DOI: 10.13009/EUCASS2017-154

Green Gelled Propellant Throtteable Rocket Motors for Affordable and Safe Micro-Launchers

K.W. Naumann, H.K. Ciezki**, P. Caldas Pinto*, J. Ramsel*, T. Meyer*, S. Rest*, A. Thumann*, G. Kurth**

** Bayern-Chemie GmbH*

84454, Liebigstraße 17, Aschau am Inn, Germany

*** DLR Institute for Space Propulsion*

Langer Grund, 74239 Hardthausen, Germany

Abstract

The Paper describes the concept of the main propulsion system concept for an affordable, safe and environmental friendly Micro-Launcher and a roadmap for the realization. The enabling element for this Micro-Launcher propulsion concept is the Green Gelled Propellant Rocket Motor. In addition to low hazard potential and easy handling operations the Gelled Propellant Rocket Motor shows very stable combustion and very fast thrust control properties. Based on designs for sounding rocket stages, the paper details on the design options and trade-offs and outlines a basic propulsion concept for a three-stage Micro-Launcher.

Nomenclature

Parameters

A	[m^2]	Area
D	[m]	Diameter
F	[N]	Force, Thrust
g	[9.81 m/s^2]	Earth gravity constant
I_{sp}	[m/s]	Specific impulse
L	[m]	Length of a body
m	[kg]	Mass
$\dot{m}$	[kg/s]	Mass flow
Ma	[1]	Mach number
p	[Pa]	Pressure
R	[m]	Radius
s	[m]	Wall thickness
T	[K]	Temperature
t	[s]	Time
V	[m^3]	Volume
W	[J]	Work
η	[1]	Coefficient of efficiency
ρ	[kg/m^3]	Density

Superscripts

*	Value at nozzle throat, sonic condition

Subscipts

b	Burn operation
CC	Combustion chamber
c	Combustion
e	Condition at nozzle exit or at end of acceleration
GRP	Gelled rocket propellant
i, j	Index
Prop	Propellant
St	Stage
T	Tank

DOI: 10.13009/EUCASS2017-154

vac	Vacuum conditions
0	Reference condition
∞	Ambient condition

Abbreviations

BC	Bayern-Chemie
CC	Combustion chamber
CFRR	Carbon fibre reinforced resin
CMC	Ceramic matrix composites
COTS	Commercial of the shelf products
DACS	Divert and attitude control system
GGG	Gelled propellant gas generator
GGPT	German gel propellant technology
G-SoRo	Sounding Rocket with GRM
G-μL	Micro-Launcher with GRM
GP	Gelled propellant
GRM	Gelled propellant rocket motor
GRP	Gelled rocket propellant
HCCC	Highly controllable combustion chamber
IF	Interface
LEO	Low earth orbit
LRM	Liquid propellant rocket motor
MECO	Main engine cut-off
MMH	Mono methyl hydrazine
MSL	Main sea level altitude
NTO	N_2O_4
REACh	Registration, evaluation, authorizationof chemicals
SRM	Solid rocket motor
SSO	Sun synchrous orbit
STj	Stage “j”
TVC	Thrust vector control

1. Introduction

The initial idea that spurred the development of Gelled propellant Rocket Motor (GRM) technology in Germany was to create a controllable rocket motor without the hazard potential of liquid propellants.
Controllable solid rocket motors were ruled out because a mechanism that controls the throat cross section area needs a lot of energy, following the law $W = p\mathrm{d}V$, and operates at the most adverse combination of gas state parameters p^*, ρ^*, T^*. In addition, interfaces and sealing of extremely hot structures in a transient heating scenario are challenging thermo-mechanical problems.
Hybrid rocket motors were ruled out because of the large volume needed, the low thrust density, caused by low regression rate of the propellant and the fact that the combustion process comprises many interacting physical phenomena (boundary layer, heat transfer, de-composition or melting of solid propellant, evaporation, mixing, turbulence, combustion, etc.) that vary along the combustor length and cannot be controlled directly, but just indirectly by the injection method.

Essentially, a GRM combines the advantages of a Solid propellant Rocket Motor (SRM) and a Liquid propellant Rocket Motor (LRM). Overviews on GRM technology are given in [1, 2]. The state particularly of the German Gel Propulsion Technology (GGPT) activities on GRM and Gelled propellant Gas Generator (GGG) technology are reported in [3, 4, 5] and the perfect free-flight demonstration in [6], see Fig. 1. Aspects of hazard potential and environmental impact of GRM and other rocket motor technologies are discussed in [7, 8]. Hence, in this paper we just briefly outline specific properties of GRM technology that are needed to understand this paper. For scientific aspects and verification data the reader is asked to study the references and the literature cited therein.

Fig. 2 shows the principle of operation of a GRM. The Gelled Propellant (GP) is solid in the tank and has to be fed into the combustion chamber by pressure. The propellant mass flow rate is controlled by a valve in the feeding line. Upon injection into the combustion Chamber (CC) the gel structure of the GP is destroyed and the GP liquefies. The spray burns in the CC almost like a liquid propellant. Whereas monopropellants represent the current state of GRM

DOI: 10.13009/EUCASS2017-154

technology, bi-propellants and hypergolic propellants are under investigation.

Figure 1: BC´s GRM demonstrator missile with smoke-free GRP 001 just after launch [6].
Fig. 2 shows a sketch of the principle of operation of a GRM.

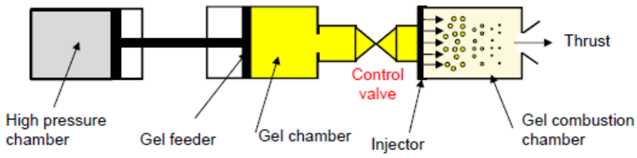

Figure 2: Principle of operation of a GRM [9]

The Gelled Rocket Propellants (GRP) consists of a blend of fluids, gelling agents, additives and if useful and tolerable, solid particles. This increases the density and the specific impulse and can be done because the nature of the GRP prevents sedimentation or buoyancy of incorporated particles even over long storage times. Many monopropellant formulations for various applications have been tested [10]. An essential guideline of the GGPT program was to develop GP that have a minimum hazard potential to no-protected persons in case of an accident, a malfunction or some other kind of mishap. This has been achieved with respect to the monopropellant GP itself as for the combustion products [7, 8]. The GP spills not out of leakages, if set free creates no leaches with large evaporation surface, has a low evaporation rate and thus does not create “fireballs”. It also does not soak into the soil or flow into sewers.
The pressurization of the GP tank can be done by compressed gas or solid gas generators if space is limited, or a combination of both. For GP tanks with high L/D and small D a piston is a good solution to separate GP and gas. Tanks with low L/D can separate GP and gas by a membrane or a bladder; the latter solution is also suitable for tanks with large diameter. Helpful in many respects is that GP shows no sloshing in the tank.
The CC is designed using materials and methods that are common in the design of exposed surfaces in SRM. The uncooled CC walls need either an ablative heat shield, or may be made of Ceramic Matrix Composites (CMC) if T_c is not too high. This design method is comparatively easy and does not need very special materials. For short times of operation, the nozzle throat can be made conventionally from graphite. If a higher degree of resistance against erosion is required, CMC materials show already good performance in SRM.
Tests showed very good scalability of GRM in the range from 0.3 – 20 kN nominal thrust and there is no indication that further scale-up should create difficulties [11]. We also observe very stable combustion and very fast and stable thrust control characteristics. Fig. 3 shows unfiltered pressure curves of a test with GRP 006 that has a very wide combustion pressure range from 0.6 to more than 10.5 MPa. For tests with the Highly Controllable Combustion Chamber (HCCC) with variable injector head, variable nozzle and appropriate control algorithm see reference [12].

A GRM is inherently safe because three independent actions are needed to set a GRM into operation. At first, the tank is pressurized. The second step is to initiate the igniter. When p_{CC} exceeds a given threshold value, the GRP valve opens and the GRM starts to operate. Hence, no ignition and safety device, essential for SRM, is needed.

By now, GRM and GGG technology have reached a maturity that allows to enter into demonstrators and developments for specific applications. Promising applications of the German GRM / GGG technology are expected where the following requirements drive the design:

- Low hazard potential, good insensitivity, strict safety requirements

DOI: 10.13009/EUCASS2017-154

- Affordable propellant, easy handling operations including launch, uncritical infrastructure and launch operations
- REACh-compatible propellant (over the entire life cycle) that produces environmentally friendly exhaust gas
- Particle-free exhaust gas and operation if required
- Long operation times due to the separation of tank and CC
- No sloshing of propellant in the tank
- A limited number of operation cycles

Some ideas about design trade-offs for the various applications and some exemplary generic concepts of space applications gives [13]. Concepts for impulse control systems like Divert and Attitude Control Systems (DACS) outline [14, 15].

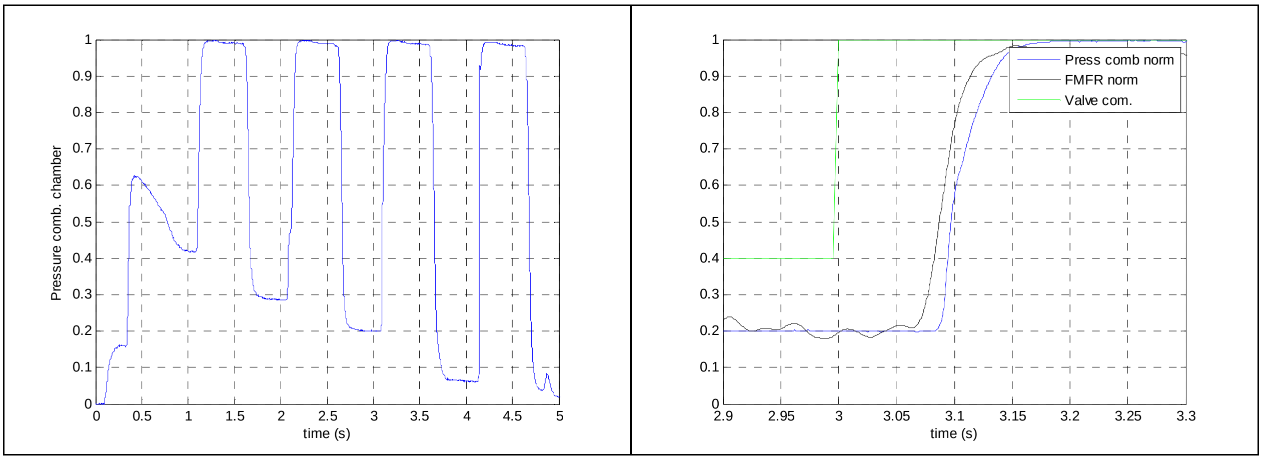

Figure 3: Normalized p_c over time for a test with 4 thrust level steps [12]. Left: complete test cycle, right: time-zoom of the sequence from t = 2.9 – 3.3 seconds. Yellow: GRP valve command, red: propellant mass flow, blue: combustion pressure.

2. Concept of a Safe, Affordable and Environmentally Friendly Micro-Launcher

The following considerations on design trade-offs, performance and functional parameters are based on the results of the efforts described in the references given above.
Stages for sounding rockets are a reasonable first application of a novel propulsion technology because the specific performance requirements in terms specific impulse and propellant mass fraction are mild. This allows a design using good margins and to evaluate from successful flights the degree of “overquality” as a base for further optimization. Generic concepts for stages of sounding rockets were outlined in [16, 13]. Based on more precise requirements, developed to realize performance parameters that match those of the VSB-30 sounding rocket, a design concept for the propulsion system of a GRM Sounding Rocket (G-SoRo) has been set up and described in more detail in [17]. The concept of the propulsion system of the GRM-Micro-Launcher (G-µL) is a derivative of the G-SoRo propulsion system and uses widely the same design features. Because the G-SoRo propulsion concept is the starting point, it is briefly outlined at this place to facilitate the understanding of the G-µL propulsion concept without the immediate need to look into [17].

2.1 Brief outline on the stages of sounding rockets with GRM

The VSB-30 [18] carries a payload section of about 400 kg mass to an apogee of about 270 km what allows some minutes of microgravity experiments. VSB-30 consists of 2 SRM stages, S-31 as first stage and S-30 as second stage. Table 1 gives some key parameters for the two stages of VSB-30 and for the two modular stages of G-SoRo.

By now, GRM and GGG technology have reached a maturity that allows to enter into demonstrators and developments for specific applications. Promising applications of the German GRM / GGG technology are expected where the following requirements drive the design:

- Low hazard potential, good insensitivity, strict safety requirements

DOI: 10.13009/EUCASS2017-154

- Affordable propellant, easy handling operations including launch, uncritical infrastructure and launch operations
- REACh-compatible propellant (over the entire life cycle) that produces environmentally friendly exhaust gas
- Particle-free exhaust gas and operation if required
- Long operation times due to the separation of tank and CC
- No sloshing of propellant in the tank
- A limited number of operation cycles

Some ideas about design trade-offs for the various applications and some exemplary generic concepts of space applications gives [13]. Concepts for impulse control systems like Divert and Attitude Control Systems (DACS) outline [14, 15].

Table 1: Key parameters of S-31 and S-30 SRM [18] and of the stages of G-SoRo [17].

		S-31	**S-30**	**G-SoRo St 1**	**G-SoRo St 2**
Diameter	[m]	0,56	0,56	0,58	0,58
Propellant mass	[kg]	616	861	1000	1000
Gross mass of SRM/GRM	[kg]	900	1200	1277	1232
Peak thrust	[kN]	240	102	180	100
Burning time	[s]	13.5[a]	29[a]	15	30

[a]According to [19]

The solid propellant of the VSB-30 stages is aluminized as this is common for launchers, and alike the GRP-002 of the G-SoRo contains energetic solid particles. The specific impulse of GRP-002 is somewhat better than that of aluminized solid propellants. The density of GRP-002 is 1300 kg/m^3 which is a good value for liquid propellants but less than the 1800 – 1860 kg/m^3 of highly aluminized solid propellants. Taking also into account that the GRM needs devices for pressurization, GRP mass flow control and has separate tank and CC means that GRM stages need more volume for the same mass of propellant than SRM stages, whereas in the case of the sounding rocket stages the dry mass of the GRM stages is less than that of the SRM.

Figure 4 shows pictures of the 1st and 2nd stage of the G-SoRo.

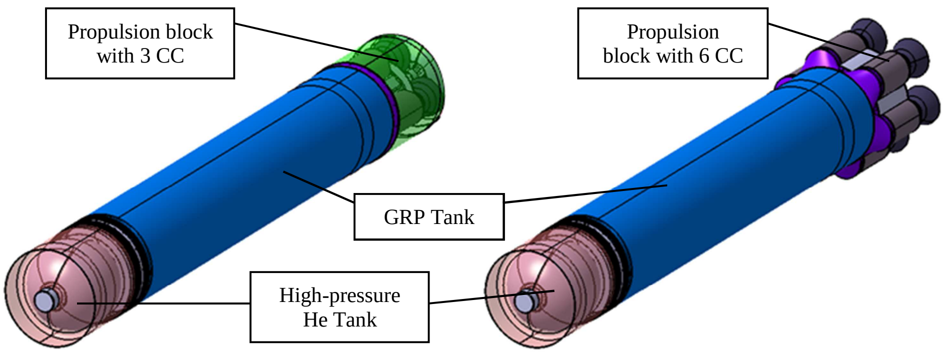

Figure 4: Picture of the modular 1st (right) and 2nd (left) stage of the G-SoRo [17]

The key design features of the propulsion systems for the stages are:

- The tanks act also as the body of the stages and carry the axial, lateral and bending loads
- The high-pressure gas tank is also the forward closure of the GRP tank and carries the front skirt of the stage
- The aft skirt of the tank carries the GRM blocks and makes the interface to the rear skirt

DOI: 10.13009/EUCASS2017-154

- The tanks are made of Carbon Fibre Reinforced Resin (CFRR); the gas tank has a liner of aluminium.
- The GRP is held in a collapsible bladder within the tank
- The GRP tank is pressurized using a pressure reducer. The maximum operating pressure p_T = 10 MPa. After equilibration of the pressure in gas and GRP tank, the pressure decreases with the expulsion of the GRP. In parallel, the combustion pressure has to be reduced to maintain the safe pressure ratio between GRP tank and combustion chamber pressure. This is controlled by the controller and the control algorithm. The corresponding decrease of thrust corresponds to the decreasing mass of the launcher and helps to reduce the peak acceleration
- All GRM CC with injector head and nozzle, in the following designated GRM, are identical, also with respect to the nozzle opening area ratio A_e/A^* = 12. This is sub-optimal for low ambient pressure, but cost-efficient in development and production. The injector head, the shell of the CC and nozzle are made of metal alloy. The CC shell and the nozzle structure are protected by an ablative heat shield. For operating times of 15 and 30 seconds, respectively (see Tab. 1), the thickness and mass of the ablative heat shield is well tolerable. We can use the aft closure and nozzle design methods and materials that are known from SRM.
- The GRM blocks consist of 6 GRM for the first stage and 3 GRM for the second stage which are identical. The bundles of GRM are chosen because:
    - The development effort needed for the development of the GRM is less than that needed for the development of one much bigger GRM
    - The bigger number of identical GRM per launcher allows lower production cost
    - The multiple GRM need significantly less length than a single GRM
    - Each individual GRM is thrust controlled by propellant mass flow rate control. This entails already a thrust vector control. No additional components like cardanic bearings and actuators are needed; just the control algorithms have to be adapted.
    - The exact number of GRM per stage depends on trade-offs that take into account not only the internal optimization, but also boundary conditions, e. g. of the launch infrastructure
- The ancillary components needed are the pressurizing gas feeding assembly, the pressure reducer, the propellant flow control valves with actuators, the propellant loading assembly and the energy supply.

The G-SoRo is more complex than the currently used SRM stages, but allows for cross-wind compensation and tailoring of the trajectory. In addition G-SoRo entails safety of operation, generally low hazard potential and environmental friendliness.

The modular nature of GRM allow to adapt the design and the dimensions widely to boundary conditions and performance requirements. Hence, the given dimensions and parameters show one of many possible versions of GRM systems for sounding rockets.

2.2 The GRM propulsion systems for a three-stage micro-launcher

Earlier presentations on space launcher applications of GRM focussed on highly controllable upper stages and orbit insertion stages with green GRP [9, 14, 13]. The studies on the performance of sounding rocket stages outlined above indicate that GRM could also be a solution for the main stages of a safe and environmentally friendly micro-launcher. In order to get a first rough idea on the potential of a G-µL, a parametric concept has been developed in the same way as this was done with very good results for the G-SoRo stage propulsion systems.

The initial sizing is based on general available information about µL-concepts and market, and uses rules of thumb without sophisticated optimization procedures:

- A satellite of about 200 kg should be placed into an unspecified LEO. This is just a design value because the launch site and inclination of the orbit have a significant impact on the payload capacity of a launcher
- The mass of payload including fairings, interfaces and ancillary components is assumed as 400 kg
- 3 stages for propulsion systems that do not use cryogenic fuel or oxidizer
- Distribution of propellant mass for a 3-stage launch vehicle according to the ratio 9:3:1 for Stages 1 – 3
- The ratio of propellant mass / stage mass m_{Prop}/m_{ST} of the launcher stages is about 0.85
- The thrust at lift-off is $1.5 m \cdot g < F < 2 m \cdot g$
- The GRM uses GRP-002 with energetic particles. The nozzle area ratio chosen is for
    - Stage 1: A_e/A^* = 20, slightly over-expanded at MSL
    - Stage 2: A_e/A^* = 40
    - Stage 3: A_e/A^* = 70

DOI: 10.13009/EUCASS2017-154

Fig. 5 shows the calculated specific impulse over p_c/p_∞ for these three A_e/A^*. Because the gas produced by the combustion of a particle-laden GRP and exiting through the nozzle is neither ideal nor adiabatic, we assume an efficiency $\eta = 0.93$ in contrast to typical values of $0.97 > \eta > 0.95$. The value of $\eta = 0.93$ is used for the performance calculations and are also given in Fig. 5. Even with a deduction of 7 % from the theoretical value, the $I_{sp,GRP\text{-}002}$ at high p_c/p_∞ is very good and matches that of thrusters with the storable bi-propellants MMH/NTO. To establish valid values of the effective $I_{sp,GRP\text{-}002}$, tests are planned at DLR Institute for space Propulsion at Lampoldshausen, Germany.

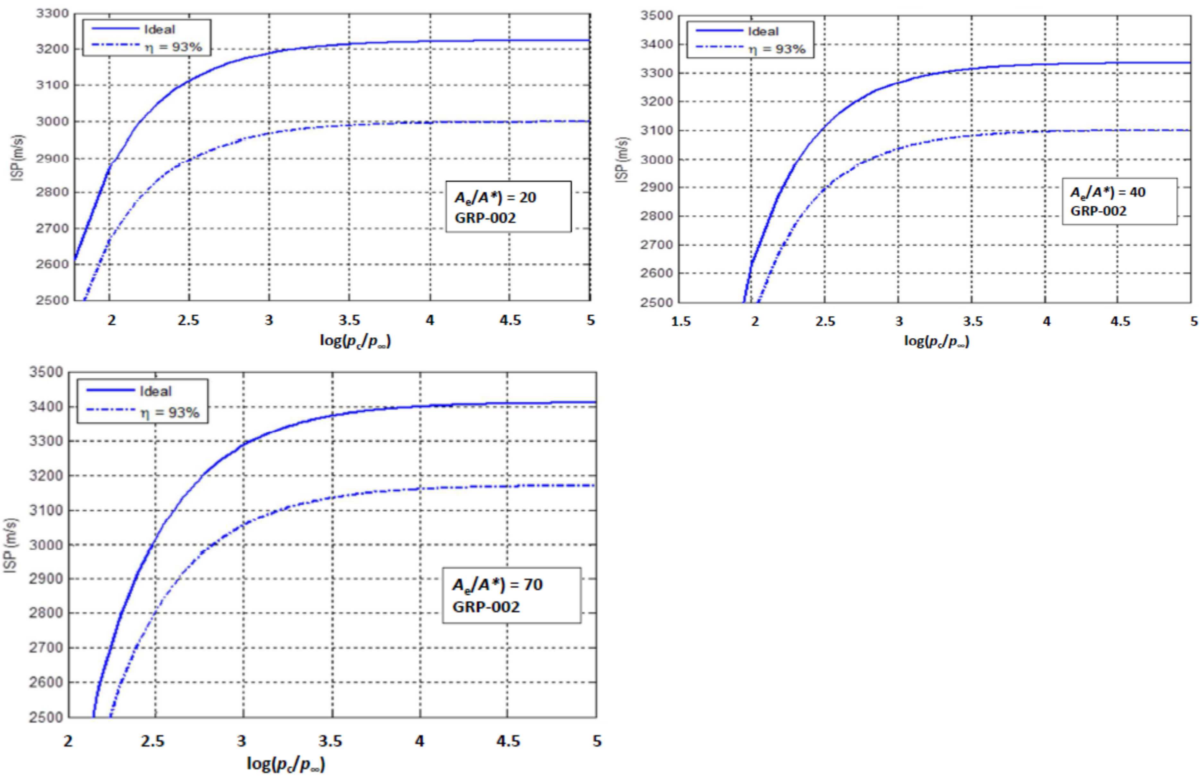

Figure 5: Specific impulse of GRP-002 over p_c/p_∞ for the three A_e/A^* of stages 1 – 3.

Using these above outlined basic relations, boundary conditions and pre-requisites, we have set the following parameters:

- Propellant mass for stages 1 – 3: 21000 kg, 7000 kg, 2300 kg. The expected launch mass is about 35 metric tons
- Diameter of stages 1 – 3: 1.4 m, 1.0 m, 1.0 m. The stage length is a function of the size of the tanks and the GRM block
- Nominal thrust of stages 1 – 3: 600 kN, 90 kN, 30 kN

The modularity of GRM propulsion systems allows to design a G-µL using the design principles of the GRM propulsion system of G-SoRo. Of course the very different size and applications pose technical challenges; in particular:

- Significant scaling up of all parts of the GRM propulsion system compared to G-SoRo, especially for the first stage
- Significantly longer operation times of the GRM for the 2nd and 3rd stage
- Re-ignition in vacuum of the 3rd stage, needed for the circularization burn of an energy-optimized orbital insertion trajectory

Key elements of the design of all stages of the G-µL are:

- GRP tank:
    - Case made of Carbon Fibre Reinforced Resin (CFRR), being also the load-carrying structure of the stage

DOI: 10.13009/EUCASS2017-154

- Collapsible bladder that contains the GRP
- The high-pressure gas tank acts as forward closure of the GRP tank
- Operating pressure: 10 MPa for stage 1, 8 MPa for stages 2 and 3

- High-pressure He tank:
  - Case made of Carbon Fibre Reinforced Resin (CFRR), being also the load-carrying structure of the forward part of the stage, with an internal liner of Al-alloy
  - Spherical shape with skirts, external diameter = stage diameter
  - Filling pressure: for stages 1 – 3: 70 MPa, 60 MPa, 20 MPa respectively; the very different filling pressure levels are caused by the given dimensions of the spherical gas tanks whose external radius is the external radius of the respective stage. A consequence is that the ratio $V_{\mathrm{Tank,GRP}}/V_{\mathrm{Tank,He}}$ changes significantly from 14.4 for stage 1 to 12.7 for stage 2 to 3.6 for stage 3. The impact on the structural mass of the gas tank is small because $p_{\mathrm{He}} \sim 1/V_{\mathrm{Tank,He}} = 1/(R_{\mathrm{Tank,He}})^3$ whereas for the mechanical load-carrying structure of the spherical gas tank $m_{\mathrm{Tank,He}} \sim A_{\mathrm{Tank,He}} \cdot s_{\mathrm{Tank,He}} \sim (R_{\mathrm{Tank,He}})^2 \cdot (p_{\mathrm{He}} \cdot R_{\mathrm{Tank,He}}) \sim (R_{\mathrm{Tank,He}})^3 \cdot p_{\mathrm{He}} \sim V_{\mathrm{Tank,He}} \cdot p_{\mathrm{He}}$ = constant if m_{He} and T_{He} are constant. Parts that do not follow this rule are the internal liner of Al-alloy with a wall thickness that is independent of p_{He} and consequently scales ~ $(R_{\mathrm{Tank,He}})^2$, and the metallic polar bosses. But the contribution of these parts to the mass is small.
- The m_{He} is chosen in that way that the pressure of the GRP tank and the He tank equilibrate when 40 – 50 % of the GRP are used (ST1: 42 %, ST2: 49 %, ST3: 39 %. After this point $p_{\mathrm{Tank,GRP}}$ decreases as the propellant is fed out. The final $p_{\mathrm{Tank,GRP,e}}$ is for ST1: 4.56 MPa, for ST2: 4.39 MPa, for ST3: 4.33 MPa (remember that $V_{\mathrm{Tank,GRP}}/V_{\mathrm{Tank,He}}$ is different for each stage). In consequence, p_c has to be throttled down about proportionally to p_{Tank} to maintain a pressure ratio $p_{\mathrm{Tank,GRP}}/p_c \geq \approx 1.5$. This method reduces the mass of the pressurization system, mainly $m_{\mathrm{Tank,He}}+m_{\mathrm{He}}$. The associated decrease of thrust after the equilibration of gas and propellant tank pressure is welcome to reduce the peak acceleration towards burn-out of the respective stage.
- COTS pressure reducers; 3 in parallel for stage 1, one for stages 2 and 3
- The propellant of all stages is GRP-002
- The design method and materials of all GRM are similar, with the exception of the dimensions and the maximum operating pressure. The metallic shell of CC and nozzle as well as the injector head are made of Ti-alloy; steel can be used if sufficient mass margin should be available. The material for the ablative internal thermal protection is the same as for G-SoRo but with thicker walls because of the longer operation times. The long operation times of the GRM also favour the use of CMC for the nozzle throat section.

The specifics for the propulsion system of each stage are as follows:

- The GRM block of stage 1 consists of 6 GRM with a nominal thrust of 100 kN each, $A_e/A^* = 20$ and a nominal burning time $t_{\mathrm{b,ST1}} = 107$ s. The geometrical arrangement of the GRM block is like that of stage 2 of G-SoRo shown in Fig. 4 because there is sufficient cross section area inside of the stage diameter. The propellant flow rate control system is located in the space between the GRM. The $\dot{m}_{\mathrm{GRP,i}}$ is controlled for each GRM i individually by a valve. This compensates for inevitable un-symmetries of geometrical parameters of the propellant feeding system and of operational parameters of the GRM. A big advantage is that by the control of $\dot{m}_{\mathrm{GRP,i}}$ a thrust vector control is implemented without the need to integrate additional mechanical and electric devices, i. e. gimbals, actuators and energy supply, that move the GRM.
  The GRP tank filling assembly is also located at the rear of the stage between the GRM
- The GRM block of ST2 consists of 3 GRM with a nominal thrust of 30 kN each and $A_e/A^* = 40$ and a nominal burning time $t_{\mathrm{b,ST2}} = 262$ s. The geometrical arrangement of the GRM block is like that of stage 2 of G-SoRo shown in Fig. 4. The benefit is a clean air flow around the interstage between ST1 and ST2. Like at ST1 the propellant flow rate control system is located in the space between the GRM, and $\dot{m}_{\mathrm{GRP,i}}$ is controlled for each GRM i individually by a valve. This also provides a thrust vector control for ST2 with a minimum of mechanical and electrical devices. The GRP tank filling assembly is also located at the rear of the stage between the GRM
- ST3 has just a single GRM with a nominal thrust of 30 kN and a nominal burning time $t_{\mathrm{b,ST3}} = 265$ s. In effect, the thrust of the GRM of ST3 is a little bit higher than that of the ST2 GRM because $A_e/A^* = 70$ whereas the injector head and the CC are identical what is possible because $t_{\mathrm{b,ST3}} \cong t_{\mathrm{b,ST2}}$. For thrust vector control (TVC), the single GRM is gimballed and the orientation controlled by two actuators. The propellant flow rate control system is located between the GRP tank and the GRM
- The roll control can be done by different methods, using inert or combustion gas. The appropriate solution has to be established by trade-offs that take into account the required total impulse, mass, technical complexity, cost and whether each stage is equipped with its own roll control system or whether a roll

DOI: 10.13009/EUCASS2017-154

control system is only integrated inST2 (for the operation of ST1 and ST2) and ST3. For the concept discussed here, the mass budget for the roll control system is considered for each stage and incorporated into the mass budget for the ancillary devices.

2.3 Parametric concept of a three-stage micro-launcher

Whereas the propulsion system, i. e. tanks, GRM and ancillary devices like TVC represent the by far major structural part and inert mass of a stage, stage- and vehicle-specific devices have not to be neglected. Hence, the mass of the vehicle is made up as follows:

- For each stage, a margin of 15 % is added to the sum of structural mass plus mass of pressurizing gas. The resulting total inert mass of the stages is:
  - ST1: 2280 kg
  - ST2: 785 kg
  - ST3: 334 kg
- The mass for the structures of the interfaces IF_{ij} between the stages *i* and *j* is assumed to be 2 % of the total inert mass of the lower stage as given above, i. e. $m_{IFij} = 0.02 m_{STi}$
- The mass budget taken into account for ancillary devices, e. g. for stage separation, energy storage, guidance, navigation, control, emergency destruction, and some propellant residuals is:
  - For ST1 to ST2: 500 kg
  - For ST2 to ST3: 250 kg
  - for ST3 to payload: 125 kg

The mass breakdown and the key dimensions of the G-µL are summarized in Tab. 2.

Table 2: Key parameters of the µL.

		ST1	**ST2**	**St3**	**Payload**
Diameter	[m]	1.4	1	1	1
Total length of STi including IF_{ij},	[m]	14.3	9.7	5.2	-
Mass at ignition of STi	[kg]	35690	11440	3250	-
Mass at burn-out of STi	[kg]	14690	4440	950	-
MECO mass of stage	[kg]	3246	1193	547	-
MECO / total stage mass	[%]	13.5	14.6	19.2	-
$I_{sp,MSL}$ [a]	[m/s]	2400	N/A	N/A	
$I_{sp,vac}$ [a]	[m/s]	3000	3100	3170	
Burning time	[s]	107	261	265	-

[a]See Fig. 5

2.4 Function and performance of the G-µL

The DLR Institute for Space Systems carried out calculations on launch trajectory using the data given in Tab. 2 [20]. Additional assumptions are:

- Launch at Kourou, French Guyana
- A simplified aerodynamics model as shown in Fig. 6, yielding aerodynamic drag coefficient over Ma-

DOI: 10.13009/EUCASS2017-154

number as shown in Fig. 6

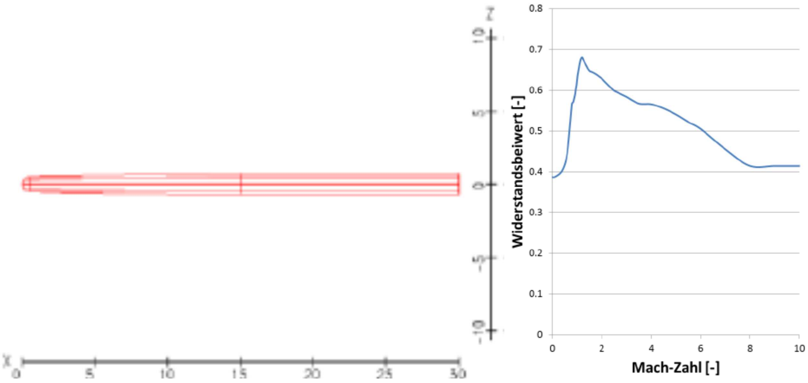

Figure 6: Aerodynamic model for the G-µL and C_D over *Ma* [20].

Fig. 7 shows for a launch trajectory till the transfer orbit 180km × 700 km to a sun-synchrous orbit (SSO) at 700 km altitude:

- Flight altitude over time of flight
- Flight altitude over flight velocity
- Angles of attack, flight path and bank over time of flight
- Thrust, drag and lift (lift is not really visible) over time of flight

The graph of the angles shows that the angle of attack at operation of stages 2 and 3 is about 15 – 20 degrees, whereas the thrust of ST1 allows a gravity turn at zero angle of attack.

The circularization burn is not given in the graphs. The mass of propellant needed for this circularization burn is negligible but respected in the calculations of the payload mass delivered to orbit.

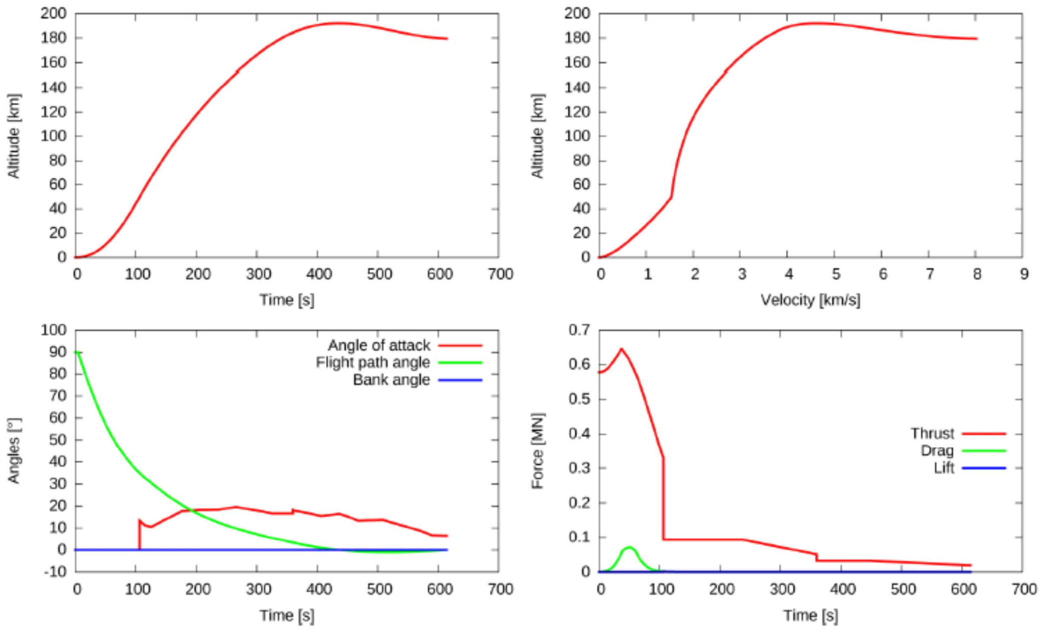

Figure 7: Flight altitude over time of flight and over flight velocity, angles of attack, flight path and bank over time of flight, thrust, drag and lift over time of flight [20].

Fig. 8 shows for the same launch trajectory acceleration, vehicle mass, *Ma* and dynamic pressure over the time of flight. The acceleration during the operation of ST2 is comparatively low. This is not critical, but in the course of an optimization it might eventually turn out that a GRM block with higher thrust is more effective. Notice that the thrust/mass ratio of the GRM is quite high, so the penalty on inert mass may be limited.

The results of the calculations on the mass transferred to LEO and SSO are shown in Table 3.

DOI: 10.13009/EUCASS2017-154

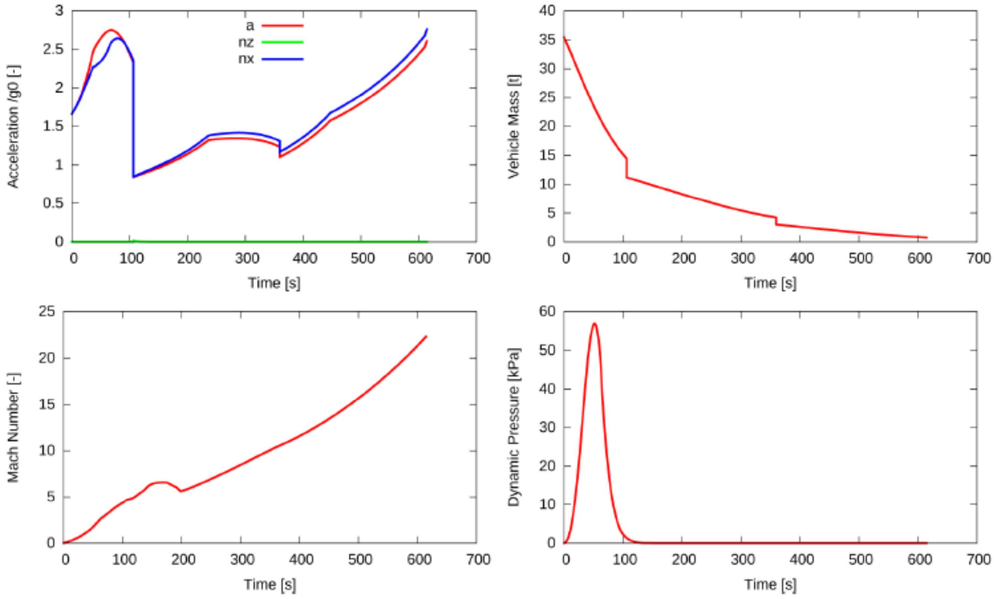

Figure 8: Acceleration, vehicle mass, Ma-number and dynamic pressure over time of flight [20].

Table 3: Payloads delivered by G-µL, figures including the propellant consumption for the circularization manoeuvre [20].

Target Orbit		SSO	LEO
Altitude	[km]	700	500
Inclination	[°]	98	5.3
Payload	[kg]	171	372

The results in Tab. 3 show that the G-µL as conceived above achieves the design target to place a satellite of 200 kg + some mass (172 kg) for support and interface structures in a LEO. Of course the mass placed in a SSO is less.

Because no optimization loops on the G-µL concept have been carried out, there exist some potentials for improvement:

- Increase of the payload by:
  - Adjusting the thrust levels
  - Adjusting the relations of the mass of the stages
- The vehicle has a high *L*/*D* (see Fig. 6). Using larger diameters of the stages
  - May save mass because the bending stiffness increases
  - Allows to accommodate larger satellites with $D > 1$ m in the payload bay of the G-µL
- In contrast, the comparatively large margin of 15 % for the mass estimation of each stage may be completely used up because the mass estimation is not based on related experience with GRM systems of a similar size.

Technical features that may be incorporated but have not been studied are for example a mixed tank pressurization system using a combination of inert gas tank and solid gas generator, or the extended use of advanced materials like CMC for the uncooled GRM CC. These elements have a mass-saving potential, but increase the cost compared to the technical solutions outlined above.

DOI: 10.13009/EUCASS2017-154

2.5 Roadmap to realization of the G-µL

As outlined in chapter 2.2, the relevant technical steps towards realization of a G-µL, compared to existing and tested hardware, are:

- Significant scaling up of all parts of the GRM propulsion system compared to existing and proven hardware
- Significantly longer operation times of the GRM for the 2nd and 3rd stage
- Re-ignition in vacuum of the 3rd stage, needed for the circularization burn of an energy-optimized orbital insertion trajectory
- Control of the individual GRM of a GRM block array
- GRP tank design, particularly the collapsible bladder

Intermediate steps of a roadmap to the realization of a G-µL could take advantage of the fact that scaling up for the propulsion systems of the upper stages is less significant. Hence, a reasonable approach could be to start with the development of a propulsion system for a 3rd stage of a µL and then proeceed to the development of a 2nd stage and eventually to a first stage.

An intermediate step towards a 3rd stage GRM propulsion system for a G-µL is a controllable upper stage of a sounding rocket as described in [21]. Elements to be developed for this vehicle are:

- Thrust vector control by a gimballed GRM
- Long operation time
- GRP tank and pressurization system design including the collapsible bladder

These are equally key elements of the upper stage GRM propulsion system of the third stage of the G-µL. The re-ignition of a GRM in vacuum can also be tested and verified with such a vehicle.

The next step could be a thrust and thrust vector controllable stage for sounding rockets as shown in Fig. 5 and described in [17]. Elements to be developed for such a GRM propulsion system are:

- Control of the individual GRM of a GRM block array
- Scaling up of all parts of the GRM propulsion system

Whereas the thrust level of the 2nd stage of a G-µL is about the order of magnitude of that of a G-SoRo stage, ST2 of the G-µL has much more propellant and an accordingly longer operation time.

With further scaling up of the GRM, this leads to the propulsion system of ST1 of a G-µL.

3. Summary

A concept for a Micro-launcher using green GRM technology for the main propulsion system of all 3 stages has been set up by parametric calculations based on related experience, accepted engineering methods and material data, but without optimization loops. The evaluated data have been given to DLR Institute for Space Systems to check the performance of the G-µL concept. The result is that the first rule-of thumb approach is by just -7 % short of the intended performance, i. e. total payload into LEO.

Concluding we can state that an environmentally friendly, safe, affordable and REACh-compliant G-µL with full GRM propulsion systems is feasible.

Acknowledments

We highly appreciate the support of Dr. Martin Sippel and Jascha Wilken form the DLR Institute for space systems by calculating the trajectories and performance data based on the concept data submitted to them. We also thank all our collegues that work with endeavour within the GGPT working group.

DOI: 10.13009/EUCASS2017-154

References

[1] Natan, B. and Rahimi, S., “The Status of Gel Propellants in Year 2000” in: *Combustion of Energetic Materials*, pp. 172-194 (eds.: K.K. Kuo and L.T. DeLuca), Begell House, USA, 2002.

[2] Ciezki, H.K., Naumann, K.W., Weiser, V., “Status of Gel Propulsion in the Year 2010 with a Special View on German Activities”, *Deutscher Luft- und Raumfahrtkongress 2010*, Hamburg., Germany, August 31 – September 2, 2010.

[3] Schmid, K., Ramsel, J., Naumann, K.W., Stierle, R.; Weiser V., “Raketenmotore mit Gel-Treibstoffen – Stand der Technologie bei Bayern-Chemie”, *Deutscher Luft- und Raumfahrtkongress 2012*, Berlin., Germany, September 10-12, 2012.

[4] Kirchberger, C., Ciezki, H. K., Hürttlen, J., Schaller, U., Weiser, V., Caldas-Pinto, P., Ramsel, J., Naumann, K.W., ”An Overview of the German Gel Propulsion Technology Program“, *AAAF Space Propulsion Conference*, Roma, Italy, May 2-6, 2016.

[5] Ciezki, H.K., Kirchberger, C., Stiefel, A., Kröger, P., Caldas Pinto, P., Ramsel, J., Naumann, K.W., Hürttlen, J., Schaller U., Imiolek, A., Weiser, V., ”Overview on the German Gel Propulsion Technology Activities: Status 2017 and Outlook“, *7th European Conference for Aerospace Sciences*, Milano, Italy, July 2-6, 2017.

[6] Stierle, R., Schmid, K., Ramsel, J., Naumann, K. W., “Free-Flight Demonstration of the Gelled Propellant Rocket Motor of MBDA-Bayern-Chemie”, *4th European Conference for Aeronautics and Space Sciences*, St. Petersburg, Russia, July 4 – 8, 2011

[7] Ciezki, H.K and Naumann, K.W., “Some Aspects on Safety and Environmental Inpact of Gel Propulsion“, *Propellants, Explosives, Pyrotechniques*, invited publication for special issue, ”Insensitive Munitions“, Vol. 41, pp. 539-547, 2016.

[8] Ciezki, H.K , Naumann, K.W., Kobald, M., “A Short Discussion on Performance, Safet and Environmental Aspects of Gel and Hybrid Rocket Propulsion Systems”, Some Aspects on Safety and Environmental Inpact of Gel Propulsion“, *31st International Symposium on Space Technology and Science*, Matsuyama, Japan, June 3-9, 2017.

[9] Naumann K. W., Ramsel J., Caldas-Pinto P.,Niedermaier H., Scheutzow S., Rienäcker C. M., Thumann A., “Gelled Green Propellant Rocket Motor and Gas Generator Technology at Bayern-Chemie – Status and Applications for Space Systems”, *50th AIAA/ASME/SAE/ASEE Joint Propulsion Conference*, Cleveland, OH, USA, July 28 – 30, 2014; AIAA-2014-3796

[10] Ramsel, J., Scheutzow, S., Caldas-Pinto, P., Naumann K.W., Thumann A., Kurth G., “Gelled Rocket Propellants – Development and Performance”, *6th European Conference for Aerospace Sciences*, Krakow, Poland, June 29 – July 3, 2015.

[11] Caldas-Pinto, P., Hopfe, N., Ramsel, J., Naumann, K.W., Thumann, A., Kurth, G., “Scalability of Gelled Propellant Rocket Motors”, *7th European Conference for Aeronautics and Space Sciences*, Milan, Italy, July 2.-6, 2017

[12] Caldas-Pinto, P., Ramsel, J., Scheutzow, S., Naumann K.W., Thumann A., Kurth G. “Control Characteristics of a Gel Propellant Throttleable Rocket Motor”, *6th European Conference for Aerospace Sciences*, Krakow, Poland, June 29 - July 3, 2015

[13] Naumann K.W., Caldas-Pinto, P., Ramsel, J., Niedermaier, H., Risse, S., Thumann A., “Green, Highly Throttleable Gelled Propellant Rocket Motors – State and Application Potentials”, *AAAF Space Propulsion 2016 Conference*, Rome, Italy, 2.-6.5.2016

[14] Naumann, K.W., Caldas Pinto, P., Ramsel, J., Scheutzow, S., Thumann, A., Niedermaier, H., Hopfe, N., Kurth, G., “Gelled Propellant Throtteable Rocket Motor and Gas Generator Technology at Bayern-Chemie”, *6th European Conference for Aerospace Sciences*, Krakow, Poland, June 29 - July 3, 2015

[15] Naumann, K.W., Tussiwand, G, Caldas Pinto, P., Hopfe, N., Ramsel, J., Eineder, L., Risse, S., Thumann, A., Niedermaier, H., Kurth, G., “Green Gelled Propellant Gas Generator for High-Performance Divert- and Attitude Control Systems”, *51st AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit*, Salt Lake City, UT, USA, July 25-29, 2016, AIAA 2016-4667

[16] Naumann, K.W., Ramsel, J., Caldas-Pinto, P., Niedermaier, H., Scheutzow, S., Thumann, A., “Rockets with Green Gelled Propellants – A Promising Technology for Sounding Rockets”, *22nd ESA Symposium on European Rocket and Balloon Programme and Related Research*, Tromsø, Norway June 7-11, 2015

[17] Naumann, K.W., Ciezki, H.K., Caldas-Pinto, P., Ramsel, J., Meyer, T., Rest, S., “A Modular Sounding Rocket Concept with Green, Safe and Affordable Gelled Propellant Rocket Motors”, *23rd ESA Symposium on European Balloon and Rocket Programmes and related Research*, Visby, Sweden, June 11-15, 2017

[18] https://en.wikipedia.org/wiki/VSB-30; May 17, 2017

[19] Garcia, A., Yamanaka, S.S.C., Barbosa, A.N., Bizarria, F.C.P., Jung, W., Scheuerpflug, F., “VSB-30 Sounding Rocket : History of Flight Performance”, *J.Aerosp.Technol. Manag.*, São José dos Campos, Vol. 3, No. 3, pp. 325-330, Sept.-Dec. 2011

DOI: 10.13009/EUCASS2017-154

[20] Wilken, J., Sippel, M., “Vorläufige Analyse eines Micro-Launchers mit Geltreibstoff-Raketenmotoren“, private communication, May 2017
[21] Caldas-Pinto, P., Naumann, K.W., Ciezki, H.K., Ramsel, J., Meyer, T., Rest, S., "Stage Concept for a Hovering Thermosphere Probe Vehicle with Green, Safe and Affordable Gelled Propellant Rocket Motors", 23rd ESA Symposium on European Balloon and Rocket Programmes and related Research, Visby, Sweden, June 11-15, 2017