gas at only one pressure level, and by correcting the different heat load by a constant factor.

Therefore, another independent and reliable method had to be found to check the flow controllers. After having established a high precision weight balance, the actual mass flow rate is now calculated additionally from an on-line weighing of the gas bottles (and thus of their mass loss) with an accuracy of 1g. The disadvantage of this procedure is the great time consumption for each test point of about 15 to 20 minutes for ammonia or the N_2/H_2 mixture, and up to one hour for hydrogen as propellant, to reduce the inaccuracy of the calculated mass flow below 1-2%.

The check of the flow controller set used for the 1.5 kW tests, produced the following results:

Hydrogen : The values of the used hydrogen controller matched the weight balance data along the examined mass flow range. Due to that fact the deviations were below 1 % in absolute value and reproducibility, the results obtained in previous test runs could be used.

Simulated Hydrazine : For this controller the recorded values rested permanently about 5 - 8 % above the balance data. The determined offset couldn't be reproduced exactly at the same test conditions. Thus, there must be a non-determinable drift of the controller signal even at constant ambient temperature. The tests already conducted couldn't be corrected by the signal offset, and therefore they had to be repeated. These first data are not presented.

Ammonia : As established for the N_2/H_2 mixture controller, there was also a similar behaviour for the ammonia mass flow sensor. However, deviations of up to 10 % in the opposite direction were found. The mass flow rates determined with the weight balance were always lower than measured with the controller. This means that the measured specific impulses during the first test series were too low. The results of these tests are also not presented in this report.

Due to these reasons the measurement procedure was changed for further tests and for the repeating of previous ones with ammonia and simulated hydrazine. The mass flow controllers now are used only to establish a constant mass flow. So it is time consuming to gain the thruster characteristics, and the detailed determination of thruster operation range, intended in the beginning of the research program, had to be limited to the main power level of the thrusters with 1.5 and 5 kW, respectively.

1.3.2. Experiments with Nozzle No 1

Tests with this small constrictor diameter of 0.5 mm were tried but the thruster couldn't be operated well. As a second approach it was attempted to do the tests with a resistor connected in series to the thruster. With this method it was possible to get some operation points.

Using hydrogen as propellant it was possible to ignite the thruster at 10 mg/s, but during the heating of the thruster the voltage got so high that it exceeded the limit of the power supply and the thruster operation came to halt.

Mass Flow	Current	Voltage	**el. Power**	Thrust	Feed Pressure	Anode Temp.	Thrust Efficiency	**specific Impulse**
[mg/s]	[A]	[V]	**[kW]**	[N]	[bar]	[°C]	[%]	**[s]**
25	11.9	126.4	**1.504**	0.148	4.99	1130	29.1	**592**
22.5	12.9	115.9	**1.495**	0.124	4.39	1223	22.8	**551**
20.0	14.2	104.6	**1.485**	0.106	3.90	1315	18.9	**530**

Table 1.2. : Nozzle 1, Ammonia as Propellant.

With ammonia it was possible to ignite the thruster and get the operation points listed in table 1.2. The measured mass flow was not corrected with the weight balance. It was not possible to reignite the thruster and get more operation points.

With the premixed gas it was not possible to ignite with 45 mg/s. The mass flow was lowered and an ignition was successful at a mass flow rate of 32.5 mg/s, but the thruster sputtered much and didn't work well. Nevertheless, some operation points could be measured and recorded as depicted in table 1.3.

Mass Flow	Current	Voltage	**el. Power**	Thrust	Feed Pressure	Anode Temp.	Thrust Efficiency	**specific Impulse**
[mg/s]	[A]	[V]	**[kW]**	[N]	[bar]	[°C]	[%]	**[s]**
32.5	11.6	129.9	**1.508**	0.183	-	1343	34.2	**563**
30	11.9	125.4	**1.497**	0.178	-	-	35.3	**593**
27.5	12.6	119.7	**1.507**	0.148	5.32	1338	26.4	**538**
25	13.4	112.3	**1.504**	0.132	4.95	1410	23.1	**528**
22.5	13.4	108.9	**1.459**	0.121	4.61	1435	22.3	**537**
20	14.4	104.0	**1.497**	0.108	4.41	1475	19.5	**540**
25	10.0	123.3	**1.233**	0.126	4.72	1320	25.7	**504**
25	12.0	118.3	**1.419**	0.130	4.99	1380	23.8	**520**
25	14.1	114.2	**1.610**	0.132	5.26	1450	21.6	**528**
25	16.0	111.8	**1.794**	0.134	5.53	1540	20.0	**536**

Table 1.3. : Nozzle 1, Simulated Hydrazine as Propellant.

The experience with this constrictor diameter showed that with the available power supply it is not possible to make reliable experiments. Therefore experiments with this nozzle diameter were stopped. Due to the early experimental state, the mass flow rates were not checked and corrected.

1.4. Experiments with Nozzle No. 2 & 3

1.4.1. Hydrogen as Propellant

As mentioned above the data received from previous test runs are reliable and don't have to be re-recorded. The test runs were conducted with mass flow rates between 8 and 25 mg/s. at a constant power level of 1.5 kW. In addition operational characteristics for different power levels were recorded at various points with fixed mass flow rates.

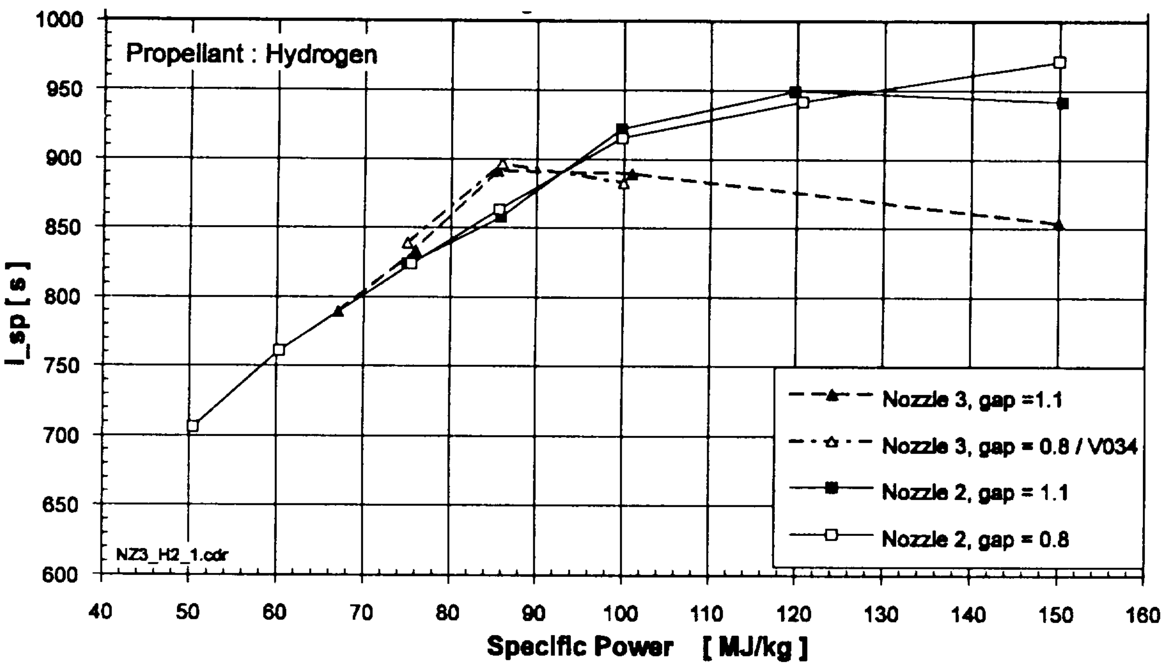

Fig. 1.3 : Influence of Electrode Configuration at Constant Power Level of 1.5 kW.

Fig 1.3 and 1.4 show the specific power versus the specific impulse. Due to the thermal overload of the DC power supply and the high anode temperatures, lower mass flows, thus higher specific enthalpies, couldn't be tested. As a main result it can be seen that the influence of the cathode gap is very small for the same nozzle insert with this propellant. Furthermore, it can be seen that the specific impulses achieved with the 0.6 mm constrictor were significantly higher for lower mass flow rates. For the lower mass flow rates the results for the larger throat diameter reached its maximum value at a lower specific power and then decreased earlier. Higher specific impulse could only be achieved by increasing the input power at mass flow rates above 15 mg/s. This indicates that the diameter of nozzle 3 is already too large for the intended power level of 1.5 kW.

In contrast to the test runs depicted in Fig 1.3, where the power level was kept constant with decreasing mass flow, Fig 1.4 shows the results of the power variation at some low mass flow rates and with two different cathode gaps.

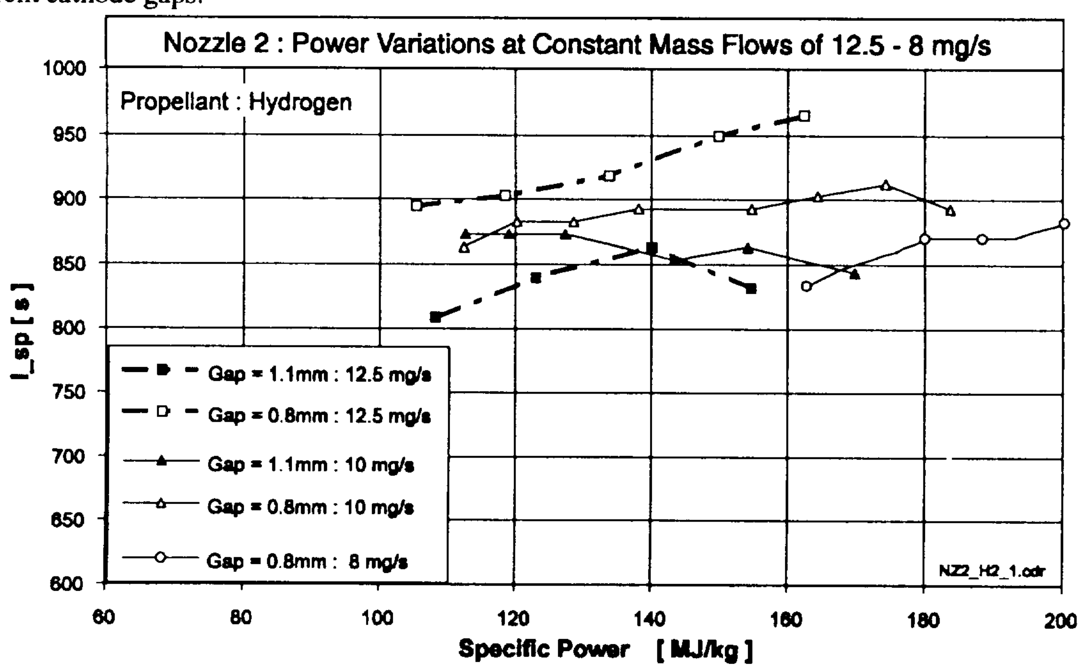

Fig 1.4 : Nozzle 2 : High Specific Enthalpies.

The specific power at 8 mg/s is the highest, but the specific impulse achieved with this configuration is not the best. It was not possible to obtain more than 1000 s specific impulse with this thruster while electric power rested below 1.5 kW.

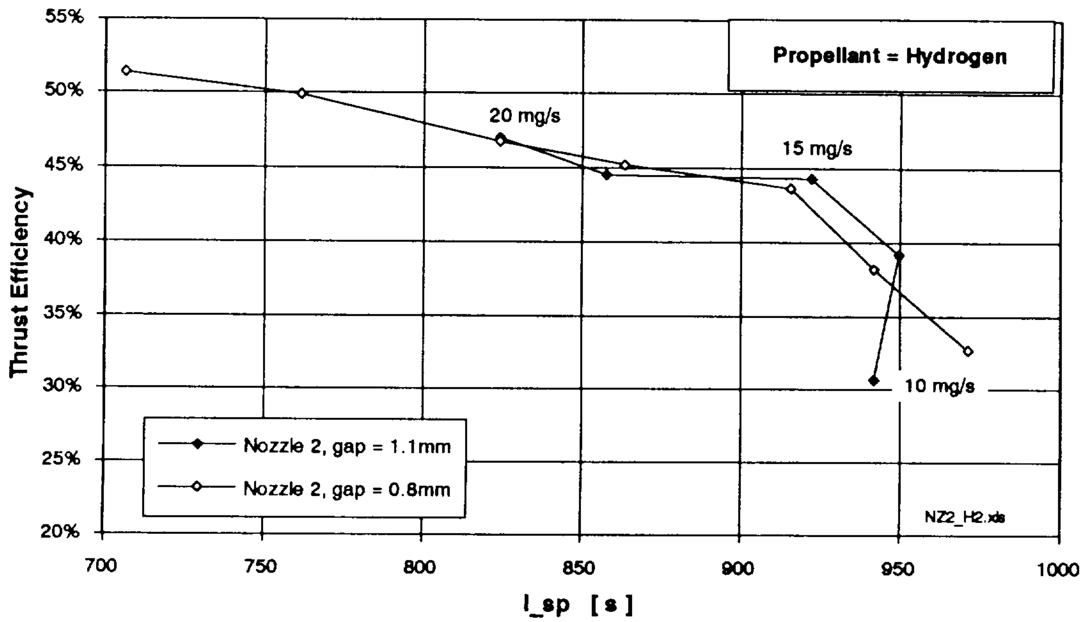

Fig. 1.5 : Thrust Efficiency versus Specific Impulse at 1.5 kW.

In Fig 1.5 the thrust efficiency versus the specific impulse is depicted. It can be seen that the efficiency decreases with increasing Isp (due to the decreasing mass flow rates) and becomes lower than 35 % at 10 mg/s. The sudden and extreme decrease of efficiency at a mass flow rate below 15 mg/s correlates with the curves in Fig 1.3. There the Isp increases for Nozzle 2 more slowly at higher specific power levels, and it even decreases above a certain power level for nozzle 3. This can be explained by the growing thermal losses and when more and more of the input power is consumed by the loss mechanisms.

Interpretating the results with nozzle 2 and 3 at a power level of 1.5 kW we expect that the data which will be meassured with nozzle 4 will produce lower values of Isp and the thrust efficiency will also slow down earlier.

In table 1.4 an overview of obtained data is given for the configurations examined at 1.5 kW.

Mass Flow	Current	Voltage	el. Power	Thrust	Feed Pressure	Anode Temp.	specific Impulse	Thrust Efficiency	specific Power
[mg/s]	[A]	[V]	[kW]	[N]	[bar]	[°C]	[s]	[%]	[MJ/kg]
Nozzle 2 : Throat Diameter = 0.6 mm									
Cathode Gap = 0.8 mm									
30	10.6	142.8	**1.51**	0.216	5.17	-	**706**	51.4%	**50**
25	11.4	132.7	**1.51**	0.194	4.52	-	**761**	49.9%	**60**
20	12.3	122.9	**1.51**	0.168	3.87	-	**824**	46.8%	**75**
17.5	12.8	117.3	**1.50**	0.154	3.6	-	**863**	45.2%	**86**
15	13.3	112.2	**1.50**	0.140	3.28	-	**916**	43.7%	**100**
12.5	14.4	105.0	**1.51**	0.120	2.99	-	**942**	38.2%	**121**
10	15.0	99.9	**1.50**	0.099	2.69	-	**971**	32.7%	**150**
Cathode Gap = 1.1 mm									
20	11.2	133.8	**1.50**	0.168	3.94	-	**824**	47.0%	**75**
17.5	11.8	127.3	**1.50**	0.153	3.61	899	**858**	44.5%	**86**

15	12.3	121.5	**1.50**	0.141	3.31	975	**922**	44.3%	**100**
12.5	13.3	112.4	**1.50**	0.121	3.04	1112	**950**	39.1%	**120**
10	14.5	103.8	**1.50**	0.096	2.72	1376	**942**	30.7%	**150**
Nozzle 3 : Throat Diameter = 0.8 mm Cathode Gap = 0.8 mm mass flow controlled with the weight balance									
22.5	11.2	133.5	**1.50**	0.175	2.92	770	**763**	45.5%	**66**
20	11.8	128.0	**1.50**	0.160	2.69	783	**785**	42.6%	**75**
17.5	12.3	122.0	**1.50**	0.152	2.55	914	**852**	44.0%	**86**
15	13.4	112.0	**1.50**	0.135	2.46	1090	**883**	40.5%	**100**
Cathode Gap = 1.1 mm									
22.5	11.2	134.7	**1.51**	0.181	3.00	783	**789**	48.3%	**67**
20	11.6	130.7	**1.52**	0.170	2.79	810	**834**	47.6%	**76**
17.5	12.2	122.6	**1.50**	0.159	2.59	883	**891**	48.3%	**85**
15	13.8	109.7	**1.51**	0.136	2.41	1030	**889**	40.7%	**101**
12.5	16.0	94.5	**1.51**	0.101	2.20	1184	**793**	27.0%	**121**
10	17.8	84.3	**1.50**	0.087	1.85	1287	**853**	25.2%	**150**
8	18.1	82.7	**1.50**	0.075	1.83	1355	**925**	23.7%	**187**

Table 1.4. Operational Results obtained with hydrogen as propellant.

1.4.2. Hydrazine Simulating Mixture as Propellant

As mentioned above the results of some test runs, conducted before the weight balance was available, are not presented. Nevertheless, the experience acquired in operating the thruster and the test facilities can be used to simplify further tests and to obtain more reliable data.

In a normal test run the thruster was ignited at a mass flow rate of 45 mg/s and with a current presetting of about 12 A. After a heating period the electric power was set to 1.5 kW and kept constant until equilibrium was reached and data were recorded. Then the mass flow rate was lowered stepwise as long as the thruster enabled a stable operating mode or until the anode temperature reached a maximum value of about 1600 °C.

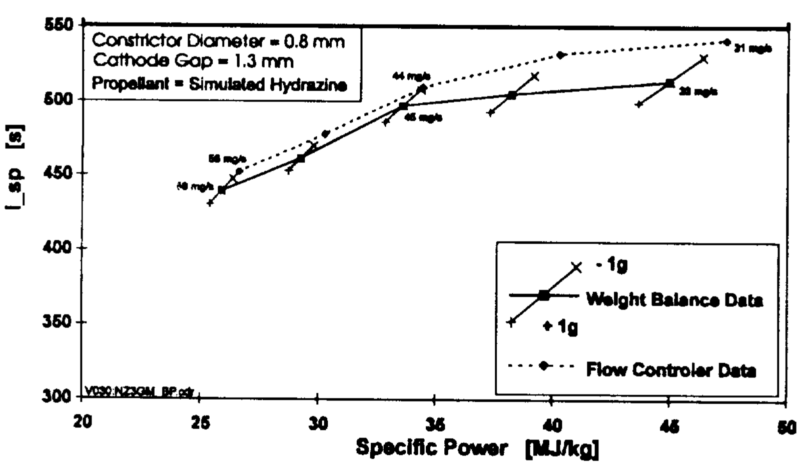

Fig 1.6 : Deviation of Mass Flow Rates.

In Fig. 1.6 a comparison of the mass flow rates, determined with the two methods, are depicted for the first examined thruster configuration and a typical test run. The maximum error bars for the weight balance data are drawn at the recorded operating points. It can be seen that the flow controller values were always higher than the balance data. The deviation increased with lower mass flow rates when the line feed pressure and thus the arc chamber pressure was lower. The results presented in the following sections were based on the mass flow determined with the weight balance.

The graphs of some tests already repeated with nozzle 3 are depicted in Fig 1.7. It can be seen that with this propellant the cathode gap has much more influence than with pure hydrogen. With the shorter cathode gap of 0.8 mm it was possible to achieve higher specific impulses. The graph of the larger gap reached almost the same values at mass flow rates above 45 mg/s and then the curve gradient decreased and a maximum specific impulse of 514 s could be achieved at 33 mg/s. Due to problems with thermal overload of the power supply, it was not possible to obtain higher specific power than 45 MJ/kg with a cathode gap set to 1.3 mm.

Equipped with nozzle 2 and using shorter gaps, the thruster should operate up to higher specific power levels, as suggested by the experience of previous tests.

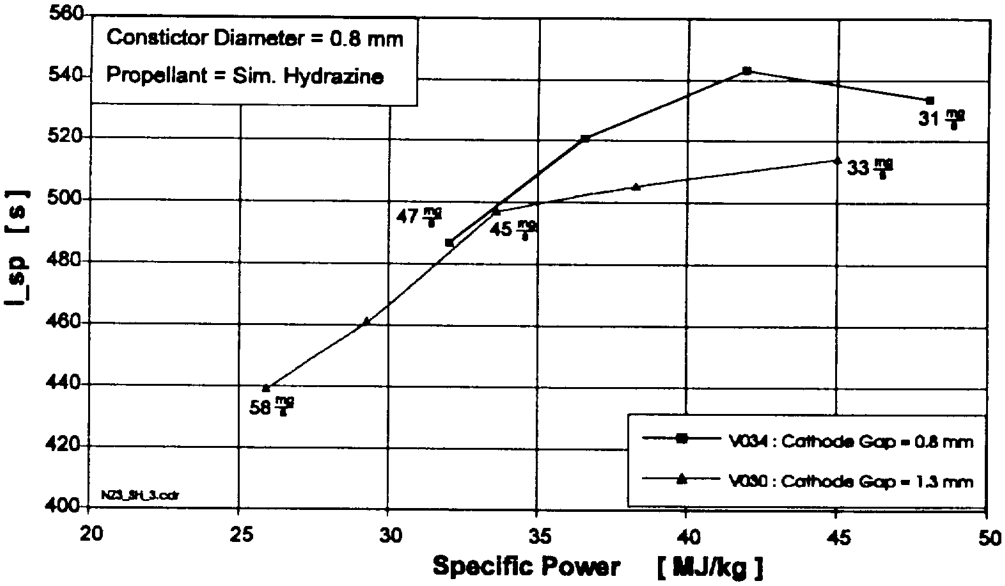

Fig 1.7 : Nozzle 3 : Influence of Cathode Gap at 1.5 kW (balance controlled).

The behaviour of the overall thrust efficency, as depicted in Fig. 1.8, shows some similarity to the hydrogen results plotted in Fig. 1.5. The efficency sinks slowly with increasing specific impulse and decreases rapidly after a sudden break at a mass flow rate of about 45 mg/s. For the 0.8 mm gap, the curve turns back after a maximum at a higher absolute level for the specific impulse. The turn in specific impulse can also be seen in Fig. 1.7.

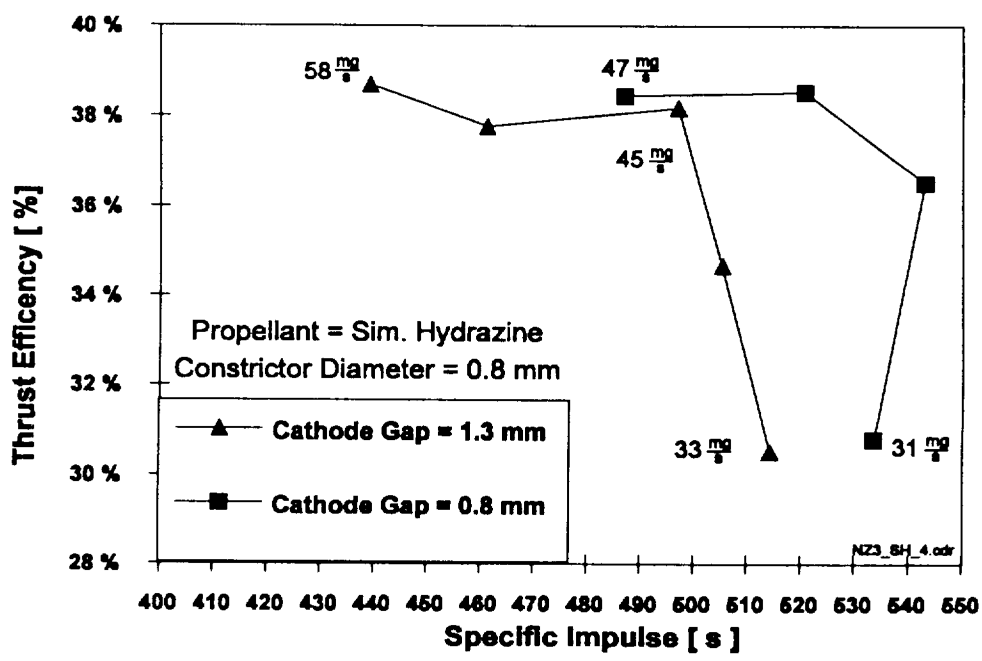

Fig. 1.8 : Thrust Efficency versus Specific Impulse at 1.5 kW.

Some results of the tests already repeated and with $\dot{m}$m determined by the balance are shown in table 1.5. As mentioned above the operation points were selected mainly at the 1.5 kW power line by decreasing the mass flow rate.

Mass Flow	Current	Voltage	el. Power	Thrust	Feed Pressure	Anode Temp.	Isp	Thrust Efficiency	Specific Power
[mg/s]	[A]	[V]	[kW]	[N]	[bar]	[°C]	[s]	[%]	[MJ/kg]
Nozzle 3 Cathode Gap = 1.3 mm **mass flow checked with weight balance**									
58,1	12,6	119,4	**1,50**	0,260	4,02	-	**439**	38,7%	**26**
51,9	13,3	114,2	**1,52**	0,244	3,85	-	**461**	37,8%	**29**
45,0	13,8	109,6	**1,51**	0,228	3,56	-	**497**	38,2%	**34**
39,6	14,6	103,8	**1,52**	0,204	3,27	-	**505**	34,7%	**38**
33,0	15,8	94,0	**1,49**	0,173	2,98	-	**514**	30,5%	**45**
45,0	10,0	118,0	**1,18**	0,213	3,28	-	**464**	42,7%	**26**
45,6	12,0	112,8	**1,35**	0,222	3,41	-	**477**	39,9%	**30**
45,1	13,7	110,3	**1,51**	0,229	3,54	-	**498**	38,5%	**34**
45,0	16,0	107,0	**1,71**	0,238	3,72	-	**519**	36,8%	**38**
Nozzle 3 Cathode Gap = 0.8 mm **mass flow checked with weight balance**									
47,3	14,6	103,9	**1,52**	0,235	3,55	1208	**487**	**38,4%**	**32**
41,1	15,2	99,1	**1,50**	0,218	3,33	1267	**521**	**38,5%**	**37**
36,1	15,9	95,6	**1,52**	0,200	3,11	1315	**543**	**36,5%**	**42**
31,3	18,2	82,5	**1,50**	0,170	2,86	1365	**534**	**30,8%**	**48**
46,5	12,0	107,8	**1,29**	0,222	3,38	1125	**468**	**40,9%**	**28**
41,3	12,0	104,9	**1,26**	0,205	3,09	1141	**487**	**40,5%**	**31**
36,0	12,0	100,4	**1,21**	0,186	2,85	1180	**507**	**39,9%**	**33**
26,3	12,0	88,4	**1,06**	0,140	2,27	1220	**522**	**35,1%**	**40**
31,4	12,0	93,7	**1,12**	0,161	2,57	1205	**504**	**36,8%**	**36**
31,4	15,0	87,0	**1,31**	0,165	2,74	1296	**516**	**33,3%**	**42**
26,3	15,0	81,7	**1,23**	0,150	2,32	1260	**560**	**34,9%**	**47**

Table 1.5 : Operational results with simulated hydrazine and checked mass flow rates.

1.4.3. Ammonia as Propellant

As mentioned above the experiments with ammonia conducted before the weight balance was available are not presented here.

The thruster was ignited also at a preset mass flow rate of 45 mg/s and a current adjustment of about 12 A. The test procedure was the same as described above for the N_2/H_2 mixture.

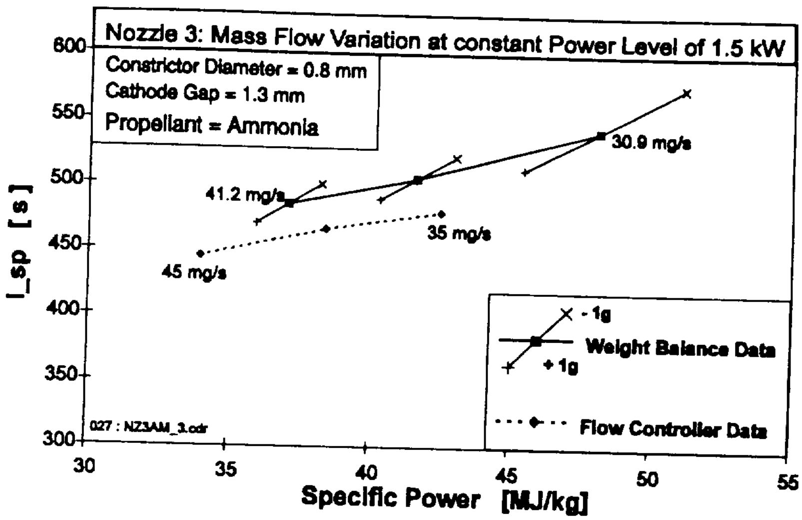

Fig 1.9 : Deviation of Mass Flow Controller.

Introducing the mass flow check with the weight balance, it was detected that the deviation of the used flow controller produced an error as depicted in Fig 1.9. The values of both measured mass flow rates are compared for a typical test run. Some test runs were carried out to principally detect the offset of the used flow controller. This offset was found to be about 5-6 %, which causes the data recorded earlier to be considerable too low, as shown in Fig 1.9. The error bars marked at the points determined with the weight balance give an idea of the maximum deviations of about ±1.5 % for each mass flow setting which could have been reduced to less than 1 % in real tests runs.

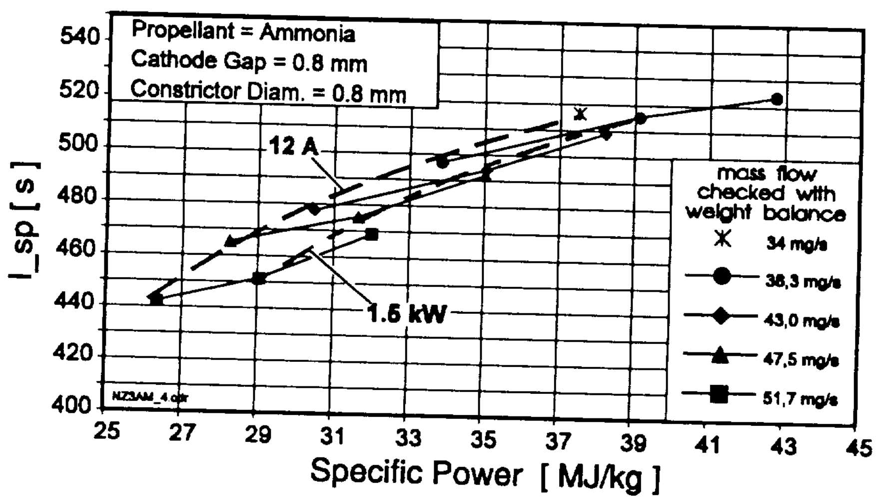

Fig. 1.10 : Nozzle 3 : Specific Impulse versus Specific Power .

The results of one of the first tests repeated with a cathode gap of 0.8 mm are depicted in Fig. 1.10. The mass flow rate was lowered stepwise and three test points were recorded for each setting, one at 12 A, one at the 1.5 kW point and one above this power level (see table 1.6). All of the corresponding 12 A and the 1.5 kW points are connected by a least square fitted function.

The anode temperatures in this test series were not extremely high and a further increase in specific values could be expected, but due to thermal overload of the power supply it was not possible to achieve higher specific power or lower mass flows with this thruster configuration.
Moreover, the thruster worked better with shorter cathode gap before the thruster operation stopped or changed in low mode when approaching the operating limits. In addition the lower voltage encouraged the operation of the DC power supply at lower mass flow rates.

Some results of the tests already repeated are listed in table 1.6. As remarked before, the operation points were selected with main emphasis on the 1.5 kW power setting.

Mass Flow	Current	Voltage	el. Power	Thrust	Feed Pressure	Anode Temp.	specific Impulse	Thrust Efficiency	specific Power
[mg/s]	[A]	[V]	[kW]	[N]	[bar]	[°C]	[s]	[%]	[MJ/kg]
Nozzle 3 Cathode Gap = 0.8 mm									
51,7	12,0	113,5	**1,36**	0,233	3,32	992	**442**	38,6%	**26**
51,7	13,4	112,1	**1,50**	0,238	3,49	1050	**451**	36,5%	**29**

51,7	15,0	110,3	**1,65**	0,247	3,65	1094	**469**	35,7%	**32**
47,7	12,0	112,2	**1,35**	0,226	3,21	1021	**465**	39,8%	**28**
47,5	13,8	108,9	**1,50**	0,230	3,37	1070	**475**	37,1%	**32**
47,4	15,5	107,2	**1,66**	0,238	3,54	1129	**492**	35,9%	**35**
43,1	12,0	109,6	**1,32**	0,210	3,07	1040	**478**	38,9%	**30**
43,1	14,3	105,6	**1,51**	0,217	3,25	1116	**494**	36,2%	**35**
43,0	16,0	102,7	**1,64**	0,223	3,37	1181	**509**	35,2%	**38**
38,1	12,0	107,5	**1,29**	0,193	2,92	1083	**497**	37,9%	**34**
38,3	14,6	102,5	**1,50**	0,201	3,09	1177	**515**	35,3%	**39**
38,4	16,0	102,6	**1,64**	0,205	3,09	1177	**524**	33,3%	**43**
34,0	12,0	106,3	**1,28**	0,179	2,67	1119	**516**	36,9%	**38**
Nozzle 3 Cathode Gap = 1.3 mm									
40,1	11,9	117,9	**1,40**	0,225	3,04	-	**551**	45,0%	**35**
29,2	14,8	103,0	**1,52**	0,185	2,9	-	**621**	38,4%	**52**
42,8	11,9	120,0	**1,43**	0,225	3,23	-	**516**	41,4%	**33**
36,8	13,2	115,7	**1,53**	0,205	3,16	-	**547**	37,4%	**42**
28,0	14,7	101,2	**1,49**	0,160	2,9	-	**561**	30,8%	**53**
41,3	12,9	118,5	**1,53**	0,204	3,26	1215	**485**	33,0%	**37**
36,9	13,5	114,2	**1,54**	0,190	3,11	1279	**505**	31,8%	**42**
30,9	14,1	105,5	**1,49**	0,171	2,92	1355	**543**	31,8%	**48**

Table 1.6 : Operational results with ammonia and checked mass flow rates.

The repetition of the tests which have to be conducted again is currently in progress and the remainig test runs to investigate the performance range of the thruster will also be continued.

2. 5 kW Power Level

2.1. Design Description

For the tests on the 5 kW level a modified version of the IRS MARC thruster *[3]* is used, which itself is a modified and scaled up version of the IRS Artus thruster with different nozzle inserts.
A schematic sketch of this radiation cooled thruster is shown in Fig. 2.1.

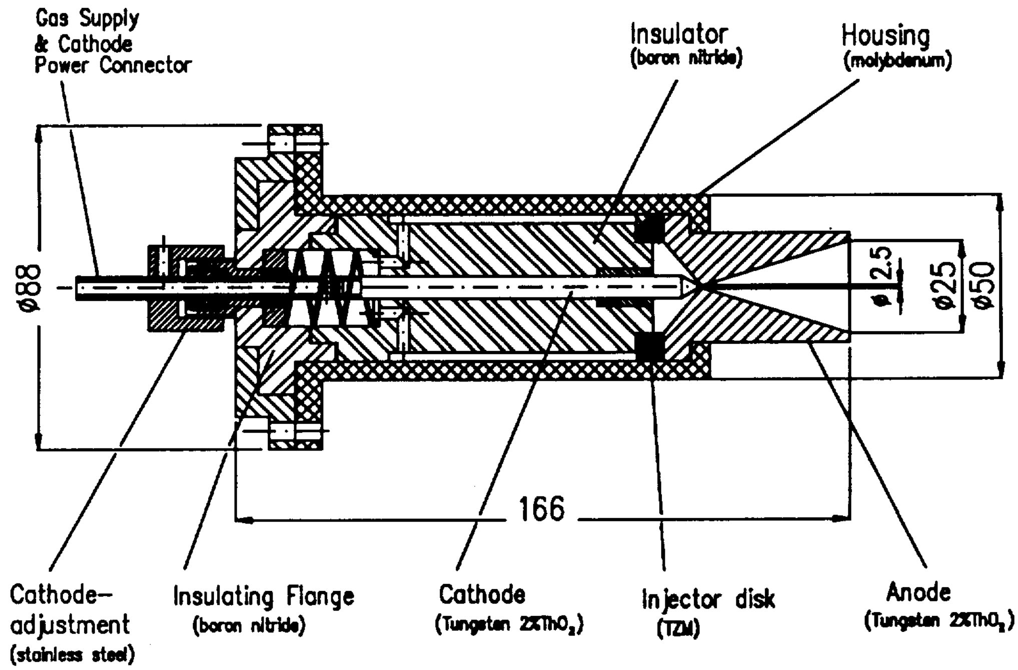

Fig. 2.1 : Assembly of MARC Thruster.

The design concept allows an easy exchange of the critical components and the investigation of other nozzle geometries by changing only one single part. The radiation cooled anode is machined from 2 % thoriated tungsten and is located in a molybdenum housing with a boron nitride insulator between the cathode and the housing. The cathode is machined from 2 % thoriated tungsten, has a 30 ° half angle and is 6 mm in diameter. It is press fitted into a stainless steel cathode feed tube. The propellant is fed in at the rear of the thruster through the cathode feed tube, cooling the rear side of the cathode and the spiral spring. This tube measures 6 mm in diameter with a 3 mm bore hole. The propellant is fed into a ring channel between the housing and the boron nitride insulator, and into the injector ring, cooling the housing and the boron nitride insulator. The propellant is finally injected tangentially into the plenum chamber through four holes with 0.5 mm in diameter.

Three different nozzle types were investigated with all three propellants. Table 2.1 shows the geometry of the inserts.

MARC	Constrictor Diameter [mm]	Constrictor length [mm]	Area Ratio	nozzle end [mm]
Nozzle 1	1,5	1.7	277	25
Nozzle 2	2	2.8	156	25
Nozzle 3	2.5	5	100	25

Tab 2.1 Nozzle Geometries for 5 kW Thruster.

2.2. The Test Facility

The thruster is mounted on the IRS thrust balance described elsewhere *[5]*. This balance is designed for a water-cooled 1-2 N thruster and therefore not optimized for the test with radiation cooled thrusters at a thrust level of about 0.2-0.6 N. Therefore, a pendulum type thrust balance with higher accuracy will be developed and mounted in a vacuum chamber with 2 m diameter which will allow a better access to the internal equipment.

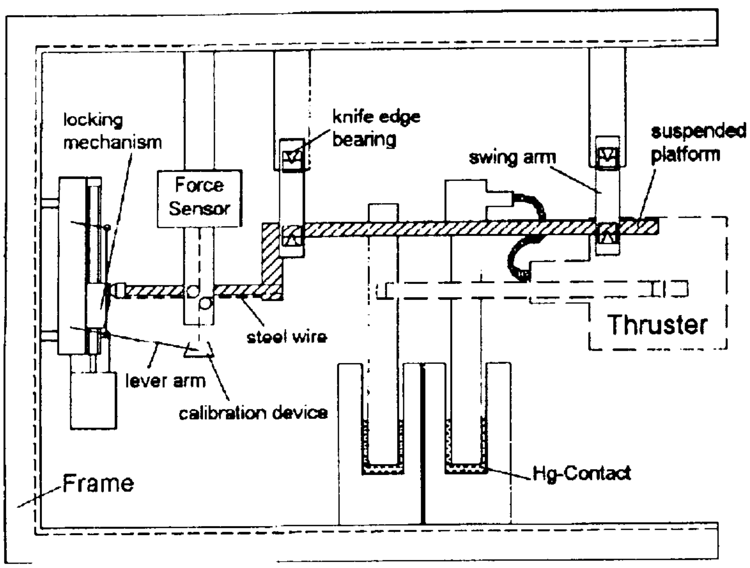

Fig. 2.2 : Scheme of Thrust Balance for 5 kW Thruster.

The balance which is available for the moment, is mounted in a vacuum chamber with a diameter of 1.25 m and a length of 4 m. Fig 2.2 shows the scheme of the complete balance unit.

The vacuum system, the power supply and the gas supply system available at the IRS and the starting procedure are described in *[4]* in more detail. It is possible to measure the voltage, thrust, current, mass flow rates and the line feed pressure of the propellant close to the thruster under computer control.

For temperature measurements on the hot anode the same movable linear pyrometer is used as described in chapter 1.3. The propellant mass flow rates were measured and controlled by a separate set of commercial mass flow controllers (MFCs) for higher measuring ranges. The accuracy of these controllers was also checked at different test conditions by determining the mass decrease of the gas bottle with the described weight balance. Deviations of about 10 to 15% were discovered for the used controller for hydrogen and the N_2/H_2 mixture and of only 1 to 3% for the ammonia controller. A possible recalibration of all flow controllers to fix the deviations will lead to a reduction but not the removal of these errors due to the wide range of the line feed pressure which affects the MFC signal. Therefore, the propellant mass flow rate has been corrected with the balance in each test.

2.3. Experimental Results

2.3.1. Hydrogen as Propellant

First tests were conducted with the nozzle No 3. The thruster was ignited at 100 mg/s and the current was regulated for an input power of 5 kW. The 5.0 kW power level could not be adjusted very accurately for every mass flow variation, because the power supply had to be operated at the low power end of its operation range.

As with the 1.5 kW tests, the mass flow rates were constantly determined with the weight balance. The first results confirmed the necessity of this procedure. Due to the higher mass flow rates for this thruster there was not the same problem with time consumption for each test point.

After the thruster was ignited the mass flow was lowered to 50 mg/s and at this mass flow rate the operation characteristics at different power levels from 3.5 kW up to 7.5 kW were recorded to investigate the thruster's behaviour and the response time to different thermal

loads. After that the power level of 5 kW was set again and the mass flow rate was lowered stepwise to 20 mg/s. At the end of this test a zero drift of the thrust balance was determined. The measured values corrected by this thrust offset are shown in Fig. 2.3 for the specific impulse versus the spec. power. This test was repeated to check the reproducibility of the thrust balance. A drift in the opposite direction was determined and after correcting the data by this new offset, the results were equal to the previous ones.
Therefore, test runs with this facility will be reduced to the number of runs which are necessary to investigate the principal behaviour of the thruster operating with the other propellants. Meanwhile the construction of the new thrust balance is speeded up.

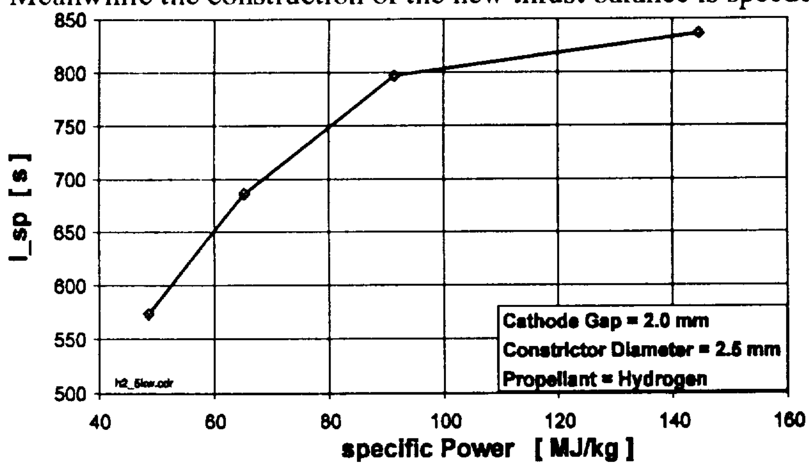

Fig. 2.3: Isp versus Specific Power for Hydrogen at 5 kW.

Using hydrogen as propellant the specific impulse couldn't reach higher values than 850 s and the specific power level remained below 150 MJ/kg. The nozzle throat diameter of 2.5 mm and the nozzle area is too large for a 5 kW thruster in comparison to the results and geometric proportions of the smaller thruster.

Table 2.2 shows the checked and corrected results of the test runs using hydrogen.

Mass Flow	Current	Voltage	el. Power	Thrust	Feed Pressure	Anode Temp.	specific Impulse	Thrust Efficiency	specific Power
[mg/s]	[A]	[V]	[kW]	[N]	[bar]	[°C]	[s]	[%]	[MJ/kg]
Propellant = Hydrogen Constrictor Diameter = 2.5 mm ; Cathode Gap = 2.0 mm ; V004/V005 **mass flow checked with weight balance**									
110	41	127	**5.158**	0.550	1.984		**509**	26.6	**46.9**
85	48	111	**5.345**	0.498	1.700	925	**600**	27.4	**63.3**
60	55	96	**5.325**	0.399	1.404	1160	**678**	24.9	**88.8**
37	76	68	**5.198**	0.252	1.012	1500	**701**	16.7	**141.8**
26	78	61	**4.777**	0.206	.788	1530	**817**	17.2	**186.4**
59	53	93	**4.899**	0.468	1.517	1222	**809**	37.9	**83.1**
111	44	123	**5.405**	0.625	1.966	720	**574**	32.5	**49**
85	51	108	**5.546**	0.572	1.739	930	**686**	34.7	**65**
58	57	92	**5.256**	0.450	1.381	1190	**797**	33.4	**91**
35	93	55	**5.058**	0.287	.978	1300	**836**	23.3	**145**

Table 2.2 : Operational results with hydrogen as propellant at 5 kW.

2.3.2. Simulated Hydrazine and Ammonia as Propellant

As mentioned in the previous chapter only a few test runs using ammonia or simulated hydrazine as propellant were conducted until now. The checked and corrected results are depicted in Fig. 2.4. for both propellants. The obtained low data are also attributedto the large constrictor diameter. The temperatures measured on the anode were comparatively high,

especially for the test runs using ammonia as propellant. This indicates that there was only a slight cooling effect by the considerable high mass flow.
Due to the small data base, no further statements concerning the to thruster performance can be made up to the current state of research.

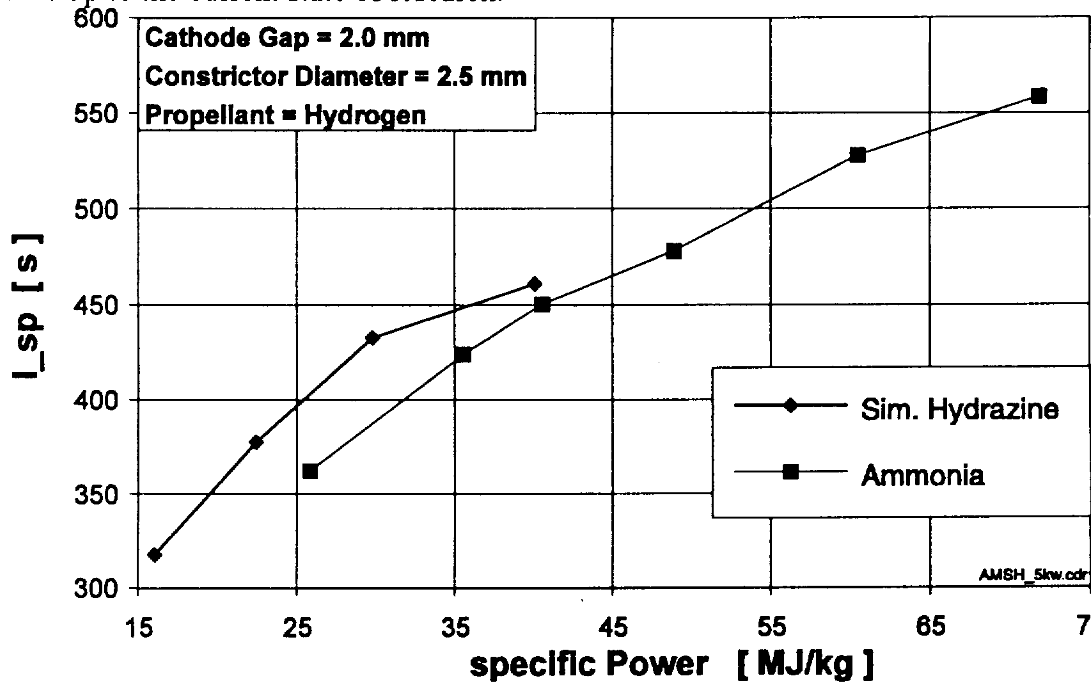

Fig. 2.4: Isp versus Specific Power for Ammonia and Sim. Hydrazine at 5 kW.

Table 2.3 shows some of the test runs. The presented data are checked and corrected for thrust balance drift and flow controller offset.

Mass Flow	Current	Voltage	el. Power	Thrust	Feed Pressure	Anode Temp.	specific Impulse	Thrust Efficiency	specific Power
[mg/s]	[A]	[V]	[kW]	[N]	[bar]	[°C]	[s]	[%]	[MJ/kg]
Propellant = Simulated Hydrazine Constrictor Diameter = 2.5 mm ; Cathode Gap = 2.0 mm ; V003 **mass flow checked with weight balance**									
329	47.	113	**5.289**	1.025	2.622	975	**317**	30.2%	**16**
232	52	99	**5.210**	0.859	2.162	1225	**377**	30.5%	**22**
171	56	91	**5.106**	0.727	1.847	1410	**433**	30.2%	**30**
127	64	80	**5.111**	0.576	1.456	1594	**461**	25.5%	**40**
301	46	113	**5.193**	1.082	3.075	1235	**367**	37.5%	**17**
281	46	109	**5.039**	1.004	2.765	1230	**364**	35.6%	**18**
Propellant = Ammonia Constrictor Diameter = 2.5 mm ; Cathode Gap = 2.0 mm ; V006 **mass flow checked with weight balance**									
200	52	100	**5.162**	0.711	1.386		**362**	24.4	**25.8**
153	59	92	**5.431**	0.635	1.344	1290	**424**	24.3	**35.6**
129	60	87	**5.245**	0.571	1.271	1370	**450**	24.0	**40.6**
104	63	80	**5.069**	0.486	1.126	1450	**478**	22.5	**48.9**
85	71	72	**5.160**	0.442	1.015	1508	**528**	22.2	**60.5**
74	78	69	**5.334**	0.407	.946	1552	**558**	20.9	**71.9**

Table 2.3 : Operational results with simulated hydrazine and ammonia at 5 kW.

2.4. First Design of a Regeneratively Cooled Nozzle Insert.

A future step in the development of thermal arcjet will be the change of the cooling concept from a radiation cooled to a regeneratively cooled thruster. The rear thruster parts are already cooled by the propellant moving downstream, so the redesignis concentrated on the anode nozzle. It promises the largest gain in specific impulse and the best preheating of the propellant during the heat exchange near the region of the arc attachment.

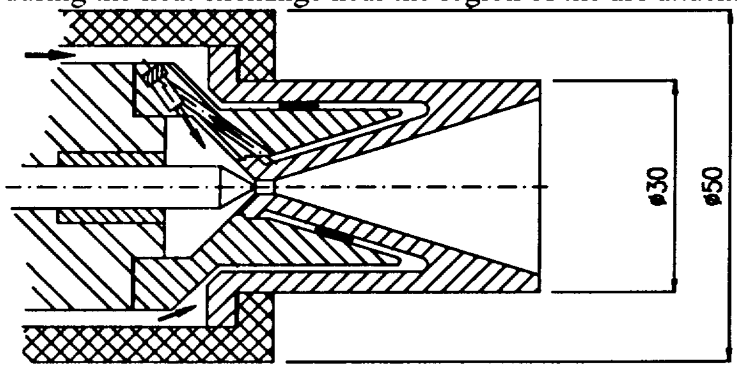

Fig. 2.5 : Scheme of the regeneratively cooled nozzle insert

The design of the nozzle section presented in Fig. 2.5 combines the function of the anode and the injector disk with the possibility of guiding the propellant along the hottest region of the thruster. The chosen geometry will be integrated into the unchanged original thruster and will allow the same easy assembly of the flange type insert.

There will be no additional problems with nozzle leakage between the contact planes of the inner nozzle parts since any leaking propellant will enter the plenum chamber of the thruster. So no propellant will be lost. The two modified parts will be manufactured from thoriated tungsten by a combination of standard milling and shaping by an electro-erosive method for the more complex contours.
The main reason for preferring this simple design is the possibility to detect directly the effect of regenerative cooling by comparing the results of both nozzle versions.

3. Conclusions

For the development, performance investigations and optimization of 1.5 - 5 kW thermal arcjets, two radiatively cooled thrusters were designed and constructed. The corresponding test facilities were modified and first test runs were conducted. The results of preliminary experiments, carried out with both thrusters and all propellants, showed the necessity to check the mass flow rates, obtained with the standard type mass flow controllers, by determining this parameter with a weight balance. In order to obtain a reliable data base some of the 1.5 kW tests will be repeated in addition to the continued experiments.
Besides, some drift phenomena occurred in the first tests with the 5 kW thruster due to the applied principle for the thrust balance which was optimized for water cooled engines. In fact, it is possible to correct the appearing deviations, but as mentioned the experience with the pendulum type balances showed their superiority in accuracy, sensitivity and reproducibility with less influence of drift mechanisms. Therefore the 5 kW facility will be re-set to another thrust balance before the main tests series are continued.
As the data available up to now is inadequate, only a few experimental results can be presented in this report. More tests are required before any clear and accurate conclusions concerning the ability and performance of the thruster can be drawn.

References

[1] Zube, D.M. et al, "Development of a Low Power Radiatively Cooled Thermal Arcjet Thruster"; AIAA IEPC-91-042, Viareggio Italy, 1991.

[2] Curran, F.M. et al. ; "The Effect of Electrode Configuration on Arcjet Performance"; AIAA 89-2722 1989, (NASA TM-102346).

[3] Glogowski, M. Glocker, B. ; "Experimental Investigation of Radiative and Regenera tively Cooled Low Power Arcjet Thrusters"; AIAA 90-2575 IEPC Orlando FL, 1990.

[4] Glocker, B. et al. ; "Medium Power Arcjet Development"; AIAA 90-2531, 21st IEPC, Orlandeo FL, 1990.

[5] Glocker, B. et al. ; "Medium Power Arcjet Analysis and Experiments"; IEPC-91-016, Viareggio Italy, 1991.