LOX/METHANE GAP ASSESSMENT REPORT

1/25/2016

International Agency Working Group

Lox/Methane (or Liquefied Natural Gas) is an enabler for future exploration with in-situ propellant production, improved performance, and fluid commonality. The team found common technology issues and gaps. All of the partners find value in continued investment through partnerships.

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LOX/METHANE GAP ASSESSMENT Report

INTERNATIONAL AGENCY WORKING GROUP

GOALS, OBJECTIVES, AND APPROACH

The International Space Exploration Coordination Group (ISECG) formed two Gap Assessment teams to evaluate topic discipline areas that had not been worked at an international level to date. Accordingly, the ISECG Technology Working Group (TWG) recommended two discipline areas based on Global Exploration Roadmap (GER) Critical Technology Needs reflected within the GER Technology Development Map (GTDM): Dust Mitigation and LOX/Methane Propulsion. The ISECG approved the recommended Gap Assessment teams, and tasked the TWG to formulate the new teams with subject matter experts (SMEs) from the participating agencies.

<u>ISECG Gap Assessment Teams</u>

- Dust Mitigation technologies
  Participating agencies: ASI, CSA, ESA, JAXA, NASA
- LOX/Methane Propulsion technologies
  Participating agencies: ASI, CNES, DLR, ESA, JAXA, NASA

TWG AND SME GAP OBJECTIVES FOR ASSESSMENT TEAM

The objectives of the Gap Assessment Team were as follows:

- **Identify and make a presentation on technology gaps** related to the GER2 mission scenario (including cislunar and lunar mission themes and long-lead items for human exploration of Mars) at the international level. This presentation should include opportunities for international coordination and cooperation in closing the identified gaps.
- **Produce a gap assessment** in the form of a summary report and presentation identifying those GER Critical Technology Needs. This also should include opportunities for international coordination and cooperation in closing the identified gaps.

Note: A small number of GER Critical Technology Needs will only be considered for the initial technology gap analysis. Additional GER portfolio analysis will be done at a later time, pending the lessons learned and direction of the ISECG.

GAP ASSESSMENT APPROACH (TASKS)

The gap assessment approach involved four tasks:

- **Identification of Key Tasks/Questions:** In coordination with the International Architecture Working Group (IAWG), the Gap Assessment Team reviewed the existing GTDM and portfolio entries for GER architecture details and performance metrics (in accordance with current elements/capabilities tied to the GER 2.0 architecture). We then identified what updates are needed, if any, to the current GTDM portfolio of technology development activities to reflect each respective agency's activities/interest related to the GER.
- **Gap Analysis:** The team identified gaps for the identified technologies and capabilities, initially focused on critical technologies.
- **Options for Gap Closure:** The team identified the key technology/engineering solutions for closing the identified gaps.
- **Identification of Partnership/Coordination Opportunities**

LOX/METHANE TEAM ACTIVITIES

The goals and objectives of this international agency working group are to determine the gaps in technology that must be closed for LOx/Methane to be used in human exploration missions in cis-lunar, lunar, and Mars mission applications. An emphasis is placed on near term lunar lander applications with extensibility to Mars. Each of the agencies has conducted work on LOx/Methane propulsion systems, so subject matter experts are available. The group of subject matter experts reviewed the concept and initial sizing of a lunar lander to extract the required key needs and performance parameters. Each international agency then provided a status of LOx/Methane technology work within their agency, as is included in this report. The gaps were then identified where the technology needs exceeded the current technology state.

EXECUTIVE SUMMARY OF GAPS IDENTIFIED

- ◆ **Develop throttleable regeneratively cooled pump-fed and/or pressure-fed engines** to address gap for throttling (5:1 – 10:1), for Specific Impulse (Isp) of 360-365 sec, and for regeneratively cooled engines in the 30 – 100-kN range.
- ◆ **Develop 100 to 220-N Reaction Control System (RCS) engines with integrated cryogenic feed systems** to addresses gap for thruster size/cost and for evaluation of Guidance, Navigation & Control (GNC) impulse bit and thrust requirements.
- ◆ **Develop cryogenic refrigeration systems** to address gap for capability to maintain zero-boil-off and performing liquefaction of in-situ produced propellants (up to several hundred watts at ~90 K).
- ◆ **Develop composite cryogenic tanks with focus on spherical geometry** to address gap in propellant tank technology.

- **Develop high performance pressurization system that improves storage density and reduces mass** to address gap for cryogenic propellants.
- **Conduct extended duration thermal vacuum testing of integrated system** to address gap of integrated system testing in thermal vacuum environment
- **Fly a zero-g cryogenic liquid acquisition experiment in space such as on the ISS or in a cis-lunar location** to address gap of lack of demonstration of LOx/methane in these conditions
- **Fly a test vehicle in space as a technology infusion mission to demonstrate integrated LOx/Methane propulsion systems** to address gap of no in-space LOx/Methane experience

APPLICATION CONCEPTS FOR LOX/METHANE

LOx/Methane has application for in-space propulsion systems such as service modules, landing or descent vehicles, and ascent stages. LOx / Methane propulsion can also be considered for access to space due to its low cost, high thrust and design of reusable engines for 1st stages of launchers, as well as re-ignitable engines for upper stages. This wide range of applicability should allow for some economy in the development of the required technologies. The fact that LOx is being used also allows for leveraging of existing technologies in many cases.

LUNAR LANDER CONCEPT

ISECG IAWG (International Architecture Working Group) developed the lunar surface mission scenario shown in Figure 1. The human lunar lander is a component of a dual launch mission concept. The lander arrives and rendezvouses with the evolvable Deep Space Habitat (eDSH) and loiters until the crew arrives on a separate launch. After docking, the crew transfers via the eDSH to the lander. The lander carries crew and cargo to the lunar surface and returns crew and samples to the eDSH at the conclusion of surface ops.

There are two configurations under investigation for lunar lander. Both configurations assume that the eDSH location is the NRO (Near Rectilinear Orbit) and transports four crew members to the surface. The main difference between configurations 1 and 2 is the descent stage.

Configuration 1 features a two-stage lander comprised of a descent module (DM) and ascent module (AM). The DM completes all descent maneuvers from the eDSH until touchdown. The DM is left on the surface of the moon while the AM returns to the eDSH for reuse. The cargo variant of the lander does not carry an AM.

Configuration 2 features a lunar module with crushable landing gear (the primary module or PM) combined with a drop stage (DS). The DS completes all descent maneuvers from the eDSH until it is safely disposed prior to touchdown. The crushable structural component of the PM is left on the surface of the moon while the rest of the PM returns to the eDSH for reuse. The cargo variant of the lander includes only the crushable landing assembly and a DS.

The descent module of configuration 1 is 2.7 m in height and 5.2 m in diameter. The main engine propellants are LOX/CH_4. The main engines consist of three 30-kN engines and the RCS thrusters consist of forty 220-N engines. The propellant tanks consist of three fuel, three oxidizer and four helium tanks. The drop stage of the configuration 2 is described as the same as the descent module of configuration 1.

Both human lunar lander concepts are barely meeting the anticipated SLS 40-mT Trans-Lunar Insertion (TLI) capability (not considering the NRO insertion delta-V.). An early presence of the ascent stage in NRO could provide different benefits to the overall exploration campaign.

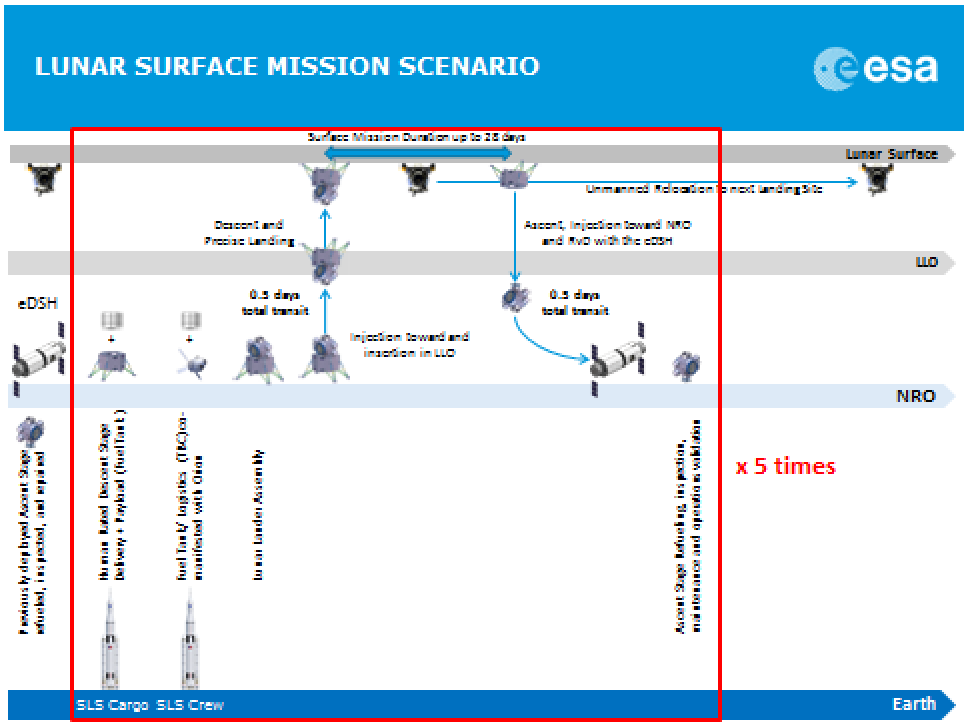


FIGURE 1 LUNAR SURFACE MISSION SCENARIO (NOTIONAL)

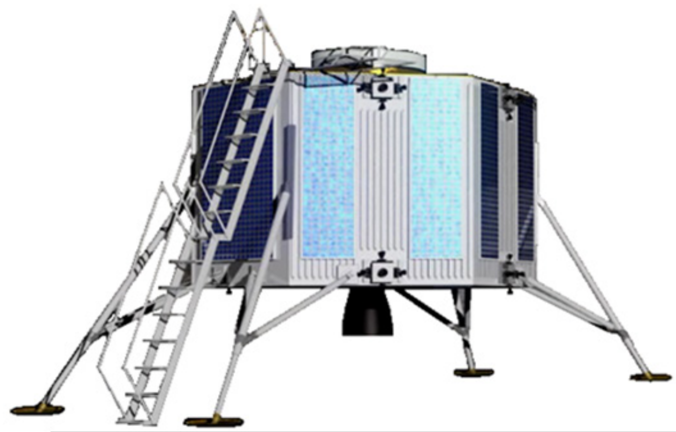
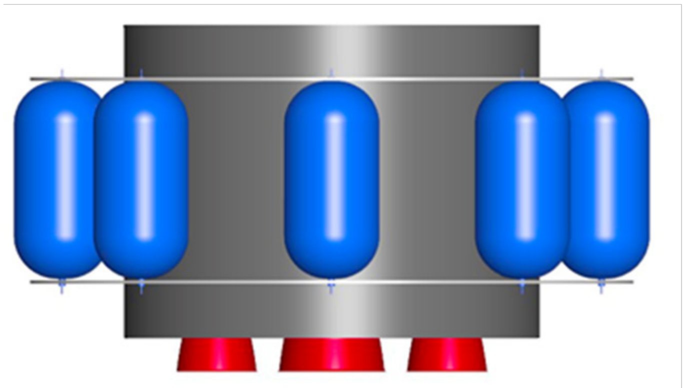



FIGURE 2 CONFIGURATION 1 ASCENT AND DESCENT MODULE (CONCEPTUAL)

TABLE 1 MAIN CHARACTERISTIC OF DESCENT MODULE

Dimensions		
	Height:	2.7 m (4.3 m with legs deployed)
	Diameter:	5.2 m (8 m with legs deployed)
	Structure:	Hexagonal box with central cylinder
Propulsion System		
	Fuel:	CH4
	Engine(s):	3 x 30 kN
	RCS thrusters:	40 x 220N
	Tank(s):	3 fuel, 3 oxidizer, 4 Pressurant
Power		
	Architecture:	Body Mounted Solar Panels
Landing Legs		
	# Landing Legs:	4
Mass Budget		
	Overall Mass:	~ 26700 kg
	Propellant Mass:	~ 22800 kg



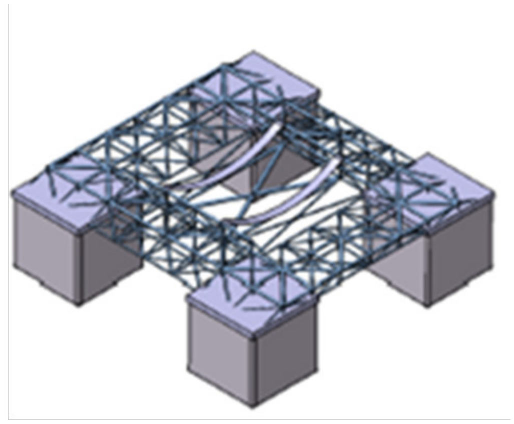


FIGURE 3 CONFIGURATION 2 DROP STAGE AND CRUSHABLE LANDING GEAR (CONCEPTUAL)

MARS ASCENT/DESCENT

For Mars ascent and descent, NASA is considering a methane propulsion system that supports mission architecture goals for common engine development, a long-term surface In-situ Resource Utilization (ISRU) strategy, and soft cryogenic fluid management requirements.[1 2] This will enable an in-space stage powered by a demonstrated workhorse engine, intended for boosting crew and cargo. The oxygen and methane propellant combination has the potential for good engine performance that will provide the required thrust for an entry and descent stage at the destination and ascent propulsion for crew return. The LOx-based economy offers system compatibility that results in a lower outbound vehicle mass and greater payload-carrying capability to the Mars surface with ISRU at the destination to support return missions. [3] NASA plans to demonstrate ISRU technologies that could enable propellant and consumable oxygen production from the Martian atmosphere. If successful, surface assets could be launched ahead of the crew to create oxidizer from the Mars atmosphere, prior to sending crew to the surface. In a long-term strategy, the use of soil processing technology and excavation techniques could allow missions to extract water from the indigenous resources and thereby produce the fuel needed to propel an ascent vehicle to Mars orbit to rendezvous with an Earth return vehicle. This capability could also support future reusable lander strategies/architectures.

FIGURE 4 MARS ASCENT VEHICLE CONCEPT

SERVICE MODULES/ORBITAL TRANSFER VEHICLE

There are several other architectural elements where LOx/Methane propulsion systems can be utilized. A high performance service module for NASA's Orion vehicle or a crew taxi is within the thrust range of this technology. A solar electric propulsion system, with a chemical stage for high thrust burns, is another option. These applications may be more tolerant of different thrust levels than a lunar lander.

LOX/METHANE PROPELLANTS ADVANTAGES AND DISADVANTAGES

LOx/Methane has several major advantages. One primary advantage is that for future Mars exploration it provides the capability to use propellants that are produced in-situ on Mars. Entry, descent, and landing on Mars is significantly easier if ~30 MT of propellant for the Mars Ascent Vehicle is not required to be on–board thereby reducing the MAV from ~40 MT to ~10 MT assuming oxygen and methane are both produced in-situ on the Mars surface.

Carbon dioxide from the Martian atmosphere, combined with proper amounts of water (found in the Martian regolith) and energy, yields both oxygen and methane. Methane production from Martian resources requires less mining than the equivalent thrust impulse production using hydrogen produced from Martian resources.

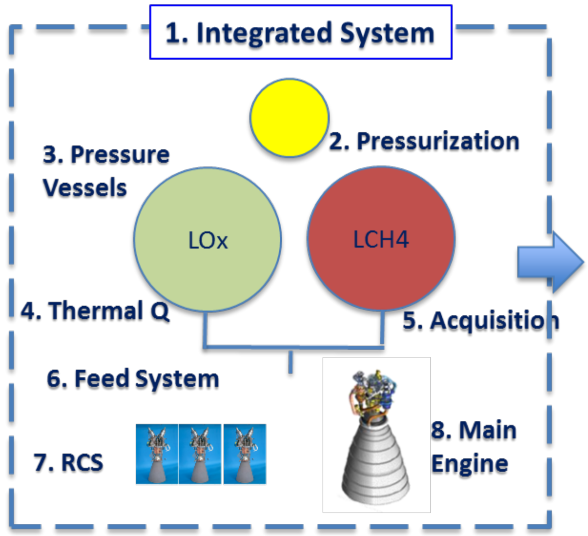
As a bipropellant propulsion system, LOx/Methane has some favorable characteristics for long life, which is critical to lunar and Mars missions and reusability. LOx/Methane propellants are non-toxic, non-corrosive, self-venting, and simple to purge. This alone enables much simpler and more reliable operations, particularly on the Mars surface. There is no extensive decontamination as required with toxic propellants mono-methyl hydrazine (MMH)/nitrogen tetroxide (NTO). The high vapor pressure provides for excellent vacuum ignition characteristics for reaction control engines and main engines. The performance of LOx/Methane is better than current earth storable propellants for human scale spacecraft.

However, unlike current storable propellants, the cryogenic storage aspect of these propellants needs to be addressed. For zero boil-off, passive techniques using shielding and orientations to deep space, or refrigeration may be required to maintain both oxygen and methane in liquid forms as they will absorb energy from both the spacecraft and the environment. An integrated spacecraft design could take advantage of these properties by absorbing the heat leak with small amounts of liquid venting which turn oxygen and methane to a gas. This gas then can be used for fuel cells, life support, or attitude control. Liquid Methane (saturated temperature of 120 K at 0.2 MPa) is thermally similar to O_2 as a cryogenic propellant, allowing for common components and thus providing cost savings as compared to liquid hydrogen (LH_2). In addition, thermal management is more energy efficient due to temperatures at 97 K instead of at 23 K for LH_2. Due to methane having a 6x higher density than hydrogen, it can be stored in much smaller volumes. However, the combination does have a 20% less efficient propulsive efficiency than does LOx/LH_2. For human class landers and spacecraft, the increase in structure mass for the larger volume of LH2 and the system complexity driven by LH2 typically offsets much of the Isp gain at the vehicle level. This is in general not true for earth launch vehicle upper stages though.

There is also strong fluid commonality with other spacecraft subsystems such as Life support (O_2) and Fuels Cells for Power (O_2, CH_4). This commonality can be used to save mass and increase redundancy.

LOX/METHANE TECHNOLOGY AREAS

The integrated propulsion system was subdivided into the critical areas of pressurization system, propellant tank thermal management, propellant acquisition, feed system, reaction control engines, and main engine as shown in Figure 5. Each area was then assessed for performance needs, goals, technology solutions, status at each agency, and the gaps. Table 3 in the appendix contains the spreadsheet summary inputs from each agency.



Lox/Methane Propulsion System Technology Areas	Performance Needs	Goals	Technology Solutions	Status	Gaps
Integrated System					
Pressurization System					
Pressure vessel					
Thermal Management					
Liquid Acquistion in zero/low g					
Feedsystem					
Reaction Control Engines					
Main Engines					

Each Agency made inputs to Needs, Goals, Technology Solutions, Status, and Gaps for each Technology Area

FIGURE 5 TECHNOLOGY AREA DIAGRAM AND EXAMPLE OF SPREADSHEET USED TO IDENTIFY GAPS

CURRENT STATUS OF TECHNOLOGY ACTIVITIES AT EACH AGENCY

ASI

LOx-Methane propulsion in Italy – Synthesis for LOx-CH_4 Working Group Report at ISECG

ASI started its commitment on LOx-Methane technologies more than 15 years ago; initially with small research thrust chambers tested in a dedicated facility in Italy.

In the frame of the Lyra program for the study of evolutions of the Vega launcher, ASI supported a significant activity of design and technologies development dedicated to the realization of the MIRA Demonstrator, a 100-kN (10-tonne) thrust class, expander cycle engine, working with LOx-LCH_4, for the a new upper stage of Vega. The program encompassed the development of critical components working with methane fuel, such as the thrust chamber injector head and the fuel turbo-pump. The demonstrator, developed in cooperation with Russian industry under a dedicated agreement with Roscosmos, was successfully tested in 2012 at level of Thrust Chamber Assembly, [4] and then tested again in May 2014 at the complete engine level, with more than 11 tests performed up to full operating condition, cumulating more than 600 s of firing.[5] Dedicated activities of development and testing have been performed on liquid methane fuel turbo-pump bearings working in liquid methane[6], supported by the realization of a dedicated test facility in Italy.

In the frame of international cooperation with JAXA, ASI is supporting further activities specifically dedicated to investigate the Methane thermal behavior, characterize bearings working in liquid methane, design and test of a regenerative thrust chamber in the 100-kN (10-tonne) class which is to be tested in Italy.

Another initiative investigated is the design of small methane thrusters to be applied as a potential reaction control system of the launcher stage, eventually integrated with a primary propulsive system; a preliminary test campaign has successfully been completed.

The Italian Aerospace Research Center, CIRA, is developing the 'Hyprob' research program, specifically dedicated to combustion phenomena studies and breadboard testing, up to the design of a medium scale (30-kN (3-tonne thrust class) regenerative thrust chamber; the program includes the realization of test facilities at both laboratory level and thrust chamber assembly (up to 100-kN (10-tonne class).[7]

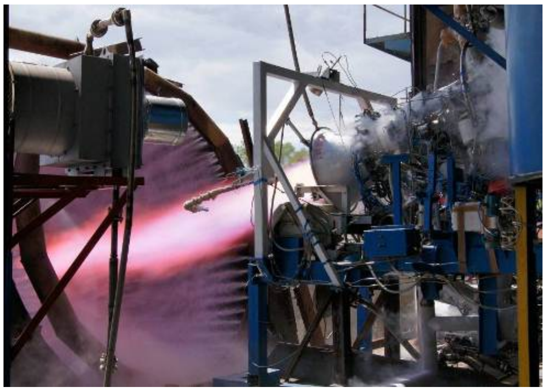


FIGURE 6 MIRA ENGINE DEMONSTRATOR TESTED MAY 2014

FIGURE 7 MIRA CH4 TURBINE MANIFOLD PRODUCED BY DMLS TECHNIQUES

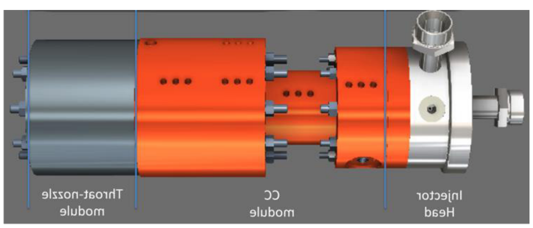


FIGURE 8 SINGLE INJECTOR RESEARCH THRUST CHAMBER

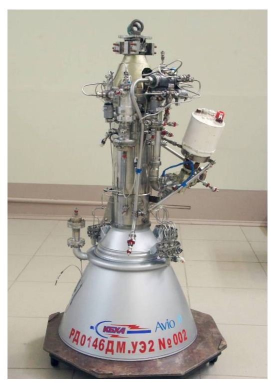


FIGURE 9 MIRA THRUST CHAMBER

CNES

For CNES, systems analysis of LOx/Methane propulsion is a critical gap area that has to be addressed. CNES regularly trades LOx/Methane propulsion with other alternatives (LOx/ LH_2, non-toxic storable propellants, etc.) to identify the best choices for each mission under consideration. Optimization is taken into consideration for an overall stage architecture (masses, volumes, technical performance as well as financial considerations). More generally speaking, opportunities for LOx/Methane propulsion devices are studied through an overall system analysis, including topics such as:

- Quality and availability of methane propellant
- Type of pressurization
- Ranges of operating domains for the engines
- Requested operations for re-ignitions capabilities

Relative to technology, no complete engine has been developed under a CNES program to date. Nevertheless, engine tests have been achieved on KVD1 Russian engine during a French – Russian cooperation.[8] Engine studies (Phase A studies) have also been performed at a high level of thrust (2000 kN (200 t)) with Russian colleagues, but no pieces were manufactured at this scale.[9]

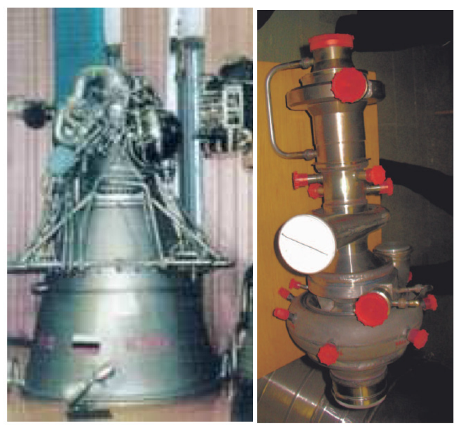


FIGURE 10 KVD1 ENGINE WITH ITS MODIFIED TURBO PUMP FOR TESTS

Research & Development activities are also being performed in parallel with several designs, manufacturing and tests at subsystem level (combustion tests and simulation capabilities including high-frequency (HF) instability analysis, pump and inducer performances, for example). [10]

Current French capabilities are also being developed for cryogenic propellants management in tanks during various dynamic and thermal phases with a resulting capability to simulate, predict and analyze propellant behavior under anticipated conditions.

The current main objective for CNES with French industrialists support is to prepare a LOx/ Methane low cost, gas-generator engine demonstration at 1000-kN (100-t) thrust level before 2023.[11]

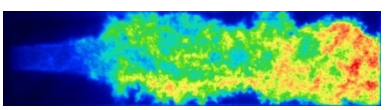
Before this phase, investigations at the engine level will be performed with a technological maturation platform (10-kN scale – bleed expander cycle). These evaluations are in progress, with ignition tests by the end of 2015 as an initial planned step. Moreover current LOx/LH_2 engine capability to operate with LOx/Methane is also being addressed.

DLR

A few launcher system studies have been performed at DLR in the past on reusable and expendable stages using LOx/-Methane propulsion. A partial reusable system with Liquid Fly-Back Booster (LFBB) has been preliminarily designed for both methane and kerosene as propellants in combination with LOx.[12] A high performance staged combustion rocket engine has been conceptually designed to reach a comparable performance level to the Russian RD-180 motor of Energomash. The study showed that the advantage of a higher energetic content of methane was counterbalanced by an increased motor mass and an increased booster size, hence higher aerodynamic drag and increased mass. The payload performances of the reusable kerosene and methane booster were found almost identical with some edge for kerosene[21].

Experimental and numerical research activities

Experimental research activities conducted at DLR have focused on the combustion and ignition of LOX/CH4 at both a sub-scale for higher thrust engines as well as a full-scale for RCS thrusters. The table below summarizes these activities and main findings.

Research Conducted	
Flame visualisation[13]	
	• In optically accessible chambers OH* and CH* visualisation was performed for sub and super-critical pressures • Optical sensors were implemented for stability investigations on full-scale (has been shown to be relevant for LOX/H2 combustion stability analysis, sensor performance qualified in high pressure LOX/H2-combustors). 
Injector behaviour[14,15]	
	• Flame stabilisation of coaxial and porous injectors was investigated • No issues encountered so far with respect to faceplate cooling, though no dedicated instrumentation was implemented. • Combustion efficiency investigations are planned for 2016.
Combustion stability[16]	
	• With a single coax injector some Low frequency instability (LF) was encountered when using CH4-film for cooling of windows for optical access. LF with CH4-film for window cooling

	(very fuel ROF for total mass flow). With H2-film for window cooling no LF-problem, stable combustion. • No dedicated high frequency instability (HF) investigations have been performed to date for LOX/CH4, however, no HF has been encountered in the LOX/CH4 test campaigns performed so far.
Ignition of LOX/CH4 multi-injector configurations[17]	

	• Ignition has thus far been performed with either
	    • Chemical igniters (LOX/H2 flame)
	    • Laser ignition
	
	• A fine tuning of ignition sequence required for the main combustion chamber (MCC) with a torch igniter. This is assumed to be consequence from narrower interval of ignitable mixture ratio (MR) as compared to O2/H2.
	
	• For pre-burner (PB) and gas generator (GG) significantly higher power was required for torch igniter than for MCC (ox-rich ROF during ignition transient; limited ignition reliability at fuel rich conditions)
	
	• With laser ignition 55 successful ignitions in a row were achieved using a directly mounted miniaturized laser. Sequencing and the mixture ratio are fundamental in smooth ignition.

High-altitude ignition of LOX/GCH4[18,19,20]	
	• Laser ignition of a full-scale 200-400 N RCS chamber was performed to determine the minimal ignition energy and demonstrate the feasibility of laser ignition. • Pre-ignition flow conditions in the chamber were visualised. Phenomena observed included supersonic gaseous flow with a barrel flow and Mach shock and LOX flashing.
LOX/Methane pre-burner applications	
	• Combustion was performed for low mixture ratio and injection temperature • Regenerative cooling with methane • Film cooling with methane was performed • No sooting issues experienced up to now. No thermal radiation in plumes was visible and no deposition on chamber walls was observed.

ESA

<u>LOX/methane propulsion at ESA Launchers - Synthesis for LOx-CH_4 Working Group Report @ISECG</u>:

LOx/Methane propulsion has been considered for ESA launchers since the mid-2000's as a potential path to reducing the cost of access to space. Launcher system studies were carried out, mostly limited to the concepts and performance rough estimates. LOx/Methane objectives have been embedded on certain engine technology demonstrators, often in dual use with LOx/LH_2, mainly for the characterisation of combustion phenomena.

In the meantime, significant activities have been developed in LOx/CH_4 propulsion by the national space agencies of several important Member States of ESA, mainly Germany, France and Italy. Most of these projects have involved at times cooperation with a partner outside Europe.

In the wake of the decision to develop Ariane 6 based on LOx/LH_2 and solid propulsion, ESA is evaluating for the longer term the LOx/CH_4 propulsion, along with reusability schemes, with the special objective of drastic cost reductions of access to space. This effort leans on some of the assets of the projects in Germany, France and Italy, and is closely aligned on the potential evolutions of the European launchers. The associated target engines are in the categories of a 1000-kN+ gas generator and a 100-kN expander cycle.

Another way to quickly address the need displayed in the ISECG Gap Assessment working group of 30-kN propulsion could be to adapt the existing Aestus engine (currently an Ariane 5 storable bi-propellant engine) to LOx/CH_4. This pressure-fed engine would offer the advantage of simplicity compared to pump–fed engines. However it has to be noted that this option is not being worked out currently at ESA and would require the support of certain Member State national space agencies. If the mission needs were more in the direction of large propellant loadings, a pump-fed engine may be more appropriate. In that case the options would be around a LOx/CH_4 expander-cycle engine, probably derived from a current LOx/LH_2 engine demonstrator project and certain Member States national programs.

JAXA

The research and development of liquid natural gas (LNG) engines have been carried out in Japan and several engines have been designed, manufactured and tested. Table 2 shows the summary of LNG engines in Japan.

TABLE 2 SUMMARY OF LNG ENGINES IN JAPAN

	LE-8 engine	30 kN-class engine	IHI in-house engine
			

Thrust(Vacuum)(kN)	107	30	98.0
Isp(Vacuum)(sec)	314	335	354
Combustion chamber pressure(Pc)(MPa)	1.2	1.2	5.2
Mixture ratio(Thrust chamber)	3.2	3.0	3.5
Chamber cooling	Ablative	Ablative	Regenerative
Nozzle expansion ratio	42	49	150

A 100-kN class LNG rocket engine, named LE-8[21], had been developed by Japan Aerospace Exploration Agency (JAXA) and IHI Aerospace Co., Ltd. (IA) till 2009 as the second stage engine of GX rocket and successfully completed more than 2000 seconds of its firing tests. Through the development of LE-8, the knowledge for the design of LNG rocket engines was accumulated and the feasibility of LNG rocket engines was confirmed.

Although almost all the technical issues of LE-8 engine had been checked and solved during the development period, the performance of the LNG engine was not confirmed under a flight condition, i.e. vacuum condition. After the development of LE-8, JAXA and IA started a research of an LNG engine for the purpose of obtaining performance data with a high altitude test stand (HATS). The thrust level was selected to be 30 kN for an experimental engine in the research, because a 30-kN class engine was considered useful for various future spacecraft and its small size was suitable for the HATS. Five firing tests with a total of 122 seconds were carried out and the functions and performance of the engine were confirmed under high altitude conditions[22,23].

The LE-8 and the 30-kN class engines consist of an ablative chamber and a liquid-liquid impinging type injector, making the engine system simpler and reducing the cost. Although the value of I_{sp} has increased as a result of the improvement on the design of chamber and injector, it seems to have reached close to the upper limit. Aimed at improving the combustion efficiency drastically, another LNG engine which has a regenerative cooling chamber was designed and demonstrated in a ground test facility in parallel with the HATS test. The activity had been carried out by the IHI Corporation (IHI) as in-house research program. The test series of the engine was successfully completed and its I_{sp} reached to approximately 350 sec.[24,25]

Although JAXA has been carrying out the R&D activities on LNG engines, Japanese LNG engines have not been used for an actual flight. The reason is that the performance and characteristics of current LNG rocket engines do not have enough advantages compared with other liquid rocket engines. Therefore, JAXA has decided to improve the performance of Japanese LNG engines drastically and is planning to apply the engine to future space transportation systems (e.g. a reusable liquid rocket booster and an orbital transfer vehicle). Currently, to achieve higher performance, JAXA is carrying out a research activity on LNG engines focusing on a regenerative cooling type engine. For example, JAXA is planning single and multi-element firing test, heat transfer measurement test for tube flow, etc.[26].

NASA

NASA has conducted system level propellant trade studies that identified LOx/Methane as a top in-space propellant for human spacecraft.[27] LOx/Methane supports in-situ production of propellants on the Mars surface for ascent vehicles.[28] NASA has been focused on the development of technologies at the system and component level for LOx/Methane cryogenic propellant management, primarily through technology projects such as the Propulsion and Cryogenic Advanced Development (PCAD) and Cryogenic Fluid Management (CFM) since the mid 2000's.[29]

At the integrated system level, under the PCAD project, NASA has conducted integrated testing of a cryogenic feed system, RCS engines, and main engine for a pressure-fed LOx/LCH_4 systems at altitude using a heavy weight vacuum jacketed tank and lines. From 2011-2014, the Advance Exploration Systems project conducted terrestrial flight testing of an integrated LOx/Methane system test bed on the Morpheus Lander. The Morpheus system consisted of blowdown helium pressurized aerogel insulated aluminum tanks, an integrated reaction control system (RCS) and a throttling main engine. The terrestrial flights demonstrated a Lox/Methane system integrated with guidance navigation and control (GNC).

Because of the cryogenic propellant temperatures, more helium is required to pressurize cryogenic propellant tanks. For a lightweight pressurization system, GHe is stored at high density and then warmed for pressurization. The Apollo Lander used cold, supercritical He storage and a fuel heat exchanger (HEX). Launch vehicles currently may use storage of GHe tanks within the propellants tanks to improve storage density. NASA has recently conducted tests of a carbon overwrapped pressure vessel (COPV) at cryogenic temperatures for use in a cold, high-density GHe pressurization system.[30]

In 2009 under the CFM project, a 1.2 m diameter spherical tank was used to demonstrate insulation and fluid performance in various environments associated with transit to a lunar or Martian outpost[31b]. More recently, a zero-boil-off (ZBO) system for liquid oxygen was demonstrated using a 1.2 m x 1.5 m cylindrical tank and a flight-like cryocooler[32]. Cryocoolers up to ~20 W at 90 K have been demonstrated in ZBO mode. Larger scale cryocooler development is in work with up to an order of magnitude increase in cooling power. Investigations into the control of pressure of sub-cooled liquid methane revealed that extreme care must be taken to achieve cooling through Joule-Thompson devices due to the formation of meta-stable conditions that can be achieved.[33] Liquid acquisition device bubble point and flow testing and characterization were accomplished. [34,35,36] The Radio Frequency Mass Gauge (RFMG) was developed and demonstrated as a mass gauge in micro-gravity for various propellants, showing promising accuracy.[37],[38] Vacuum chamber testing of cryogenic feed systems[39] with multiple RCS engines and single main engine have been conducted. On the Morpheus vehicle, a Joule-Thompson device was used to pre-chill the transfer lines of the auxiliary control propulsion system.[40]

Currently, the Evolvable Cryogenics project is developing a flight rated version of the Radio Frequency Mass Gauge for inclusion in the cryogenic subsystem of the Robotic Refueling Mission 3.[41] Work is on-going in the testing and development of flow boiling correlations for heat transfer in hydrogen, oxygen, and methane. Several other topics within the Evolvable Cryogenics project, while using a hydrogen boundary, will also be useful for oxygen and methane systems. These include the development of Multilayer Insulation (MLI) design details (such as what is the best way to seam MLI thermally) and large scale applications of MLI. Other work is focusing on the liquefaction system for handling in-situ generated oxygen within lander propellant tanks. Much work on the modeling of cryogenic propellants and two-phase systems using both commercial and in-house codes continues. From the simplest multi-node models to complex computational fluid dynamics (CFD) analysis, various effects and couplings such as two phase flow and heat transfer mechanisms are being investigated computationally. To investigate various fluid mixing phenomena with cryogenic simulants, the Zero Boil-off Transfer (ZBOT) payload is currently being developed for the ISS Microgravity Science Glovebox[42]. For slosh damping, tests have been conducted on baffles in LOx/Methane propellant tanks.[43]

Several smaller funding sources (such as Small Business Innovative Research (SBIR) projects) are focusing on possible advances in the state of the art such as spherical cryogenic rated composite vessels, lightweight vacuum jackets, and advanced soft vacuum insulation systems. Methods of reducing structural loads are being looked at from several angles such as vapor cooling and sacrificial structures. However, for long duration storage systems, it is key that thermal problems be considered in the design of the stage structure; this will lower the heat load into the propellant significantly reducing cryocooler lift (and mass) requirements.

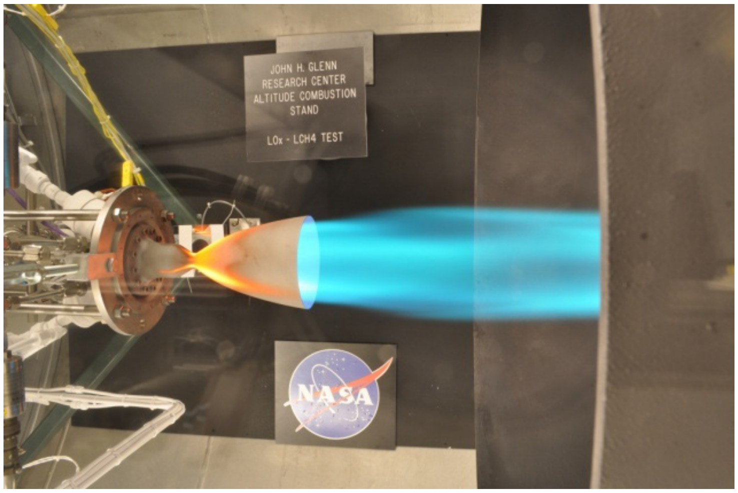


FIGURE 11 AEROJET 100-LBF LOX/LCH4 REACTION CONTROL ENGINE IN ALTITUDE TEST AT GRC.

NASA has conducted tests for LOx/LCH_4 RCS and main engines. RCS engines at thrust levels of 88 N (20 lb_f), 444 N (100 lb_f), and 3.8 kN (870 lb_f) have been tested over a range of gas-gas to liquid-liquid conditions. [44],[45],[46] Both ablative and regeneratively cooled RCS engines were successfully design and tested. A Pressure-fed 24.5-kN (5500-lbf) film-cooled/ablative engine for lunar ascent has also been tested at vacuum conditions. Tests with a 129:1 ratio nozzle demonstrated performance values that, when extrapolated for a 150:1 ratio nozzle, were within 2% of target 355 s Isp. Pressure-fed LOx/Methane engines, both 24-kN (5400 lb_f) and 8.8-kN (2000 lb_f) scales, with ablative and film-cooled chambers capable of throttling, were operated on the Morpheus test bed at sea level.[47],[48] Some limited combustion chamber cooling tests using heated tubes and engine tests have been conducted for regeneratively-cooled engines. Injectors designed for use in pump-fed engines for LOx/Methane, such as gaseous methane and liquid oxygen swirl coax, have also been conducted. [49,50]

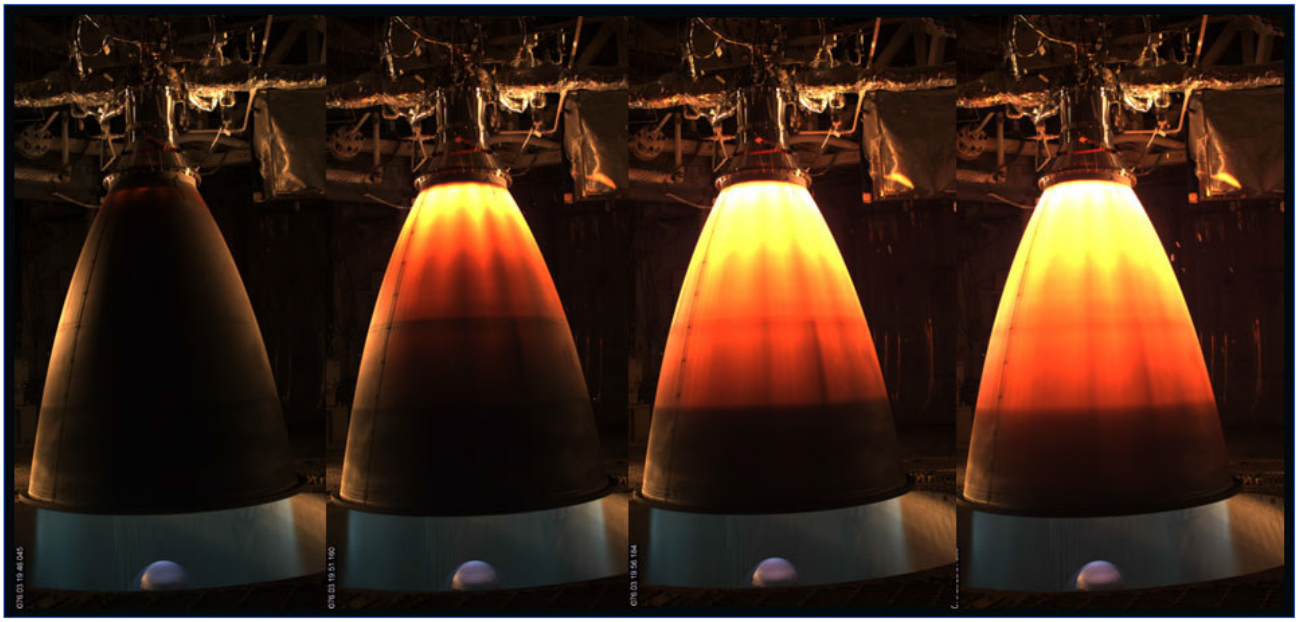


FIGURE 12 AEROJET LOX/LCH4 ASCENT MAIN ENGINE DURING ALTITUDE TESTING AT WSTF

NASA is continuing to build upon prior methane engine development experience. This includes testing of pressure-fed and pump-fed thruster and engine designs with the potential for scalability to achieve the approximately 88 kN to 155 kN (20,000 to 35,000 lb_f) of thrust, high performance, and throttling capability needed for larger descent/ascent landers on Mars. The total thrust required would be achieved by teaming engines to required performance commensurate with the final human scale lander. Currently, small-scale pressure fed (17-kN (4,000-lb_f)) and large-scale pump fed (111-kN (25,000-lb_f)) engine components are being tested, with the goal of performing integrated breadboard engine testing at both scales within the next year. The breadboard engines will provide a test bed capability for developing the component and system technology required to ultimately develop and certify a large methane engine for future human Mars Missions.

When used in combination with the primary propulsion for ascent, descent and transit, an integrated Lox/Methane RCS approach delivering on the order of 444 N to 4 kN (100 to 1000 lbf) thrust at 325 seconds I_{sp} supports the mission architecture goals for common engine development and compatibility.

Within NASA advanced additive manufacturing techniques are being applied to thruster assembly and component technology development to improve performance, reduce fabrication processes and compress schedule. Additionally, the application of additive manufacturing has allowed engineers to more readily incorporate features that facilitate component testing and accelerate the learning curve on engine behavior.

GAP ASSESSMENT

The working group members identified the gaps by assessing the application needs and performance requirements for Human lunar landers and future Mars vehicles against the current status at each agency. The group identified the technology gaps that need to be addressed in future technology efforts. These gaps can either be worked on jointly in a partnership or the gaps can be spread-out among different agencies. In addition, some gaps such as main engine technology may require multiple agency partners to have the most effective results.

- **Develop a throttleable regeneratively cooled pump-fed and/or pressure-fed engines to address gap for throttling (5:1 – 10:1), 360-365 sec, and for regenerative cooled engines in the 30 – 100 kNKN range**. The rationale is that regenerative cooling provides a performance increase over fuel film cooling. For landers, throttling is required. The combination of these two requirements, as well as the application dependent need for both pressure-fed and pump-fed engines results in a significant gap.
- **Develop 100 to 220-N RCS thrusters with integrated cryogenic feed systems to addresses gap for thruster size/cost and then to evaluate GNC impulse bit and thrust requirements.** The rationale is that this smaller size thruster has not been fully developed and tested. As part of the RCS engine technology, system integration analysis using GNC vehicle models need to demonstrate that GNC impulse bit and thrust requirements can be met with LOx/LCH4 thrusters.
- **Develop long duration reliable cryogenic refrigeration systems capable of maintaining zero-boil-off and performing liquefaction of in-situ produced propellants (several hundred watts at ~90 K).** The rationale is that this size cryocooler is required for future Mars surface operations. The use of proving grounds will be critical to the demonstration of this technology.
- **Develop composite cryogenic tanks with focus on spherical geometry to addresses gap in propellant tank technology.** The rationale is that cylindrical cryogenic all-composite tanks have seen some development, but not all-composite cryogenic tanks in a spherical geometry. A spherical tank geometry offers advantages in the reduction of surface area for thermal storage improvements and ease of vacuum jacketing. Mass performance, helium permeability and off-gassing will be key performance factors.
- **Develop high performance pressurization systems that improve pressurant storage density and reduce mass to address gap for use with cryogenic propellants.** The rationale is that significant mass savings can be obtained by technologies such as the use of cold (high density) helium storage and heat exchangers to warm helium prior to propellant tank pressurization or the use of autogeneous pressurization, but that there is gap in the components and models for use with LOx/methane.
- **Conduct extended duration thermal vacuum testing of integrated system to address gap of integrated system testing in thermal vacuum environment.** The rationale is that component level tests, such as tanks with MLI, have been conducted in thermal vacuum to simulate space, however system level tests in a thermal vacuum environment have not been conducted. This would include GHe pressurization, propellant tanks, feed system, and engines.
- **Fly a zero-g cryogenic liquid acquisition experiment in space such as on the ISS or in a cis-lunar location to address gap of lack of demonstration of LOx/methane in these conditions**. This experiment would provide critical data, scaling, and model validation which can be used for the design of future vehicles.
- **Fly a test vehicle in space as a technology infusion mission to demonstrate integrated LOx/Methane propulsion systems to address gap of no in-space LOx/Methane flight experience**. The rationale is that lack of space flight experience is an impediment to program acceptance of the risk.

CONCLUSION

Lox/Methane (or Liquified Natural Gas) is an enabler for future exploration with in-situ propellant production, improved performance, and fluid commonality. The team identified common technology issues and gaps. All of the partners find value in continued investment through partnerships. The team hopes that technology implementers at each agency will be able to use this to help focus on critical needs.

APPRENDIX TABLE 3.

Lox/Methane Propulsion System Technology Areas	Performance needs for a Deep space based lander or other vehicles (service modules, tugs, etc)	Goals	Technology Solutions	Status	Gaps
Integrated System	High Reliability - several usages of the same vehicle, Low cost, HIgh delta-v, long life reusable, High mass fraction, quality and origin of methane	1. Reliability > 0.995 2. Life > 10 operating cycles 3. Mass Fraction > 0.85 (Prop dry+structure)	1. Design for re-usability	1. (NASA) Integrated technology prototype system level test of pressure-fed Lox/LCH4 systems at altitude (WSTF) and terrestrial flight testing of integrated Lox/Methane on Morpheus Lander	1. (CNES) Availability of low cost process, verification of materials compatibility 2. (CNES) Safety margins. Maintenance Repair and Overall after each usage 1. (NASA) integrated system prototype technology in thermal vacuum conditions 2. (NASA) Space zero-g flight testing
Pressurization System	High density light weight storage	50% reduction in volume and mass over ambient temperature storage	1. Ghe cold storage with heat exchanger 2. partial autogenous systems	1. (NASA) Apollo Lander used Lhe storage and HEX. Launch vehicles may also use this.	1. (CNES) Optimised pressurization device(s) to be studied from Earth ground up to end of the mission 1. (NASA) Ghe Heat exchangers, thermodynamic and thermal modeling
Pressure vessel	Lightweight	1. PV/W > [Tank Pressure*volume/Tank Mass]	1. Metallic (aluminum lithium) 2. Composite Overwrap 3. All Composite	1. (NASA) SOA existing materials for spherical/cylindrical vessels. 2. (NASA) spherical composite vessel development in work. Cylindrical vessels developed.	1. (NASA) Development of lightweight composite vessels near spherical
Thermal Management	low boiloff, thermodynamical management of propellants during launcher phase from Earth ground	1. deep space Zero boil-off (0 W/m2) storage at EML1 2. lunar surface surface heat leak 0.25 Watts/m2	1. Passive 2. active	1.(JAXA) Passive storage. Approx. 50 W/m2 in LOX/LH2 2nd stage vehicle. 2. (JAXA) ZBO(Active storage) Investigate cooling performance by elementary test level. 1. (CNES) Improvement of simulation code 1. (NASA) Structural solutions need to be designed with thermal results in mind. High-Performance MLI for in-space/lunar surface applications (especially smaller lander tanks) essentially ready (some engineering characterization of specific design needed). 2. (NASA) Cryocoolers up to ~ 20 W @ 90 K have been demonstrated in ZBO mode. Larger scale cryocooler development in work.	1. (CNES) On orbit demonstration 1. (NASA) Structural and deep space shielding solutions to meet zero watts passive thermal requirements. Insulation systems for inside Mars atmosphere. 2. (NASA) Cryocoolers greater than 20 W @ 90 K.

Lox/Methane Propulsion System Technology Areas	Performance needs for a Deep space based lander or other vehicles (service modules, tugs, etc)	Goals	Technology Solutions	Status	Gaps
Liquid Acquistion in zero/low g	zero-g start, refueling capability, slosh damping, and anti vortex	1. 2% residuals	1. Screens channel 2. Vanes 3. Sponges	1. (JAXA) 2.7% residuals restart was verified by simulation in Lox/LH2 2nd stage vehicle. 1. (NASA) Computer simulations, simulant ground testing, Bubble point and Screen channel tests with LOx and LCH4	1. (CNES) Minimizing propellant losses during refueling 1. (NASA) Demonstrate zero-g accurate cryogenic propellant liquid acquisition and gaging.
Feedsystem	Redundancy management, propellant distribution	1. Lightweight 2. Low Pressure drop 3. low heat leak	1. Cryogenic Feedsystem for LOx/LCH4 main engine and RCS	1. (NASA) Vacuum chamber testing of cryogenic feedsystems with multiple RCS engines and single main engine	1. (CNES) Minimizing propellant losses during chill down phase(s) 1. (NASA) Thermal vacuum chamber testing
Reaction Control Engines	Provide Min. Impulse Bit, Thrust, and high cycle life	1. Min Ibit TBD 2. Thrust Range 3. Cycle Life > TBD		1. (JAXA) Gox/GMethane RCS is investigated and would be tested in 2015 1. (IT) Lox/GMethane RCS demonstrator has be tested in 2014 1. (NASA) 25, 100, and 870 lbf RCS engines have been tested over a range of gas-gas to liquid-liquid conditions. Both ablative and regen cooled RCS engines	1. (JAXA) Pulse mode feasiblity 2. (JAXA) Life cycle demonstration 1. (NASA) 220N (50 lbf) RCS

Lox/Methane Propulsion System Technology Areas	Performance needs for a Deep space based lander or other vehicles (service modules, tugs, etc)	Goals	Technology Solutions	Status	Gaps
Main Engines	Throttle capability including idle mode, High Isp, High reliability, Thrust / Weight ratio	1. > 4:1 throttle depending on T/W 2. Isp > 355 sec 3. Thrust >30 KN 4. Helium free design	1. Pump-fed, 2. Pressure-fed Ablative 3. Pressure-fed regen	1. (JAXA) Throttling capability of methane engine have not been enough verified in the past JAXA's R&D activities. 2-1. (JAXA) The Isp of JAXA's pressure fed type methane engine reached 335 sec (using ablative cooling chamber, expansion ratio is 49, thrust value is 30 kN). 2-2. (JAXA) The Isp of IHI's in-house research engine reached 356 sec (using regenerative cooling chamber, expansion ratio is assumed 150, thrust value is 100 kN). The value is estimated from Sea level firing test result. 3. (JAXA) From 30 to 100kN engines were developed in past JAXA's R&D activities. 1. (CNES) Design and tests at subsystem level for potential elements for a LOX / methane engine 1. (ASI) MIRA Demonstrator 100kN class Pump fed Expander cycle engine tested in 2014, confirmed possibility to obtain high reliability, multi ignition capability without any maintenance, re-usability and possibility to obtain Isp 364s.Throttling capability down to 70% has been demonstrated. Deeper, low risk, throttling leads decreasing of the nominal pressure. 2. (ASI) From 20kN to 30kN pressure fed regenerative cooled combustion chamber have been investigeted. (ASI) pressure fed thrust chambers (30kN class) regenerative cooled under realization within Hyprob program (within 2017). 3. (ASI) ALM technologies for main components (TP, chamber) under investigation 1. (NASA) pump-fed engine type injector tests	1. (JAXA) Throttling capability. We have to assess the risk of deep throttling, such as chug stability limits, responsiveness. 2. (JAXA) Increasing of Isp. Isp have to be increased. To achieve that, regenerative cooling engine is preferred. 1. (NASA) Isp > 355. 2. (NASA) Pump-fed LOx/LCH4 engines with throttling 3. (NASA) Regen channel with throttling

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