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Highly reusable LOx/LCH4 ACE rocket engine designed for SpacePlane: Technical Maturation progress via key system demonstrators results

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Abstract

LOX/Methane is currently regarded worldwide as a propellant combination for future rocket-propelled vehicles. ArianeGroup, (previously Airbus-Safran-Launchers) had studied this technology for low-cost expandable launcher application, and moreover confirmed the selection of this propulsion technology in 2008 (at that time Astrium) for the other end of application potential: a highly reusable rocket propulsion system for the innovative SpacePlane vehicle.

In order to prepare for future launch systems, Astrium, now ArianeGroup, had been investing since few years in research and technology of LOX/Methane engines through a demonstrator program including the design and manufacturing of a 400kN class thrust chamber, a Gas Generator, and a TurboPump (done in cooperation with IHI Corporation of Japan).

This paper will briefly recall the ArianeGroup LOx/Methane R&T project with above mentioned rocket engine critical sub-system demonstrators tests.

A general presentation of ArianeGroup "Rocket Propulsion System" and associated ACE42R engine concept for a SpacePlane application will be provided. The main design drivers are presented: LOx/Methane rocket propulsion technology, highly reusable, Designed-to-Safety and Designed-to-Cost, together with the essential requirement, and the general architecture.

This paper will then focus on the maturation obtained, through those demonstrator tests (done at scale 1 against a SpacePlane technical and economical typical requirement), crossing the demonstrators test objectives achieved with the capability of ACE42R design foreseen, and putting in evidence some world-class results obtained for this technology.

Acronyms/Abbreviations

ACE	ArianeGroup Cryogenic Engine
CC	Combustion Chamber
DLR	German Aerospace Centre
FCSB	Flight Control System Ballistic
FDIR	Failure Detection Isolation Recovery
GG	Gas Generator
H/W	Hardware
I/F	Interface
LCH4	Liquefied Methane
LOX	Liquid Oxygen
MPa	Mega Pascal
NPSP	Net Positive Suction Pressure
PDM	Pathfinder Demonstrator Model
PP	Power Pack
RAMS	Reliability, Availability,Maintainability, Safety
RPS	Rocket Propulsion System
SoA	Sub-Orbital Aeroplane
TC	Thrust Chamber
TP	Turbopump

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1. The ArianeGroup LOx/Methane propulsion R&T project: Summary

In order to prepare for future launch systems, Astrium (now ArianeGroup) had initiated investment for a few years-long project in Research & Technology concerning LOX/Methane propulsion including engines. The scope of this R&T project was to mature the core-technology aspects, in order to allow for a large range of applications, from low-cost expendable launch vehicle to highly-reusable propulsion for future regular space flight vehicle.

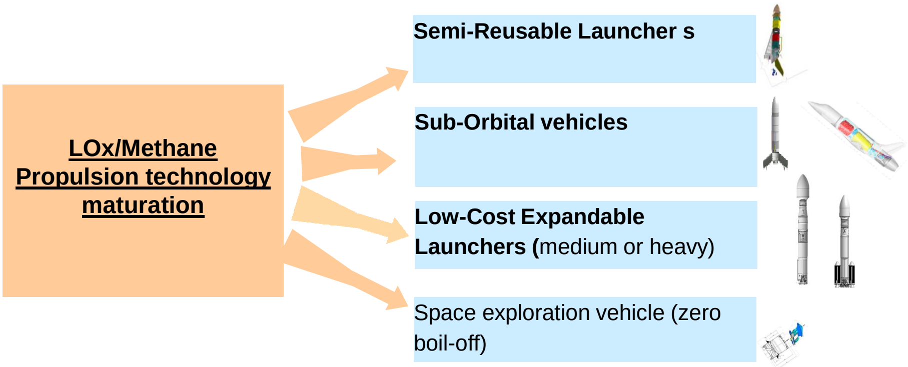

Figure 1: Application scope of Astrium (now ArianeGroup) LOx/Methane R&T

The low-cost expendable part was investigated with a driving "Design-to-Cost" approach. Enabling manufacturing and design/architecture features for engine together with stage propulsion bay, were identified.

The High-reusability part, with application to a passenger-transport sub-orbital vehicle with high flight rate (some flights per week), based on Airbus DS SpacePlane high level needs, was identified as the most demanding technological effort, having however many aspects which allow a spinning-out to expendable LOx/Methane as well. A set of Pathfinder Demonstrator Models (PDM), dedicated to the most critical rocket engine sub-systems, was decided as part of the LOx/Methane propulsion R&T project. They were therefore built for answering the requirement for a Spaceplane rocket engine (42t thrust and highly reusable, dubbed ACE-42R), and sized at scale 1.

They consisted in the 3 following most technology driven critical sub(systems:

- A Gas Generator PDM (GG-PDM), tested in 2013
- A Turbopump PDM (TP-PDM), tested in 2015
- A Thrust chamber PDM (TCA-PDM, also named "romeo"), tested in 2016
- As an option: a Power-Pack PDM (consisting in the assembly of individually tested GG PDM and TP PDM above mentioned), which could be tested by 2018/2019

Note that by design, and due to the coherency of the technical requirement used, as well as to the coherency implemented for physical interfaces and functional performance (and its adjustment), those demonstrators can be assembled into a demonstrator engine, dubbed ACE-35R: this is a potential option allowing the propulsion of a LOx/Methane flight vehicle demonstrator, or a technological demonstration, if any needs are expressed.

It is worth mentioning that ahead of PDM tests, subscale and equipment tests have been performed in order to validate the design choices of critical components (as injector elements, combustion damping elements, dynamic seals and bearings, for mentioning a few), following a step by step maturation approach driven by optimum risk reduction.

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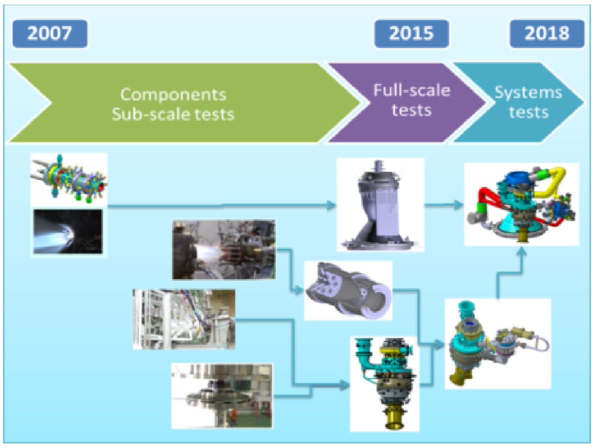

Figure 2. Airbus DS LOx/Methane R&T PDM demonstrators logic and schedule

2. General presentation of ArianeGroup highly reusable Rocket Propulsion System

While the LOx/Methane R&T demonstrators above mentioned were designed on basis of the requirement of Airbus DS Spaceplane, these Rocket Propulsion System (RPS) general features are presented here;

The RPS chosen architecture resulted from many trade-offs, including for design parameters, or design choices in some cases, driven by the lay-out in the vehicle (flight stability, access for maintenance, safety analysis, among others), not developed here.

RPS is made around 2 separate insulated propellant tanks (one for LOx and one for LCH4) which diameter is driven by the vehicle fuselage diameter (optimised vehicle / propulsion solution), which internal equipment feature, in particular, devices for avoiding large bulk liquid movement and damping unwanted sloshing. 2 feed-lines route the propellant to the rocket engine (which is described further). The total amount of stored propellant is about 10t.

The pressurisation of propellant tanks is achieved by mean of vaporised Methane (for LCH4 tank), tapped off the engine thrust chamber cooling circuit, and by mean of Helium for the LOx tank. Helium is stored at high pressure in several composite tanks located according to lay-out constraints. Of course, specific pressure and flow control equipment and heat exchangers are part of the pressurisation system.

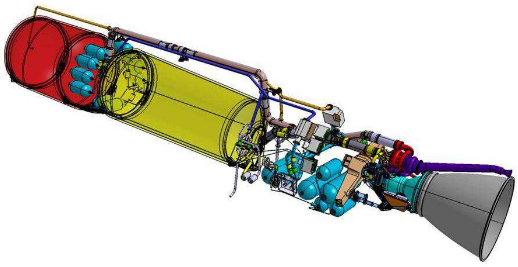

Figure 3. LOx/Methane Rocket Propulsion System physical architecture for a sub-orbital aeroplane

The rocket engine is of Gas-Generator cycle type, with regenerative cooled (LCH4) thrust chamber, and a single shaft 2MW turbomachine designed and manufactured by the cooperation partner IHI (Japan). The Nozzle extension is also fully regeneratively cooled (cooling circuit is interconnected with the thrust-chamber one). Engine trimming is done during acceptance hot-firing test, and is then passive. The starter is a patented architecture, which consist in a spin-start fed by a by-pass circuit from aero-engine compressors.

The rocket engine is named ACE-42R, after the thrust level of 42t at ignition altitude (about 10km), and the design for high reusability.

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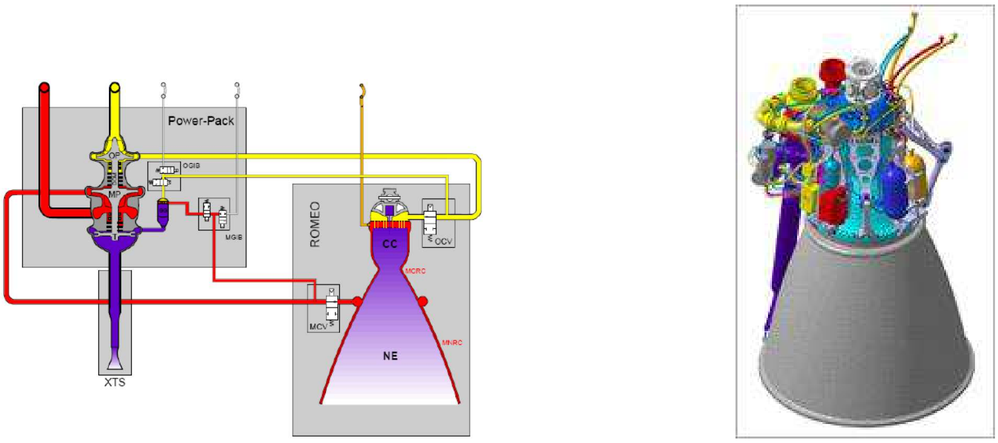

Figure 4. LOx/Methane highly reusable Rocket Engine ACE-42R

The main operational objectives for the ACE-42R engine are summarized in following Table1.

Table 1: ACE-42R engine main characteristics (reference conditions)

Operational life cycle: > 30 flights	
Total Thrust	420 kN
Engine ISPv	343.4 s
Pump inlet total Mass flow rate	124.9 kg/s
Engine mixture ratio	3,12
Chamber pressure	4,7 MPa

3. Rocket Engine main design drivers / essential requirement,

The engine main design choices and requirement elaboration were not made for the sole engine “per se”, but are the result of a concept, then preliminary design analysis and requirement elicitation, done at Rocket Propulsion System level and its integration aspects in the vehicle.

The different trade-offs have been driven by vehicle top-level objectives (translated into trade-off parameters) and an associated set of weighing factors. These trade-off parameters and their ranking are given, here, without the weight value but from highest weight (letter “*a*”) to the smallest weight (letter “e”). They are shown in Table 2.

Table 2: Trade-Off criteria ranking

a	**Design-To-Safety**
b	**Design-To-total Cost**
c	**Design-To-Performance**
d	**Design-for-Budget**
e	**Design-for-Environment**

Note that the prime requirement is related to safety: this is reflecting the project objective of a certified vehicle and rocket propulsion (main rocket propulsion, and FCSB attitude control system which is Reaction Control System

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like with small thrusters). The certification will be given by an independent safety agency, typically the European Aviation Safety Agency (EASA). Let us say here that the level of safety will be of some orders of magnitude higher than the today “manned space safety level”.

The second highest weighing factor is related to Total Cost, which takes into account not only the cost of newly produced vehicle propulsion system, but also the flight operation costs (in particular check-outs, fluid and propellant loading for the RPS) and the maintenance cost (i.e. the rocket engine exchange after every 30 flights, or more if achieved by design, but also the propellant tanks at the other end of life capability). The need for high-reusability of the rocket engine and all other RPS equipment, in the standard of current liquid rocket propulsion technology, is directly coming from the economic target, let us say: the RPS cost per flight. As Astrium, then Airbus-DS, made it public at that time, the flight ticket price is in the range of 200 to 250 kEUR. This price has been broken-down to the lowest significant level of hardware and software, as well as to the flight operations and maintenance (cost of spare and cost of activities).

The performance objectives are only ranking 3 for the rocket propulsion system. This is rather unusual for space propulsion development, but it is the sign of the needed adaptation for innovative vehicle propulsion.

Also innovative of its kind, the design criteria related to minimisation of environmental impact (so called “Design-for-Environment”, “Eco-design”) has been introduced in the early approach for the Spaceplane Rocket Propulsion System, and for the LOx/Methane rocket propulsion technology project. The ultimate target is to reach a zero environmental impact product: this is expected to become the requirement, sooner or later, for any new transportation system, hence the implementation of the propulsion for this innovative vehicle. Even if the ultimate target is not fully reached, the effort and new engineering approach will be very beneficial for progressing along this trend.

Globally, the major design solutions, resulting from the design analysis and trade-offs by applying the above mentioned design criteria and weighing factors can be illustrated by the significant data provided in the Table 3 here below:

Table 3: Selection of most significant technical requirement resulting from design optimisation

Ranking	**General Design Criteria**	**Most significant tech. requirement for Rocket Engine ACE-42R**	**Value or characteristic**
a	**Safety**	1 FS + remaining probability of catastrophic failure < $10.E^{X}$ *(X being currently a proprietary information)*	Includes tolerance to failures
		Compliance to anticipated Certification Specification for SoA rocket propulsion	Includes tolerance to extreme conditions and non-nominal conditions
b	**Total Cost**	TCA life-cycle Reduced thermo-mecha loads & consequent design and sizing choices	>30 cycles Pc_{Ref} = 4,7 MPa (not detailed here)
		Turbo-Pump and Gas Generator life cycle Reduced thermo-mecha loads & consequent design and sizing choices	>60 cycles GG MR_{Ref} = 0,37 (not detailed here)
c	**Performance**	Total reference mass-flow-rate (Pc_{Ref} =4,7 MPa)	125 kg/s
		ISPv (reference)	343 s
		Engine Mixture-Ratio (reference)	3,12
d	**Development Cost and risks**	(Existing project Risk File, and Development cost evaluation)	_
e	**Environmental Impact**	Green-house gas emission balanced to zero over life cycle	i.e.use of bio-sourced propellant
		Row material consumption limitation over life-cycle	
		Toxic products released = zero	

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4. Summary of ACE-42R, LOx/LCH4 highly reusable engine, critical sub-system demonstrators tests

In order to mature the LOx/Methane rocket engine technology, and the most demanding application which is related to high reusability, Airbus DS (now ArianeGroup) has been investing since a few years in a R&T demonstrator program, which included elementary components or small scale element testing, for consolidating technical choices and for anchoring some fundamental aspects of mathematical modelling, and ultimately lead to the design and manufacturing of a 400kN class Thrust Chamber, a Gas Generator, and a TurboPump (done in the frame of a B-to-B cooperation project with IHI of Japan), all based on the same engine level requirement (i.e. the ACE-42R reusable engine, including throttling-ability requirement for down to 25% power). This series of scale 1 demonstrators was called the “PDM program”

Two main testing facilities contributed to the PDM test campaigns:

- DLR / Lampoldshausen, Germany for Thrust-Chamber and Gas-Generator tests (P8 and P3.2 benches)
- IHI / Aioi, Japan, for Turbo-Pump and capable of future Power-Pack tests

The Gas-Generator tests have been done end 2013 (and a new series is planned in 2017 in order to test final design choices, “flight-like” feed circuit and technologies for low-cost/high-reliability ignition system).

The Turbomachine has been tested in 2015. The Thrust-Chamber has been tested in 2015/2016

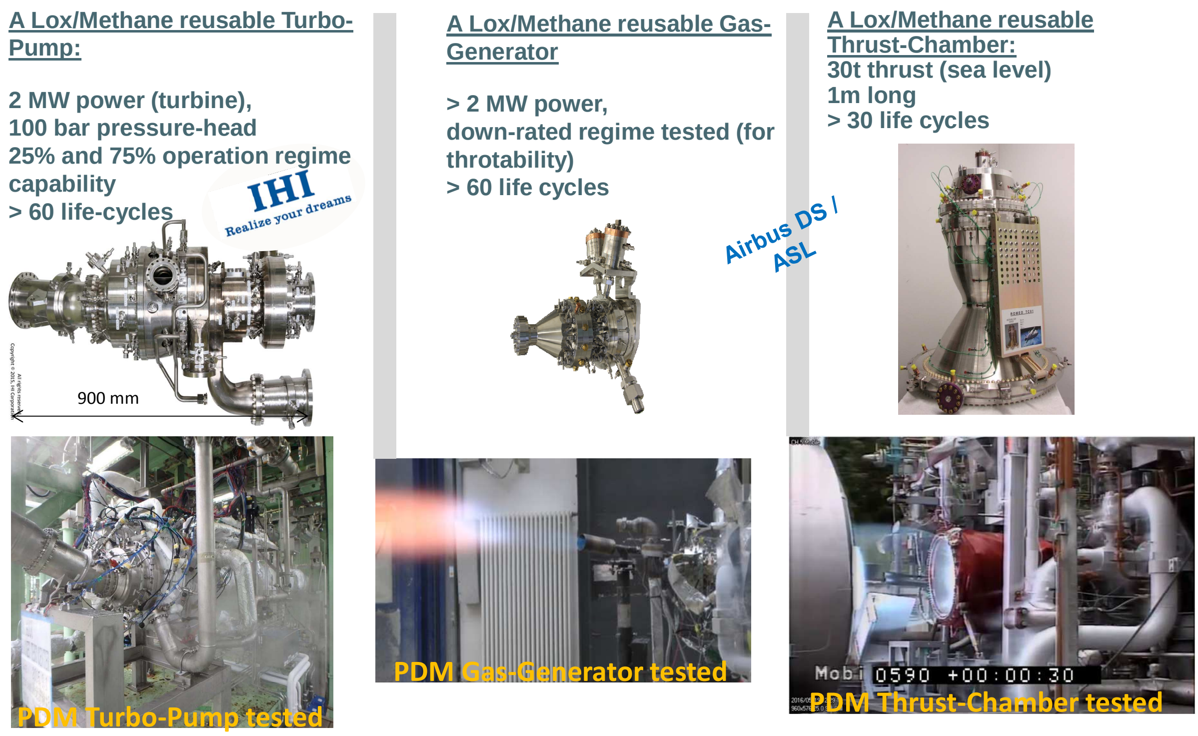

Figure 5: Overview of the LOx/Methane engine sub-system demonstrators tested

The Figure 5 provides some views of the demonstrator hardware, and of their test in the relevant facilities. The main characteristics are recalled. Test results were presented in the reference [1].

5. Maturation obtained, crossing the demonstrators results with the foreseen capability of ACE42R design

Considering the Table 3, which provides the driving criteria for the design of a sub-orbital aeroplane reusable rocket propulsion system, the following paragraphs present a summary of the maturation of the LOx/LCH4

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propulsion technology which has been obtained, for this kind of demanding and innovative vehicle application, by mean of the demonstrator program.

5.1 Criteria of rank “a”: Safety

Needless to say that there were no “statistic testing” in the program, but some of the objectives were constructed in order to test some extreme operating conditions, also covering some typical disturbances in feeding conditions in particular, in order to validate safety-driven design choices.

5.1.1 At level of Gas-Generator:

The program was technology oriented, and included the testing of many different hardware configurations, in order to have a “hardware parametric experience plan”, as presented in reference [4].

For example, 5 different types of injectors were tested, crossed with different injector-head volumes, as well as 3 different hot-gas mixing devices, just to name some of them. Those experimental investigations, coupled with correlation of predictive models (combustion, heat-transfer, flow-dynamics in particular) have allowed to better master the relationship between hardware critical design parameters and effect on combustion and dimensioning conditions. This means that knowledge about design margin was consolidated and fed into predictive tools, thus contributing to failure divergence and prevention ability, and to reliability construction (margins to safety limits).

Focus was put on mastering ignition conditions, on mastering the internal hot-gas heat flux (via film cooling variation up to failure case of cooling channels obstruction, and also via the identification of design parameters for reducing combustion high frequency roughness), and on homogenisation of hot gas temperature in outlet flow area.

On this basis, a reference design for a safe and reusable engine could be selected. The additional GG test program, foreseen in 2017, will then contribute to further experiment the robustness of the GG.

5.1.2 At level of Turbomachine:

The LOx/Methane, highly-reusable Turbomachine demonstrator (designed in a cooperation project by IHI of Japan) test objectives and features were presented in reference [2] and the main results of the test campaign are highlighted in reference [1].

Mitigating the potentially catastrophic risks of a rocket engine turbomachine is indeed extremely influencing the “classical rocket engine” design choices. It is indeed implying a reshaping of the engineering approach. This has been done together by IHI and Airbus DS (now ArianeGroup) in order to implement a coherent and efficient approach embedding , in this case the Turbomachine design, the gas Generator design the engine design, and the vehicle design (involving wide range of design domains as Zone Analysis, or architecture of software system).

The driving approach was, naturally, based on the risk analysis engineering (using RAMS tools) for the vehicle mission with priority on safety specifications determined strongly from civil aviation certification rules as, but not limited to, reference [3]. We can cite here for example 2 major risks related to debris generated by rotating parts, and to the initiation of fire.

The consequences on the design choices for fulfilling the safety requirement (which has the highest ranking) were numerous, and had to, of course, also comply with the other requirement for cost (in particular high re-usabity and the consequent wearing-out of some life critical parts) and performance.

Some particularly specific features and approach had to be implemented, as example for the design and dimensioning of the 2-stage turbine disk assembly, or the mechanical design combined with shape for liquid propellant impellers. The robustness of bearing cooling, of sealings for rotor parts, of tolerance to particle pollution was also influencing design choices.

Moreover, safety barriers implemented through anomaly monitoring are part of the design, and have to be associated to data behaviour analysis and anomaly detection criteria, to be managed by a FDIR system.

All those design features are influencing the performance of the turbopump, global (hydraulic performance) but also locally inside the machine, and were evaluated thanks to the test campaign performed in IHI Aioi test centre.

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Particular safety related, test have been performed, with extreme and non-nominal conditions for pump suction (see Figure 6), including gas ingestion test, and also tolerance to disturbed inlet flow, to particles (see Table 4) , for example.

The test model was heavily instrumented also for allowing the identification of best suited measurement and data treatment for FDIR function.

The obtained test results, despite a limited amount of life cycle applied, have allowed validating a lot of safety features and related design choices. Now, the test model is ready for accumulating addition test cycles and experience when associated with ArianeGroup Gas Generator demonstrator (above mentioned) by mean of "PowerPack" assembly test.

5.1.3 At level of Thrust Chamber:

The LOx/Methane, highly-reusable Thrust-Chamber demonstrator tests took place in 2016, and some preliminary results (because the test campaign was still on-going at that time) were presented in reference [1].

The design methodology and adaptation of engineering applied to the Thrust Chamber design toward Safety and other criteria was similar to what is presented in previous paragraph for turbomachine.

The test results obtained allowed covering a large amount of operation cycles: up to 46 ignitions and 36 life-cycles with the same hardware and without any refurbishment (in particular, no sealing connection needing repair). One link with safety is related to the contribution for building up the "probability of no-failure", and to the validation, and anchoring, of life modelling, thus allowing to better predict failure initiation modes.

The test model was, for this purpose, heavily instrumented. Here also, FDIR measurement confirmation for later flight operations was part of the test results assessment.

Note that a large amount of "beyond extreme design border" test were performed, in particular at very low chamber pressure, and with forced LCH4 phase change in cooling channels: this allowed to more precisely characterise the thermodynamic and fluidic behaviour of Methane fuel in critical conditions and for design critical functions (cooling, injection).

The occurrence of high-frequency combustion instability which damaged the injector head during one test is remarkable: thanks to the large and appropriate number of sensors, the initiation conditions could be investigated, counter measures were identified, hardware repaired, and new safer ignition and operation conditions validated during the remaining of the test campaign, over the whole extreme design domain.

The obtained test results, and the large amount of hot-firing cycles, have allowed to validate a lot of safety features and related design choices.

5.1.4 Conclusion for criteria "Safety":

Using the Table 4 related to Design-to-Safety criteria:

Ranking	General Design Criteria	Most significant tech. requirement for Rocket Engine ACE-42R	Value or characteristic
a	**Safety**	1 FS + remaining probability of catastrophic failure $< 10.E^{X}$ *(X being currently a proprietary information)*	Includes tolerance to failures
		Compliance to anticipated Certification Specification for SoA rocket propulsion	Includes tolerance to extreme conditions and non-nominal conditions

- Validation of performance and behaviour with design choices driven by safety
- Extreme and non nominal conditions tested and behavioural models adjustment
- Characterisation of some failure conditions and their propagation profile (i.e. pump cavitation, TC high frequency, GG T° peaking, …)
- Verification of FDIR potential measurements and physical significance

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5.2 Criteria of rank “b”: Total Cost (including High-Reusabilty)

The design-to-cost results are no the purpose of this paper, but the results obtained in view of the high-reusability requirement are summarised here. This is a key design feature for reaching the targeted cost-per-flight allocation to the rocket propulsion system, as well as for reaching the minimum flight rate (many flights per week) that is required from a spaceplane operational economical model.

5.2.1 At level of Gas-Generator:

Table 4: GG demonstrator required life versus tested life

Required Nr of life cycles	Performed operation cycles	Operating load-points tested	Remarks
> 60	98 cycles Cumulated time = 4.000s	Pc from 18 to 70 bar Ox/Fuel from 0,05 to 0,7	Good mixing of hot gas achieved, and low amount of cumulated soot deposit

The hardware status is very moderately affected by the accumulated hot-firing cycles at various Ox/Fuel operated conditions. The demonstrated cycles compared to the goa is of more than + 30%, and the hardware is in good health for further testing: it should be used for next step consisting in Powerpack test.

The moderate soot deposit with the LOx/LCH4 combination, despite low combustion mixture ratio as indicated above, is very favourable for high reusability: see typical views in Figure 6 and figure 7. Similar low impact is expected on the turbine blades and stator.

Figure 6: injector face-plate featuring low soot deposit

Figure 7: turbulence ring, coloured with few soot

The control of GG chamber wall heat-flux via film cooling, and impact of film cooling variation, has been settled during test, and relevant combustion/cooling models were correlated: this makes the design ability more robust for life capability of a final flight chamber configuration.

5.2.2 At level of Turbomachine:

Table 5: TP demonstrator required life versus tested life

Required Nr of life cycles	Performed operation cycles	Operating load-points tested	Remarks
> 60	8 cycles	Low Power mode: 25% Intermediate power: 75% Full Power: 100% (2MW on turbine)	At Turbomachine level
Life-time: >60*80s (> 4.800 s cumulated)	8.900s cumulated running time	Maximum design rotational speed and maximum mechanical load	Test at level of critical dynamic elements: bearings together with dynamic seals on the real rotor assembly.

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Only little number of operating cycles has been performed in the first test series of IHI designed turbomachine (see reference [1]) The test model is planned to be coupled to ASL Gas Generator demonstrator in order to perform additional hot-firing PowerPack test.

However, life critical dynamic components (i.e. the bearings, with their cooling system, and the dynamic seals) have been tested beyond the cumulated life requirement, in extreme load conditions, and a final design has been selected and has demonstrated more than 1,8*nominal cumulated life time in real conditions (flight design rotor axis and cold conditions).

At level of turbomachine assembly test, focus has been put on extreme operating conditions in order to measure, and model the life sensitive behaviour of the hardware. This is what has allowed, for example, to identify a need for increasing cooling flow of bearings, in low power mode (25% of nominal power / rotational speed), in order to increase the cooling margin, and thus increasing life capability margin. The modification could be done by using some transducers ports, connected by tubing, thus avoiding to dismount and machine modifications on the turbopump. All other features, designed against high-reusability, have been verified with positive results, giving confidence in the ability to go-on testing in a PowerPack configuration.

5.2.3 At level of Thrust Chamber:

Table 6: TC demonstrator required life versus tested life

Required Nr of life cycles	Performed operation cycles	Operating load-points tested	Remarks
> 30	46 ignitions (among which 10 red-line stops after ignition). 36 operation cycles	Low Power mode: 25%Pc Full Power: 100% Pc Ultra-low Pc (stability test)	Extremely wide range of (Pc;Ox/Fuel) domain tested.

The tested TC demonstrator (also nick-named “Romeo”) has remarkably achieved the high reusability target which was set for the design, and even beyond by +20%. The hardware has been inspected and x-rayed, and presents no initiation of life-typical damage, in particular no crack initiation yet in the chamber internal liner.

The performed series of hot-firing is described in Figure 8 here below:

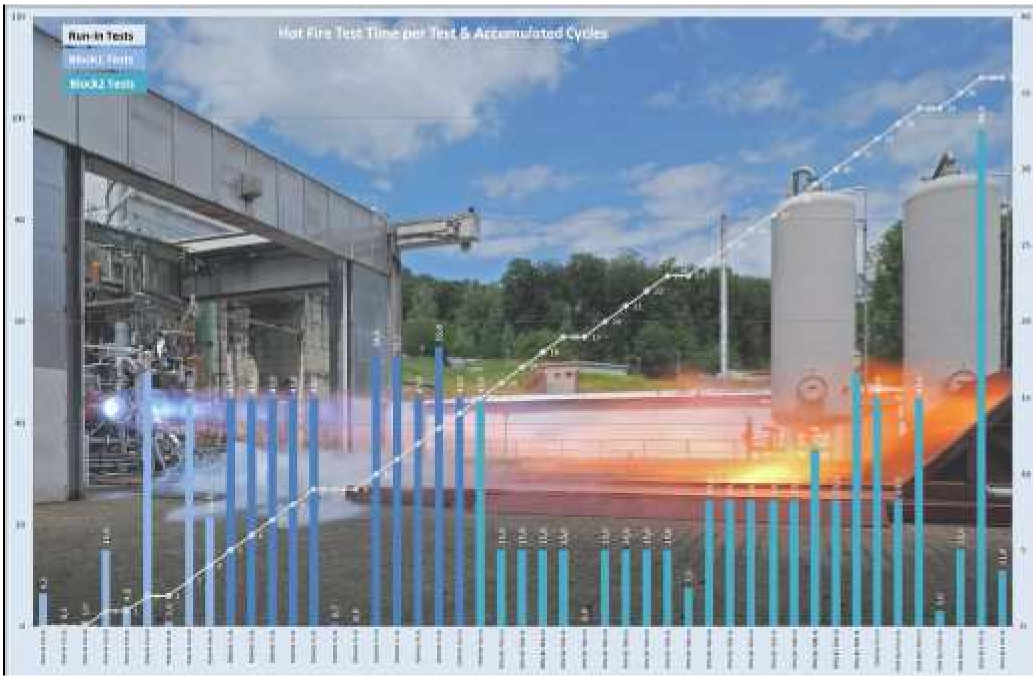

Figure 8: history of performed test and accumulated life cycle for TCA demonstrator.

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The different sealing design for long life did also perform well, without requiring any change during whole campaign;

It has to be noticed that 1 dismounting of the injector head was performed in order to repair it after the high-frequency experience, in order to change the injection elements (design modifications for stability margin). The relevant seals were replaced on this occasion, early in the test program (6th test). In term of life cycle, a higher heat-flux was experienced for a short period, damaging the injectors, but the complete test duration was however achieved, in those degraded conditions, showing the robustness of the design provided that the operating mixture-ratio is controlled (which was achieved by control ground propellant flow-valves)

Also worth to be noted, the cycle accumulated covered ignition sequence set with aLCH4 lead, and also sequences modified for achieving a LOx lead during transient.

Various operating load points [Pc;Ox/Fuel] have been tested as presented next paragraph, and accounted for the life capability demonstration.

From the measures heat-loads and pressure from the different test performed, the life model correlation results lead to an expectation of 2 times the required life capability. The TC is kept ready for further testing, should budget be available. It could also be used for an engine demonstrator when assembled with the other engine sub-system demonstrators also tested (TP and GG).

5.2.4 Conclusion for criteria “High Reusability”:

A very strong technical maturation of the high-reusability design choices and technology selected could be achieved through this LOx/Methane demonstrator R&T test program, at a scale typical of operational need (in particular, based on a 42t thrust engine) which allows correlatively a significant risk reduction for any project requiring such capability. The Table 7 summarises the results achieved versus the targets.

Table 7: Overview of demonstrators required lice versus tested life

Required Nr of GG life cycles	Performed operation cycles	Operating load-points tested	Remarks
> 60	98 cycles Cumulated time = 4.000s	Pc from 18 to 70 bar Ox/Fuel from 0,05 to 0,7	Good mixing of hot gas achieved, and low amount of cumulated soot deposit
Required Nr of TP life cycles			
> 60	8 cycles	Low Power mode: 25% Intermediate power: 75% Full Power: 100% (2MW on turbine)	At Turbomachine level
Life-time: >60*80s (> 4.800 s cumulated)	8.900s cumulated running time	Maximum design rotational speed and maximum mechanical load	Test at level of critical dynamic elements: bearings together with dynamic seals on the real rotor assembly.
Required Nr of TC life cycles			
> 30	46 ignitions (among which 10 red-line stops after ignition). 36 operation cycles	Low Power mode: 25%Pc Full Power: 100% Pc Ultra-low Pc (stability test)	Extremely wide range of (Pc;Ox/Fuel) domain tested.

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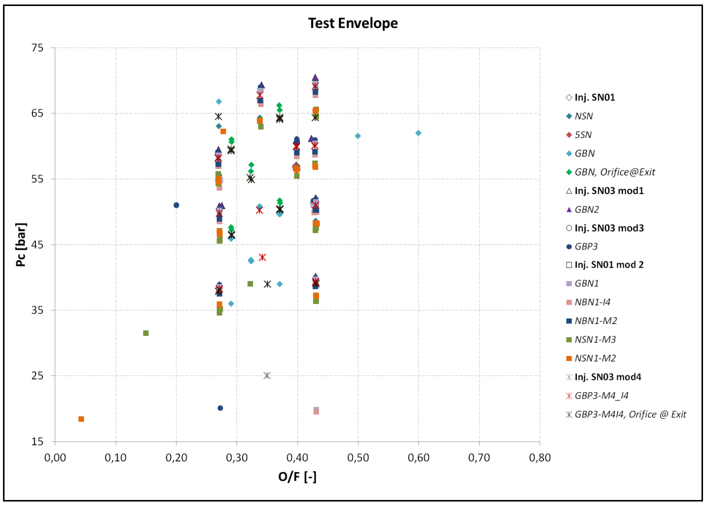
5.3 Criteria of rank "c": Performance, including Throttleability

For all 3 demonstrators, the design domains have been tested and the good behaviour of the hardware has even allowed to go beyond the extreme domain in order to explore the design limits, and to perform also some pure technological tests related to particular behaviour, such as combustion extinction limit for GG, or LCH4 phase change in Thrust Chamber regenerative circuit. The tests have shown a very good achievement of predicted performance, global and specific, therefore this paragraph puts emphasis on the variable thrust capability (Throttleability) demonstrated at level of each critical engine sub-systems.

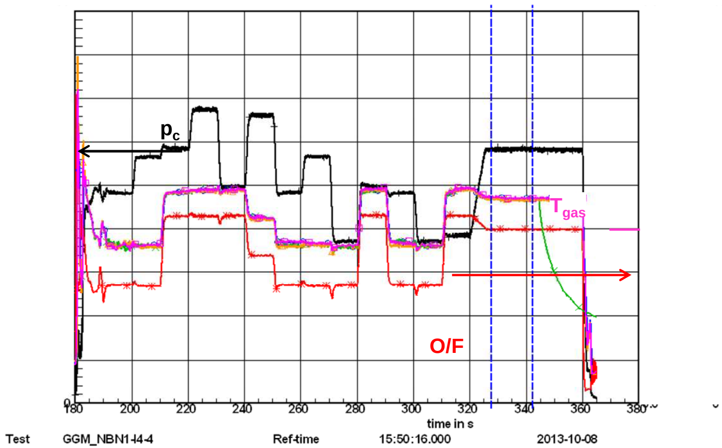
Gas Generator: Test have been performed over an extremely large domain in term of power (chamber pressure and mixture ratio variations, as illustrated by Figure 9) for exploring also the hot gas temperature temperature, for the different injection elements tested and mixing devices (homogeneity / stratification of T° in the hot-gas flow).

Figure 9: domain [Pc;Ox/Fuel] demonstrated with the GG



The Throttleability, from combustion device point of verification, is illustrated with the example of test profile of Figure 10. Lower operating points have also been experienced during a hot-run.

Figure 10: example of a variable power test sequence



In the foreseen operating range of the related engine ACE-42R, which is 25% -to- 100%, the GG has operated without generating high pressure oscillations.

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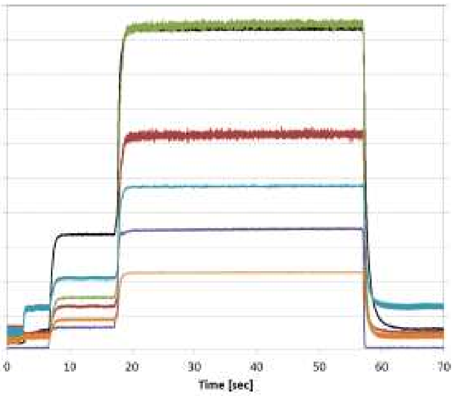
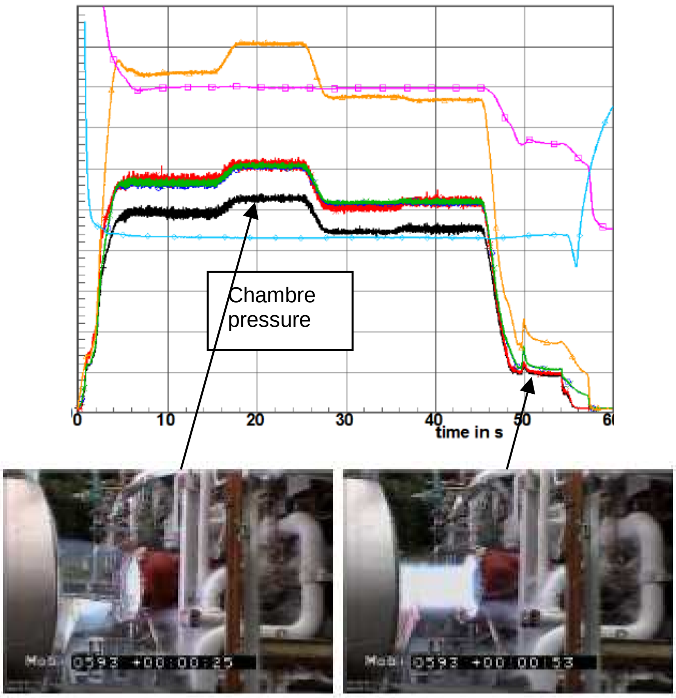
- Turbomachine: Throttleability demonstration tests were performed at the following power levels: 25%, 75% and 100%, with a continuous but rapid variation between Low Power and nominal power, as illustrated below:

Figure 11: typical throttling profile with state at low power (rotational speed=NT1)



Thrust Chamber: The large operating range of load-points tested is shown by the Figure 12, and it could be achieved with variation of feeding conditions by mean of a controller and ground flow control valves. A LCH4 heater was added in order to simulate a regeneratively cooled nozzle but also for allowing parametric LCH4 inlet thermodynamic conditions (technological investigation for LOx/LCH4 propulsion).

Figure 12: PC plot (black) and pictures of TC exhaust during throttling test



Engine: The engineering and design maturation approach is based on the use of a “virtual engine” ACE-42R, which is a set of mathematical models representing engine elements. The demonstrator’s requirements were coherent of this “virtuel engine”, and the test results are also aimaing at validation / anchoring some models for each demonstrators. Those verified models are then re-integrated in the virual engine, and the operability (in particular the trimming and transient management) and performance for different utilisation profile and conditions are

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verified. At the same time, a specific demonstrator engine (dubbed ACE-35R) is also virtualised and adjusted according to test reslts.

5.3 Criteria of rank "e": Environmental impact minimisation

The LOx/LCH4 demonstrator programme was also used for "testing" the way to introduce Design-for-Environment in the Rocket Propulsion System Engineering structure and working logic. A few desing trade-offs were done for adjusting a methodology. Some particular studies were performed in order to get first usable results (for design trade-off), in particular concerning the life-cycle analysis of Rocket Propulsion System for the step "manufacturing" of the hardware, based in particular on the material and manufacturing process applied on the demonstrators presented here. An interesting output was the computation of the environmental impact reduction brought by the recycling of the hardware at end of life (made possible thanks to the recovery / reusability). Some orientation to change the type of material were also identified fo reducing amount of toxic products produced from raw material mining to parts finishing.

The most spectacular approach, and technical progress, was the application of "emission balancing" approach by testing Bio-sourced liquid methane, which quality was carefully analysed and fixed for a given production process (vailable on commercial market) and which has been used for 100% of TC test campaign.

Figure 13: From bio-methane production up to Bio-LCH4 use in combustion test

All those results can start feeding a set of environmental data, for orienting a development phase, and the approach allows to further elaborate a methodology for early introduction of Design-for-Environment in development logic.

6. Conclusion

A very strong technical maturation of the high-reusability design choices and technology selected could be achieved through the LOx/Methane demonstrator R&T test program performed by Airbus-DS (now ArianeGroup) and its cooperation partner IHI for the turbo-pump, between end 2013 and end 2016.

This R&T project, with requirement based on the demanding application for a sub-orbital, passenger transport, aeroplane, related to engine ACE-42R concept, has driven to achieve some World-Top class results in the technology of Lox/Methane Highly-Reusable rocket engine , in particular with the combined demonstration of a so high reusability of major subsystems (i.e. the 36 hot-firing cycles of TC), large throttling capability for all major sub-systems (100% down to 25%), representative test-models at scale 1 (related to a 42t engine), in addition to beyond-the-limit test cases. And all hardware are in good health, as expected from design choices done, and ready for further hot-firing testing in order to demonstrate more margin.

Those results, with the associated design and modelling tools maturation, pave the way to a lower risk application of the studied Engine and Rocket Propulsion System for innovative vehicles projects that are burgeoning, in particular for sub-orbital aeroplane (which was used as requirement baseline) and also re-usable launcher stages, among others.

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