ISRU HYDROGEN ENGINE FOR SUSTAINABLE PLANETARY EXPLORATION

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ABSTRACT:

Rocket Engineering develop low-cost propulsion technology to enable a UK or European lander to launch from the surface of the Moon, after refuelling using hydrogen and oxygen derived from ISRU, before the end of this decade. Knowledge transfer from partners in the UK & Europe including Steamology Motion, LENA Space, Cranfield University & Arceon, has enabled first steps towards a prototype gaseous hydrogen engine proof-of-concept. Initial work done on systems studies to evaluate design options including dense, gaseous pressurised hydrogen storage, ablative and boundary layer film cooled combustion chambers is described, Testing of engine components has begun and will continue through 2024.

1. RATIONALE FOR A REFUELLABLE PLANETARY DESCENT ENGINE

Large numbers of lunar surface missions are planned this decade and the next under the NASA lead Artemis exploration programme. These numbers are matched by a growing interest from many other nations with existing beyond-Earth-Orbit space capability, and some who are new to national space exploration, including the Middle East, South America and the UK. A Rocket Engineering survey for the UK Space Agency Commercial Spaceflight team in 2022 identified an average of 6 lunar missions a year through 2030, the majority being landers. NASA's 2023 budget request identified 22 Commercial Lunar Payload Service or CLPS surface missions through 2029 [1], two of which launched in 2023 and attempted landings in 2024. This number is expected to be matched by the lunar interest from the rest of the world, in a form of 'New Space Race'.

Despite advances in transportation systems and their propulsion, such as pump fed throttleable lander engines (Nammo Reliance) and cryogenic landing stages (Intuitive Machines IM-1), all lunar missions to date, and all planned small landers, are designed to be single use - utilising primarily solar-battery power, and constrained to one lunar day or fourteen Earth days of operation.

Given global mission plans, and the building of the Lunar Gateway space station starting in 2028; regular access between Earth and the moon is likely to become of increasing importance, in particular, the Lunar South Polar region.

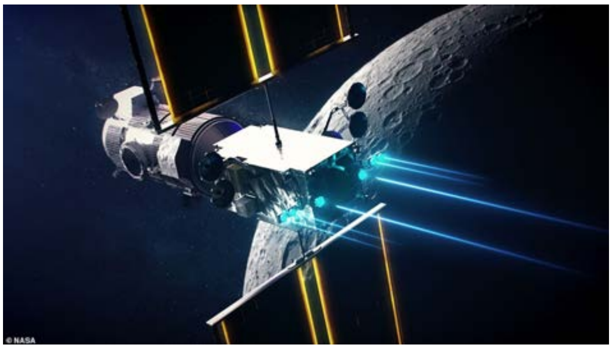

Figure 1. Artists rendering of Lunar Gateway Space Station and its orbit maintenance electric propulsion (Image credit: NASA).

An ability to regularly and reliably transport goods between Gateway in its elliptical, lunar South Pole focused orbit, and locations on the lunar surface will support the long-term surface presence planned for humans as part of Global Exploration plans. The Global Exploration Roadmap (GER) [2] produced by the ISECG in 2018, identified: '*reusable lunar landers: shifting cost from developing recurring units to enabling other elements such as an in-space refuelling infrastructure…' as critical for lunar surface exploration and noted 'affordability' as a key sustainability principle.* A 2022 update [2] highlighted a key Lunar Surface Exploration Scenario Objective as *'establish[ing] regular access to and from the lunar surface'*. The same ISECG in its 2019 Global Exploration Roadmap Critical Technology Needs report [3] identified, *inter alia*, cryogenic propulsion, zero boil-off technologies (for cryogenics, in particular hydrogen) and lunar ISRU technologies which *'advance the TRL for Mars missions'* as of interest, but did not explicitly identify the ability to refuel propulsion on the lunar surface as critical technology. This is despite the value of other transport applications including Lunar / Earth return, and Lunar point-to-point transport, and Gateway shuttle services, in supporting sustainable exploration at all lunar latitudes, including the far side and providing building blocks for Mars scenario transportation.

2. CHALLENGES OF HYDROGEN AS A PLANETARY EXPLORATION PROPELLANT

Hydrogen-Oxygen propulsion systems are of immense value for space transportation because of (a) their higher Isp performance, compared to all other chemical propulsion; (b) their 'green' or low carbon exhaust credentials; and most importantly (c) the potential for both propellants to be sourced from (electrolysed) water / ice known to be available throughout the solar system and in particular in shadowed lunar polar craters. The latter is termed ISRU or In Situ Resource Utilisation, and is acknowledged as an essential route to long-term sustainable space exploration goals in keeping with the GER aim [2], while also supporting science and commercial development.

If space transportation can remove its dependancy on propellant resources shipped at great cost from the surface of the Earth, the affordability and hence sustainability of space exploration make a step change upwards. Hydrogen-oxygen propulsion technology is therefore able to support a long-term lunar economy, more effectively than any other propulsion. Given that no hydrogen-oxygen rocket engines are commercially available outside a few ultra-high thrust high performance launcher applications, all using liquid hydrogen, there is an opportunity to offer a new, small engine system product to exploration users. This would support UK technology contributions into critical space exploration technologies [3] and would be of interest to commercial lunar lander systems integrators.

All propulsion used for lunar surface access to date has used non-renewable propellants, typically employing hydrazine storable propellant combinations, and most recently LOX-CH4 used by the IM-1 / Nova-C / Odysseus lunar lander [4].

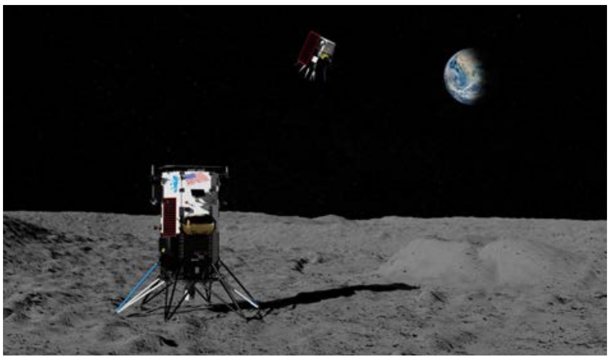

Figure 2. Intuitive Machines NOVA-C lander and micro-NOVA 'hopper' artists rendering (Image credit: Intuitive Machines).

Except for a single SLS-Block 1 launch in 2022, hydrogen has not been used in deep space since the Apollo era, and only then as a fuel cell reactant. Liquid hydrogen, LH2 was also stored on the LEO operating space shuttle or STS, having a maximum duration of 30 days and in practise no longer than 17 days for the record breaking STS-80. Limited LH2 usage in space stems principally from difficulty in maintaining hydrogen in its liquid state, around 20K or lower temperature, without considerable energy expenditure - generally considered non-viable in the low energy density, large solar powered space environment. Blue Origin have made a number of statements about their commitment to developing zero boil-off technology [5] for its large, human rated lunar lander for NASA, noting *'We want to make hydrogen a storable propellant because we believe that not only does that open permanency on the Moon… it enables the rest of the solar system…'.* However, this fails to mention the challenge of obtaining liquid hydrogen on the lunar surface, except by prior shipment from Earth, which fails the GER sustainability principle. Production of propellant in-situ, in this case LH2 on the lunar surface is considered extremely unlikely in the next several decades, not least due to the energy requirements. Liquefying hydrogen from a pure source of the gas requires expenditure of over 1/3 of its calorific energy content, around 10-13kWh/kg compared to 33kWh/kg respectively [6], requiring application of a number of technologies such as cryocooling on a large scale which are not well developed for space, and in particular on the lunar surface.

It is likely that the abundant resources of water ice identified in cold traps on the Moon, Mars and elsewhere in the solar system will become a refuelling resource. Robust autonomous chemical processing technology will in the next decade be deployed to the lunar and other planetary surfaces, allowing water to be extracted, purified and them processes to hydrogen and oxygen. Although hydrogen technology, particularly liquefaction and long term storage presents some of the most challenging processes, its mastery is key to space based ISRU or In Situ Resource Utilisation refuelling capabilities. At present, LH2 is not produced in the UK, so opportunities to carry out research and development on LH2 based space propulsion are very limited. Although this is set to change, driven in part by commercial interest hydrogen based aviation [7], gaseous hydrogen and oxygen (possibly in liquid form) need to be considered in the near term, to benefit from the attractive Isp of this propellant combination. However this poses the question whether sufficient gaseous hydrogen propellant be stored to allow delivery of the large impulses needed for high deltaV manoeuvres, without a punitive inert mass penalty due to the perceived bulk of a gaseous propellant?

This paper starts to explore challenges for the UK, in contributing to space transport requirements of the global exploration roadmap, namely:

1) Can UK hydrogen-oxygen rocket engine expertise, absent on any scale for over 50 years, be rebuilt?
2) Commercially useful approaches to hydrogen based space propulsion, avoiding use of liquid hydrogen that is unavailable to developers in the UK?
3) Elaborating approaches to store, long term hydrogen at densities which would be useful specifically in a small lunar transport system.

Approaches to (1) and (2) are outlined in this paper. (3) is the topic of ongoing research and collaboration and will be reported on in due course.

3. PROPULSION DESIGN APPROACH

a. System level

Preliminary analysis considered a small lunar lander using publicly available parameters from the iSpace Hakuto-R M1 lander as a reference system design, to explore the design trade space. Key system parameters were: Dry mass ~340kg, Wet mass ~1000kg, Payload 30kg, main body volume in form of an octagonal prism, 1.64m high $\times$ 1.6m across widest diameter. Propulsion used a central deceleration engine and six further smaller attitude control engines, as shown below:

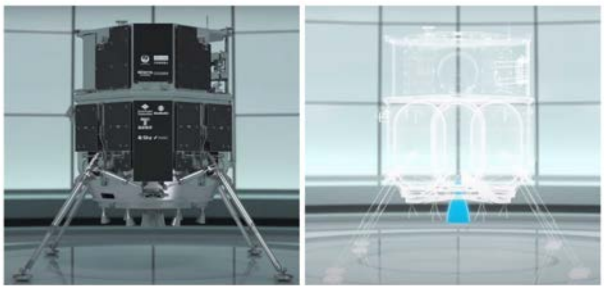

Figure 3. Hakuto-R M1 lander cutaway view (Image Credit: iSpace)

Propellant storage and resulting mass were assumed to drive feasibility. A number of system options were explored, based around different forms of propellant storage, considering cooling / storage / overall performance factors, all propellant combinations contained lunar derived hydrogen and oxygen. A constraint derived from the lander system parameters was a maximum propellant tankage volume limit of 0.68m^3. A conservative DeltaV value based on landing, not liftoff, was assumed to be 2.74km/s. The minimum engine thrust was estimated from the lander dry mass plus payload (assuming that all propellant tanks were empty), under one lunar gravity, with a minimum required thrust:weight of one. A minimum thrust of 600N can then be estimated, or <1000N to include some margin and to allow for lander system growth, and throttling. This is in the same thrust class as European liquid apogee engines LAEs such as the LEROS. An engine Isp performance of 4438m/s based on a chamber pressure of 1.0MPa, an expansion ratio of 115 and an O:F of 4.2:1 was assumed.

The table below compares three of the multiple options for propellants and storage options, against the requirement. The options include (1) ultra-high pressure gaseous propellants, noting engineering safety issues related to ultra-high pressure and oxidising gases (2) liquid hydrogen and gaseous oxygen, to understand what could be achieved as a baseline reference point and (3) moving to a liquid High Test Peroxide or HTP oxidiser with GH2, which could, in principle be produced on the Moon, at the expense of a greatly reduced Isp performance (estimated as 3250m/s at the minimum oxidiser + fuel volume mixture ratio of 20:1).

Table 1: System impact of fuel / oxidiser storage options

	Concept	Propellant volume	Lander wet mass
	REQUIREMENT	**<0.68m^3**	**<1000kg**
1	GH2, GO2 @ 100MPa pressure	1.31m^3	630kg
2	LH2, and GO2 @100MPa pressure	0.89m^3	641kg
3	GH2 at 100MPa and HTP oxidiser.	0.76m^3	790kg

None of the above combinations meet the lander volume requirement, although being able to store oxidiser or fuel at liquid density levels allows the closer match to the requirement. Moving away from gaseous oxygen to less energetic oxidisers such as HTP greatly increases lander mass, and while meeting the requirement, may have undesirable mission level impacts. HTP production in the lunar scenario also poses an unknown technical risk, compared to the production and pressurisation of gaseous oxygen and hydrogen, and like LH2 production is likely to take place well after 2030.

Rocket Engineering are exploring methods to use the lunar regolith temperature of around 100K in shadow to increase the density of hydrogen in gaseous form without resorting to the extreme pressures of 1000Bar needed to stored hydrogen efficiently in gaseous form, which poses a number of other engineering difficulties. In the limit, a density approaching that of liquid hydrogen, 70kg/m^3 may be practical. Assuming oxygen can be stored in liquid form for extended periods on the lunar surface, which is supported by the move to LOX / CH4 propulsion by IM-1 [4], a propellant volume for this reference mission of around 200% of the design requirement may be achievable. Although this would mean a considerable increase in the propellant tank and hence lander body volume, the benefit of the increased Isp performance of hydrogen-oxygen propulsion and its ability to be refuelled make additional volume less of a disadvantage. *Further trade studies are ongoing, to understand how the changes in propulsion system volume, for example increased propellant tank mass, will impact on overall system mass.*

This preliminary trade study suggests that there is merit in exploring the feasibility of a lunar surface refuellable gaseous hydrogen / oxygen propulsion system, provided that it can make use of higher density, lower temperature (around the 100K range) compressed hydrogen in combination with liquid oxygen. On this basis a conceptual propulsion system design has been created.

b. Propulsion system

A spacecraft propulsion system, using hydrogen or any other propellant requires development or procurement of the following elements:

a) Thrust Chamber Assembly, TCA (consisting of combustion chamber and expansion nozzle)

b) Ignition mechanism if propellants are not hypergolic.

c) Propellant Feed Assembly, PFA (consisting of a distribution manifold, injectors, acoustic cavities)

d) Valve assembly (consisting of propellant valves, filters and trim orifices)

e) Engine hard mounts and, if required, a thrust vectoring system

f) Radiation shield to minimise radiant heating effects on host spacecraft.

g) Upstream propellant storage and management system (including low gravity propellant management, pressurisation and additional control valves)

Proof-of-concept for propulsion using dense, cryogenic, but gaseous hydrogen requires consideration of at least the first three, in particular TCA, ignition and PFA.

Given an engine length constraint dictated by the likely clearance under a typical lander of 600mm, and an estimate chamber length of 100mm, the expansion nozzle length is restricted to 500mm. An engine sized to deliver 1.5kN thrust in vacuum, operating at a P_c of 1MPa is shown below:

Figure 4. Approximate sizing for H2/O2 lunar lander concept design engine

After sizing of the thrust chamber, the key design trade-off is the cooling mechanism needed to sustain the heat flux from hydrogen-oxygen combustion, with a flame temperature of at least 3000K. Targeting as high an operating chamber pressure as possible to keep the expansion nozzle small increases chanced of fitting within lander envelope constraints. The approach is to keep the overall Mixture Ratio, MR fixed at its optimum, reduce the MR near the walls and increase the MR in the core of the chamber to maintain a cooler burning environment neat the chamber walls and reduce the requirements for the chamber materials. Conditions near the walls could be further improved with the use of a boundary layer cooling technique. Preliminary heat transfer analysis conducted at the throat, has indicated 20wt% hydrogen film cooling would maintain the throat at ~1900degC. Gas film cooling while less effective than liquid film cooling employed in engines such as the LEROS, would also have less impact on the engine Isp.

The lowest cost design approach compatible with a refuellable, reusable system is for chamber and nozzle to employ an ablative liner, a thin layer of insulation, and a high-strength outer shell. A bolt-up thrust chamber assembly is proposed as ablative inserts need holding in place from both axial positions. The engine concept design shown below includes a Ti-6Al-4V nozzle extension welded to the metallic chamber shell and nozzle. This design could achieve a theoretical vacuum specific impulse, assuming reasonable losses, of 4423m/s.

Figure 5. Bolt up ablative engine design with Titanium expansion cone.

The image below shows a longitudinal section of concept engine including ablative inserts and a throat film cooling manifold (in addition to film cooling introduced at the injector). A sea level test version of this engine is being constructed for hot fire testing.

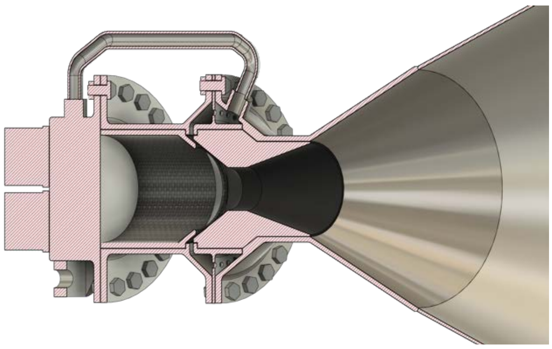

Figure 6. Section view of low cost lunar lander engine (injector details omitted)

Provision for an ignition system based on commercial-off-the-shelf or COTS spark or glow-plug, or a small, low flow hydrogen-oxygen gas torch using a bypass feed has been allowed.

4. EARLY PROTOTYPING & TESTING

Rocket Engineering have begun prototyping a number of testbed items to allow proof-of-concept testing of its low-cost lunar lander engine design. These include:

a) **Mk 0 manufacturing testbed chamber:** Trialling additive layer manufacturing, water cooled to allow robust evaluation of throat film and injector film cooling approaches. This engine chamber is shown below along with an image from a hot fire ignition test carried out at Snowdonia spaceport with the assistance of Newton Launch Systems Limited.

Figure 7. Mark 0 water cooled monolithic engine chamber manufactured using ALM in Inconel 718.

Figure 8. Hot fire ignition test of Mark 0 engine chamber at Snowdonia Aerospace Centre

Further testing of this chamber with a modified fuel film cooled injector is planned in 2024.

b) **Mk 1 ablative chamber, outer shell:** Inconel 718 chamber shell segments have been manufactured using additive layer manufacture, to support sections of an ablative liner to be procured form a European partner. Chamber sections following removal from the build plate are shown below. additional valve mounting and flange plate interface sections are not shown. A number of support struts to facilitate ALM are presented in the parts.

Figure 8. Mark 1 chamber pressure shell parts in Inconel 718 (throat fuel film cooling manifold not implemented)

c) **Ablative chamber, liner material:** An ultra-high temperature, oxidation resistant material thermo-mechanically compatible with the Inconel engine pressure shell is needed. Although many of the engine design principles are common to those for liquid spacecraft apogee engines, the predicted throat temperature of close to 2200K in this design is well outside the thermal limit for Hitemco coated Niobium alloy chambers typical of Nammo or Aerrojet LAE's. Rocket Engineering are exploring the use of a carbon precursor, SiC-SiC ablative material 'ArSiC' manufactured using a pyrolysis and liquid infusion process patented by Arceon BV in the Netherlands.

d)

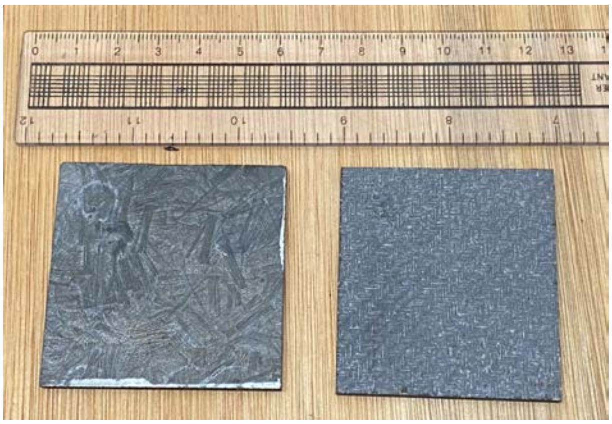

Figure 9. ArSiC samples provided by Arceon, NL

ArSiC is a lightweight high performance composite able to withstand temperatures above 1200ºC in a long-term oxidising environment, comprising composed of SiC fibres embedded in SiC matrix. A prototype chamber liner is being manufactured with a view to testing this towards the end of Q2, 2024.

5. SAFETY AND FLUID CONTROL

Considerable strides have been made toward design and procurement of a small portable hydrogen feed system able to deliver a flow of hydrogen at up to 0.05kg/s and oxygen at up to 0.18kg/s. This is designed to be trailer mounted, able to support ignition and short firing tests able to reach steady state operation of the combustion chamber to establish validity of the thermal design. An example of a small trailer mounted feed system of this type built by Rocket Engineering for a commercial client is shown below.

Figure 10. Portable engine test rig (hybrid rocket example). (Image Credit: Pulsar Fusion)

High pressure oxygen and hydrogen safety are of paramount importance when constructing this type of feed system, and Rocket Engineering are pleased to have the support of both Steamology Motion Limited and LENA Space who are advising on safety and providing access to clean assembly facilities for this development.

6. COLLABORATION BENEFITS AND ACKNOWLEDGMENTS

Rocket Engineering is fortunate to have benefited with UK Government grant funding from the following sources

i. Innovate UK Edge, RTO Research and Technology Organisation - Catapult access fund for facility usage (supporting ALM of Inconel parts, at the Westcott Venturer Park)
ii. STFC Campus Cross-Cluster Proof of Concept Grant Call - Connecting Space Clusters Grant number POC23-27.

In addition Rocket Engineering are extremely grateful for collaborative knowledge sharing from the following organisations:

Steamology Motion Limited, Dean Hill Park, Salisbury UK	Feed system design, procurement and safety consultancy
LENA Space, Dean Hill Park, Salisbury UK	Vacuum ignitor design reference information and supply chain consultancy.
Newton Launch Systems at the Snowdonia Aerospace Centre, Llanbedr, UK	Hot fire ignition test support at Snowdonia Spaceport, Wales.
Protolaunch, Westcott Venture Park, UK	O2 / H2 propulsion , feed system and chamber design discussions, potential future vacuum test support
Arceon, Delft, The Netherlands	Provision of samples and composite material liners for combustion chambers
Sheffield Hallam University, Sheffield, S Yorks, UK	Materials advice on high temperature materials, special thanks to Dr Hywel Jones.
Cranfield University, Cranfield, Beds, UK	Astronautics & Space Engineering MSc Project support. Advice on hydrogen fluid feed components.

Rocket Engineering would also like to thank Mr Richard Dinan, CEO of Pulsar Fusion for providing the inspiration for this development.

7. KEY FINDINGS AND NEXT STEPS

Rocket Engineering's goal is to develop and qualify low-cost propulsion technology to enable a European lander to launch from the surface of the Moon after refuelling using hydrogen and oxygen derived from ISRU, and to do so before the end of this decade. In the six months between October 2023 and March 2024, initial steps have been taken towards this goal.

1. Systems study activities have ascertained that compressed, cooled hydrogen gas and liquid oxygen, which could be processed on the lunar surface using near term ISRU technology, may be a competitive means of storing propellants for launch, and descent of small landers. However there is considerable further work required to rebuild high thrust hydrogen-oxygen propulsion know-how in the UK since it was lost in the late 1960s [8].
2. The Isp performance benefit of hydrogen-oxygen propulsion, and its unique ability to be refuelled using entirely indigenous propellants (on the Moon and many other solar system bodies), are likely to considerably outweigh the inert mass tankage penalty in storing hydrogen, particularly where compressed hydrogen gas and the heat sink provided by the lunar regolith are used to support densification of the fuel.
3. A key next step for the UK is to develop, successfully and repeatably ignite, a proof-of-concept small combustion chamber, representative of that needed to enable liftoff of a small lander from the lunar surface. This engine needs to have a thrust of at least 1000N to be commercially useful, although practical constraints may limit initial tests to around 500Newtons.
4. Prototype parts for a combustion chamber design lined with a European high temperature ablative material, supplemented with boundary layer cooling, have been manufactured developed and after final assembly, hot test firing to validate the design approach is planned in 2024. Further design iterations to confirm the most efficient and low cost cooling approach, and to evaluate the benefits of generative chamber design are planned.
5. A supply chain and academic knowledge base support has been identified to (a) stress test a number of assumptions regarding the systems design, (b) to support development testing to at least TRL7, including extended duration vacuum testing in the UK. Development of a wider European partnership with the support of the European Space Agency is being explored.
6. Discussion with customers in the form of system integrators who are developing, building and flying lunar lander designs in Europe and the rest of the world is also underway.

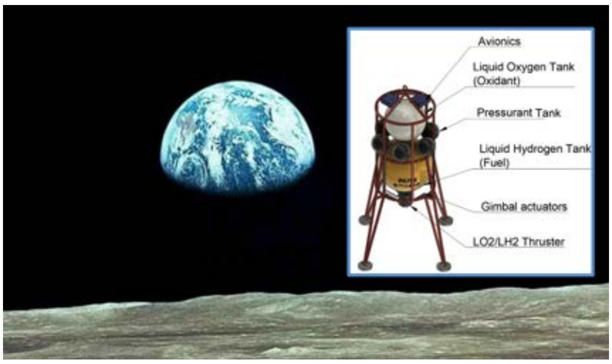

Figure 11. Rocket Engineering's goal for 2030: enabling technology for refuellable launch from the Moon

8. REFERENCES

[1] NASA: Fiscal Year 2023 Budget Request. NASA FY 2023 Budget Request Summary. Available from https://www.nasa.gov/nasa-fiscal-year-2023-budget-request/ . [Internet : Accessed 15/04/2024].

[2] ISECG International Space Exploration Coordination Group: The Global Exploration Roadmap (2018 / 2022). 7 October 2022. Available from https://www.globalspaceexploration.org/wordpress/?cat=3 [Internet: accessed 15/04/2024].

[3] ISECG Global Exploration roadmap critical technology needs (2019). 13 December 2019. Available from https://www.globalspaceexploration.org/wordpress/?page_id=811 [Internet: accessed 15/04/2024].

[4] Wikipedia: Intuitive Machines Nova-C. https://en.wikipedia.org/wiki/Intuitive_Machines_Nova-C. [Internet: accessed 15/04/2024].

[5] Foust J., (2023). A Lunar Lander Makeover. The Space Review, Monday May 22, 2023. https://www.thespacereview.com/article/4587/1 [Internet: accessed 15/04/2024].

[6] Gardiner, M. (2009). Energy requirements for hydrogen gas compression and liquefaction as related to vehicle storage needs. DOE Hydrogen and Fuel Cells Program Record #9013. July 7th, 2009. Available from: https://www.hydrogen.energy.gov

[7] Cranfield University (2024). £69 million boost for hydrogen at Cranfield. Press release number PR-CORP-24-27. Released on 26 March 2024.

[8] Bond, A. (2018). The RZ20 LOX/Liquid HYDROGEN ENGINE PROJECT – a personal memoir. British Interplanetary Society Space Chronicle, *Vol. 71, pp. 126-130.* 2018.