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A Review of In-Space Propellant Transfer Capabilities and Challenges for Missions Involving Propellant Resupply

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Nomenclature

AFIS	Automated Fluid Interface System
APAS	Androgynous Peripheral Attach System
APS	Auxiliary Propulsion System
APU	Auxiliary Power Unit
ARCS	Aft Reaction Control System
ARRM	Asteroid Retrieval and Redirect Mission
ATV	Automated Transfer Vehicle
BAC	Broad Area Cooling
CDR	Critical Design Review
CFM	Cryogenic Fluid Management
CH_4	Methane
CM	Crew Module
COLDSAT	Cryogenic On-Orbit Liquid Depot Storage, Acquisition, and Transfer
COPV	Composite Overwrapped Pressure Vessel
CPST	Cryogenic Propellant Storage and Transfer
CRS	Commercial Resupply Services
CRYOSTAT	Cryogenic Propellant Storage and Transfer
CSV	Cooperative Service Valves
DARPA	Defense Advanced Research Projects Agency
eCryo	Evolvable Cryogenics Project
EPR	Ethylene Propylene Rubber
ETU	Engineering Test Unit
EVA	Extravehicular Activity
ExIS	Exploration and In-Space Services
FARE-1	Fluid Acquisition & Resupply & Expulsion Experiment
FDV	Fill and Drain Valve
FFKM	Perfluoroelastomer
FGB	Functional Cargo Block
FRCS	Forward Reaction Control System
g	Gravity
GEO	Geosynchronous Orbit
GH_2	Gaseous Hydrogen
GHe	Gaseous Helium
GN_2	Gaseous Nitrogen
GO_2	Gaseous Oxygen
GPIM	Green Propellant Infusion Mission
GRC	Glenn Research Center
GRO	Gamma Ray Observatory
GSFC	Goddard Space Flight Center
H_2O	Water
H_2O_2	Hydrogen Peroxide
HF	Hydrofluoric Acid
HLS	Human Landing System
HMA	Hose Management Assembly
HNO_3	Nitric Acid
HPU	Hydraulic Power Unit
HSA	Hose Subassembly
ICM	Interim Control Module
IMLI	Integrated Multi-Layer Insulation
ISRU	In Situ Resource Utilization
ISS	International Space Station
JSC	Johnson Space Center

KDP	Key Decision Point
KSC	Kennedy Space Center
LCH_4	Liquid Methane
LEO	Low-Earth-Orbit
LH_2	Liquid Hydrogen
LN_2	Liquid Nitrogen
LO_2	Liquid Oxygen
MLI	Multi-Layer Insulation
MMH	Monomethylhydrazine
MON	Mixed Oxides of Nitrogen
MSFC	Marshall Space Flight Center
N_2H_4	Hydrazine
N_2O_4	Dinitrogen Tetroxide
NO	Nitric Oxide
NRL	Naval Research Laboratory
NTO	Nitrogen Tetroxide Oxidizer
OMS	Orbital Maneuvering System
OMV	Orbital Maneuvering Vehicle
ORS	Orbital Resupply System
ORU	On-orbit Replacement Unit
OSAM-1	On-orbit Servicing, Assembly, and Manufacturing 1
OSCRS	Orbital Spacecraft Consumables Resupply System
PCAD	Propulsion and Cryogenic Advanced Development
PDR	Preliminary Design Review
PMD	Propellant Management Device
PTA	Propellant Transfer Assembly
PTFE	Polytetrafluoroethylene
PTS	Propellant Transfer Subsystem
PVT	Pressure-Volume-Temperature
RCS	Reaction Control System
RFMG	Radio Frequency Mass Guage
RRM	Robotic Refueling Mission
RST	Roman Space Telescope
SBIR	Small Business Innovation Research
SBIRS	Space-Based Infrared System
SFMD	Storable Fluid Management Demonstration
SiC	Silicon Carbide
SM	Service Module
SmCo	Samarium Cobalt
SPS	Service Propulsion System
STP	Space Test Program
STS	Space Transportation System
TDT	Technical Discipline Team
TRL	Technology Readiness Level
TVS	Thermodynamic Vent System
UDMH	Unsymmetrical Dimethylhydrazine
URO	On-orbit Replacement Unit
USAF	United States Air Force
USPM	U.S. Propulsion Module

1.0 Introduction

1.1 Motivation, Objectives, and Approach of this Paper

The motivation for writing this paper was to capture and transfer knowledge of in-space propellant transfer accomplishments and communicate the remaining challenges to a new generation of NASA engineers as multiple NASA programs and centers have worked related and applicable efforts. Multiple senior engineers are approaching retirement and the agency faces the risk of losing hard-won corporate knowledge. The Propulsion Technical Discipline Team (TDT) identified this need in 2017 and established a plan to draft this paper as an effort to help engineers and decision makers recognize and access the various islands of experience that have formed within the agency as different NASA directorates and projects have embraced the challenges associated with propellant transfer. While this paper primarily focuses on in-space propellant transfer, many of the technologies and lessons discussed are also applicable to non-terrestrial propellant transfers such as on the moon or Mars.

Hence, the primary objectives of the activities summarized in this paper have been: (1) to provide an integrated assessment of inter-vehicular, in-space propellant transfer technologies, (2) to provide a historical overview of key experiments and projects that have advanced propellant transfer technologies in a significant way, and (3) to provide insight from retirement-eligible engineers regarding strategic investment needs and design pragmatism to enable propellant-transfer architectures at acceptable levels of risk in near-term space missions. This paper also provides a summary index and bibliography of related past program references.

1.2 Historical Background

1.2.1 Aviation Analogs

Early in the history of the airplane, aviators realized that flight range could be extended and trip duration could be reduced if planes could be refueled during flight. The first mid-air refueling occurred on June 27, 1923, and the practice has continued for nearly a century. While commonplace, aerial refueling continues to be an ingenious feat that relies on careful design, rigorous flight demonstration testing, highly trained pilots, and highly skilled operators. And, even after 97 years of experience in mid-air refueling, mishaps still occur.

For decades, space mission designers have also recognized the potential advantages of in-space resupply of rocket engine propellants. Yet, transferring propellants in space has proven to be more challenging than its aviation analog for several reasons, as summarized in Table 1.

Table 1. Notable Differences between Aerial and In-Space Transfer

Aerial Refueling	In-Space Propellant Resupply
Planes often land for maintenance.	Spacecraft rarely undergo hardware maintenance, if ever.
Planes typically need only liquid fuel.	Larger spacecraft also need liquid oxidizer (usually corrosive and sometimes multi-phase) and high-pressure gases, which require pauses for cool-down due to heat of compression.
Aircraft typically operate in a convective cooling environment.	The space environment precludes convective heat transfer, increasing time required for cool-down.
Planes are typically flown by human pilots.	Spacecraft are often remotely operated or autonomous.

Aerial Refueling	In-Space Propellant Resupply
Planes benefit from over 90 years of precedents with a range of aircraft designs.	Orbital Express (circa 2007), at the date of this writing, is the only known U.S.-designed full demonstration of inter-vehicular in-space propellant transfer.[1] hydrazine (N_2H_4) refueling scheduled as part of On-orbit Servicing, Assembly, and Manufacturing 1 (OSAM-1) demonstration mission in CY 2024.
Aviation enjoys a robust supplier base and a wealth of heritage designs.	The space hardware supplier base has shrunken in recent years, and there is little-to-no design heritage for some propellant transfer components.[2] And no major market for repeat use / sales other than unique government projects.
Aircraft are manufactured in relatively large quantities, helping to maintain healthy corporate knowledge.	Spacecraft (especially human-rated) are low-quantity buys, sometimes leading to dispersal of skills in the gaps between major projects.

1.2.2 Ground Processing Analogs

Servicing of propellants and pressurants on the ground has the advantage of a steady acceleration field (Earth gravity), which fixes the orientation of the liquid and gas phases for liquid propellants in relationship to each other. Due to density differences between liquid and vapor phases of a propellant, the liquid resides lower than the gas or vapor in the propellant tank. Even with turbulent flow into a tank, which can intermingle the liquid and gas phases, only a few minutes are needed for the liquid and gas phases to orient themselves as separate phases due to Earth gravity. Therefore, venting of tanks can easily be accomplished from the top of a tank or other high points in the propellant system. Servicing of propellants and pressurants on the ground is also aided by the availability and use of highly accurate, and often heavy, transducers, gages, flow meters, mass scales, and valves. The accuracy of ground instrumentation usually allows for gauging of propellant and pressurant quantities that are loaded onto spacecraft to be known with accuracies better than 1% and occasionally better than 0.25%. Typically, multiple methods are employed to determine a propellant load, which often consist of mass scales or load cells, flowmeters, run time integration, and pressure-volume-temperature (PVT) calculations. These methods used in conjunction with each other improve overall accuracy and provide double-checking or corroboration between the methods. Historic methods used for ground servicing of hypergolic propellants are shown in Table 2. If for some reason, a propellant servicing operation fails so badly that the propellant load is unusable, the propellant can be offloaded, the problems resolved, and the propellant serviced again.

[1] The Russian Space Program has, however, routinely transferred propellants on orbit for the Mir Program and for the International Space Station. See Section 2.1 for a discussion of the relevance and potential shortcomings of the Russian approach for some future missions.

[2] Examples of limited or non-existent design heritage include, for example, flight-weight flight-qualified automated cryogenic fluid couplings, flight-qualified helium compressors, flight-qualified phase separators for tank venting, etc. Some components (e.g., automated and cooperative fluid couplings for hypergols, mass flow transfer devices, quantity gauging systems, pumps, etc.) have relevant design precedents (from ground and flight applications and qualification testing) but will require significant modification to meet new mission requirements involving propellant transfer.

Table 2. Historic Methods for Hypergolic Propellant Servicing

Program	Propellants	Ground Servicing Techniques
Mercury Reaction Control System (RCS) [ref. 1]	hydrogen peroxide (H_2O_2), gaseous helium (GHe)	• Pressure transfer from ground tank(s). • GHe bubbler device connected to GHe port showed bubbles until tank bladder was full.
Titan II [ref. 2]	Unsymmetrical Dimethylhydrazine (UDMH), dinitrogen tetroxide (N_2O_4)	• Pumps used to transfer into flight tanks until the 95% sensor was activated (or 60% for Stage 2 oxidizer tank). • Ground Vernier tank (calibrated in gallons) was then used for final flight load via pump. • High-level capacitance-type sensor light activated when full and thrust-mount load-cell weighing system was used as confirmation.
Gemini RCS [ref. 3]	Monomethylhydrazine (MMH), N_2O_4, gaseous nitrogen (GN_2)	• Vacuum backfill of Teflon bladder tanks. • Ground tanks with volumetric metering sight glass used to verify load within ~0.25% accuracy.
Apollo Service Propulsion System (SPS) [ref. 4]	Aerozine-50, N_2O_4	• Used on-board capacitance-type gauging to verify loads.
Apollo Service Module (SM) RCS [ref. 5]	MMH, N_2O_4	• Evacuation backfill performed with mass scales along with fill to overflow for some tanks and PVT.
Apollo Saturn IV-B Auxiliary Propulsion System (APS)	MMH, N_2O_4	• Propellant was pumped into the tanks after collapsing the internal bladder.
Apollo Lunar Module [ref. 6]	MMH, N_2O_4	• Used on-board gauging calibrated offline to verify loads.
Space Shuttle OMS/RCS [ref. 7]	MMH and N_2O_4	• Remote loading form console with autosequencer following Space Transportation System (STS)-26R for Aft Reaction Control System (ARCS) and STS-71 for Forward Reaction Control System (FRCS). • FRCS was fill to spill or PVT with flow meters as backup (turbine until STS-125 and then Coriolis). • ARCS were always fill-to-spill. • Orbital Maneuvering System (OMS) used internal gauging probe along with turbine flow meters both until STS-125 and then Coriolis flow meters used.
Space Shuttle APU/HPU [ref. 7]	N_2H_4	• Auxiliary Power Unit (APU) vacuum backfilled using a mass scale under the ground supply tank. • Hydraulic Power Unit (HPU) used mass scales and PVT to verify the pressure driven load.
Orion Crew Module (CM) [ref. 7]	N_2H_4	• Redundant load cells in conjunction with Coriolis flow meters used to measure pressure-driven transfer amount via evacuation and backfill of diaphragm tanks.
Orion SM [ref. 7]	MMH, N_2O_4	• Redundant load cells in conjunction with Coriolis flow meters used to measure pressure-driven transfer amount into Propellant Management Device (PMD)-style tanks.
General Spacecraft [ref. 7]	MMH, N_2H_4, and N_2O_4	• Loading technique is vehicle/tank dependent and is typically verified with a mass scale with a flow meter as a backup/redundant measurement.

An in-space servicing/resupply system by contrast has to overcome many obstacles that a ground-based propellant servicing system does not. Some of these obstacles include the following: (1) a 1-g acceleration field does not automatically exist for separating and orienting liquid- and gas-phase propellants within a tank to enable venting or recirculation flow, (2) space-qualified hardware typically does not have the accuracy and robustness of ground-based hardware that has little-to-no mass and/or volume constraints, (3) the small (or possibly the absence of) crew to perform the servicing operations affects the allowable duration of a servicing operation and the diagnosing of anomalies and recovering from the anomalies. Anomalies are likely to happen and will be very challenging to overcome without direct human presence. A detailed study of the final 45 Space Shuttle launch-pad hypergolic propellant servicing operations showed that there was not a single propellant load in which an anomaly did not occur. In fact, the least problematic propellant loading operation, STS-128, had a total of four anomalies that had to be overcome to successfully complete the servicing operation.

The cryogenic propellant ground servicing process also has many unique challenges as summarized in the following detailed (general) cryogenic propellant servicing assumptions, preparations, and filling process steps. These steps have many lessons that should be acknowledged and possibly modified for the space environment during a potential future in-space cryogenic propellant servicing operation.

- <u>Assumptions</u>
  - Internal system has been cleaned to commodity specifications
  - Internal system has been leak checked at low and operating pressure levels using a gas commodity (typically helium or nitrogen)
  - Internal system has been cold filled once (cold shocked) with an inert cryogenic fluid (typically liquid nitrogen (LN_2))
    - No cryogenic leaks verified at all system connection fittings and flanges
    - System piping and tubing contraction and expansion travel is verified within mounting support gaps along with verification of no excessive binding between mounts and system components
      - If adjustments are required, re-perform cold shock and inspection process to verify proper system travel during contraction and expansion
  - System connection fastener torques verified after system has warmed to ambient temperatures
  - Perform system final leak check at low pressure using a gas commodity
  - Ensure system is maintained clean and stored with an inert gas pad pressure
- <u>Tank Servicing Operations Preparation</u>
  - Pulse purge (low-pressure pressurization and venting cycles) with an inert gas that has a saturation temperature that is less than the saturation temperature of the cryogenic liquid being used in the system (i.e., do not use GN_2 for liquid hydrogen (LH_2) system) to remove any ambient atmospheric gasses
    - Concentration percentage verified by sampling process (most conservative check)
    - Use a volume fraction algorithm to determine the number of pressure vent cycles to achieve an acceptable inert gas concentration (less conservative check)

- Tank Servicing Process
  1. Initiate system chill operation (cold gas chill)
     a. Open high-point bleed vent located at the top of the tank or near the inlet of the first closed isolation valve
     b. Throttle valve open very slowly to allow cryogenic fluid to enter ambient system tubing or piping at a very low rate (large pressure fluctuations will occur if opened too rapidly)
        - Monitor system pressure and temperature instrumentation
        - If high-pressure cycling occurs, reduce cryogenic liquid flow rate
     c. As system temperature instrumentation and pressure cycles start to reduce to liquid saturation temperatures and pressure cycles reduce, increase cryogenic flow rate with the following considerations:
        - Bottom tank temperatures will equal the liquid saturation temperature when the transfer tubing or plumbing are completely filled and the tank starts to receive liquid
        - Bottom tank pressurization instrumentation will start to increase as the liquid height starts to increase
        - Top tank pressurization instrumentation will reduce to ~ambient pressure when the cryogenic fluid gas density increases to a temperature just above the saturation temperature
  2. Increase flow rate (slow fill)
     a. Increase flow rate to an acceptable level until tank reaches 5 to 10% full
        - Structural thermal analysis or empirical-derived acceptable level
     b. Monitor tank ullage pressure instrumentation and verify no pressure and temperature oscillations that would indicate a rapid thermally induced structural contraction is occurring, reduce flow rate if required
  3. Increase flow rate (fast fill)
     a. Increase flow rate to an acceptable level until tank reaches 90 to 95% full
        - Structural thermal analysis or empirical-derived acceptable level
     b. Monitor tank ullage pressure instrumentation and verify no pressure and temperature oscillations that would indicate a rapid thermally induced structural contraction is occurring, reduce flow rate if required
  4. Decrease flow rate (slow fill)
     a. Decrease flow rate to an acceptable level until tank reaches ~90 to ~100%
        - Structural thermal analysis or empirical derived acceptable level
  5. If required, decrease flow rate (topping fill) to an acceptable level that matches or maintains ~100% fill level due to boil-off
  6. If required, stop servicing operations by closing transfer valves, opening transfer system vent valves, and or purge/vent system until ambient temperature is indicated and all cryogenic liquid fluid is removed

- <u>Transfer System Servicing to a Closed Isolation Valve Process</u>
  1. Initiate system chill operation (cold gas chill)
     a. Open high-point bleed vent located near the inlet of the closed isolation valve
     b. Throttle valve open very slowly to allow cryogenic fluid to enter ambient system tubing or piping at a very low rate (large pressure fluctuations will occur if opened too rapidly)
        - Monitor system pressure and temperature instrumentation
        - If high pressure cycling occurs reduce cryogenic liquid flow rate
  2. Increase flow rate (system fill)
     - Monitor system pressure and temperature instrumentation
     - If high-pressure cycling occurs reduce cryogenic liquid flow rate
  3. System filled (maintain system fill)
     a. Open (fully) the downstream supply tank throttling valve
     b. Monitor system pressure and temperature instrumentation and cycle/throttle open/close high-point bleed or isolation valve vent valve and maintain system fluid saturation temperatures throughout the transfer system

Ground servicing of pneumatic systems is a fairly straightforward process. The flight high-pressure gas tanks are loaded one of two ways on the ground prior to launch: 1) pressure transfer from a large volume, high-pressure supply source, or 2) pump/compressor transfer from a low-pressure, high-volume supply source. Flight tanks are typically loaded until a temperature (or sometimes pressure) limit that is near to the tank qualification limit (minus some margin) is reached. At this time, the isolation valve is closed (or the compressor is deactivated). After that, the tank is allowed to cool through natural convention and conduction and the process starts again. For NASA, this ground operation is typically done remotely due to safety requirements if the high-pressure flight tank is pressurized above 1/3 of the design burst pressure [ref. 8]. The design burst pressure for a flight composite overwrapped pressure vessel (COPV) is typically 1.5 to 2.0 times the operating pressure [ref. 9]. A pneumatic servicing process in space will likely be the same except that the natural convention heat transfer will not be present.

1.3 Potential Demands of Future Missions

In recent years, NASA has given very serious consideration to missions that seek to extend capabilities, range, and duration using technologies such as in-space inter-vehicular propellant transfer, extra-terrestrial or in-space propellant production facilities, and in-space propellant depots. These considerations have ranged from resupplying hypergolic earth-storable propellants into low-Earth-orbiting (LEO) and geosynchronous orbit (GEO) satellites all the way to very large-scale space architectures based on *in situ* resource utilization (ISRU). While ISRU could be used for a range of potential propellants, pragmatic consideration of existing and maturing propulsion technologies has focused NASA's larger investments on the potential benefits of two main approaches: (1) LO_2 and liquid methane (LCH_4) using Martian atmosphere and melted, electrolyzed Martian ice as primary feedstocks and (2) LH_2 and LO_2 derived from water electrolysis. There is also strong interest in LH_2 itself as a propellant for nuclear thermal propulsion systems. Strategic trade studies comparing LO_2/LCH_4 and LO_2/LH_2 have been notoriously challenging due to the high sensitivity to factors such as mission duration, thermal environments, the difficulty of storing LH_2 versus LCH_4 over the long term, the risk tolerance of robotic versus human missions, and a complex mix of technology mitigation efforts required for risk acceptability in time for a project's exit from the preliminary design review (PDR).

1.4 Scope and Intended Use of this Document

As can be seen from this very brief discussion, any assessment of technology gaps for in-space propellant transfer is heavily dependent on mission objectives and mission design. Propellant transfer technology infusion scenarios range from relatively straightforward cooperative satellite refueling with N_2H_4 to large-scale system-of-systems architectures involving more challenging cryogenic propellants that are manufactured, liquefied, stored, and transferred on the Martian surface with semi-autonomous or autonomous control. The consequences of failure among these scenarios range from potential loss of a robotic satellite to devastating loss of a human crew and a multi-billion-dollar exploration mission. Therefore, the authors wish to point out that this document is not intended to address the full range of questions that will inevitably arise as future programs and projects assess their technology risks. Instead, this document purports to serve as a guide to identifying and accessing experience and expertise derived from the agency's various experiences and investments related to in-space propellant transfer. Our hope is that this collection of experiences will help future projects avoid unnecessary repetition of previous work and leverage previous accomplishments by providing an integrated view of past work across NASA's internal organizational/geographic boundaries and highlight key reference documents that contain further details.

2.0 Chemical Propulsion Systems

2.1 Chemical Propellant Transfer Background

Since most of NASA's in-space propulsion systems have been based on past proven and highly reliable chemical propulsion technologies, it is not surprising that nearly all the agency's investments in and experiences with in-space propellant transfer have been linked to chemical propulsion. While the concept of in-space propellant resupply has existed since early in NASA's history, the agency's move toward the Space Station era prompted serious study of propellant resupply for on-orbit servicing of station elements and future exploration vehicles, using concepts such as the "space tug," which eventually evolved into the Orbital Maneuvering Vehicle (OMV) Project. During the 1980s, recognizing a lack of experience transferring liquids across vehicle interfaces in space, the agency began a series of ground and flight experiments to simulate the surface-tension-based acquisition and transfer of storable propellants (see Section 2.2). In the early days of the Space Station Freedom Program, the Agency evaluated two primary options for Freedom's reboost propulsion capabilities: (1) a Marshall Space Flight Center (MSFC)-sponsored option using gaseous oxygen (GO_2) and gaseous hydrogen (GH_2) based on water electrolysis using projected excesses of water aboard Freedom, and (2) a Johnson Space Center (JSC)-sponsored option using monopropellant N_2H_4. The program selected the N_2H_4 concept, which would have allowed positive expulsion tanks to simplify propellant transfer concerns. Hence, early investments in propellant transfer technologies focused first on storable hypergolic propellants (particularly N_2H_4), due to ease of long-term, reliable use and storage in space, as was already evident in U.S. satellites. During the Space Station restructuring of 1993, which added Russia to the list of international partners, responsibility for station reboost propulsion was deleted from the U.S. segment and added to the Russian segment of the Station.

This enabled the Space Station Program to leverage Russia's existing propellant resupply systems from the Space Station Mir and Salyut Programs. Initial on-orbit capabilities were to rely on the Russian Functional Cargo Block (FGB, now named *Zarya*) and ultimately the Russian Service Module (now named *Zvezda*). Delays and funding issues with the Zvezda module, however, led to the agency's investment in two U.S.-led backup options: (1) the Interim Control Module (ICM) based on the Naval Research Laboratory (NRL) bus and (2) the U.S. Propulsion Module (USPM), which was to leverage excess hardware from the Shuttle Program (particularly from the Shuttle's OMS and RCS). The ICM spacecraft was assembled but never launched. The USPM was redirected and eventually cancelled following the late but successful deployment of Zvezda in August, 2002. When Zvezda arrived on orbit, the USPM Program had successfully passed its PDR and was progressing toward critical design review (CDR). At the time of this writing, the Russian Zvezda continues to provide reboost propulsion for the International Space Station (ISS) along with the occasional reboost from the European Space Agency's Automated Transfer Vehicles.

Interest in U.S.-based in-space transfer of storable propellants was soon reasserted in a Defense Advanced Research Projects Agency (DARPA)-NASA program called Orbital Express. Launched March 8, 2007, this mission still stands as the most significant known U.S.-developed demonstration of storable propellant transfer between two cooperative (designed to be serviceable) spacecraft.

Shifting our focus back to cryogenics, however, we need to revisit the late 1980s and the 1990s. While much of the activity in this period focused on storable hypergol options, NASA also undertook in-depth studies of the more thermodynamically challenging storage and transfer of cryogenic propellants, due to the performance advantages in the LO_2/LH_2 propellant combination for large-scale human rated delta-V propulsion systems (see Section 2.3). Some of the most valuable products of this late-1980s interest in cryogenic storage and transfer are the NASA and industry study reports for an envisioned flight demonstration known as the Cryogenic On-Orbit Liquid Depot Storage, Acquisition, and Transfer (COLDSAT) spacecraft. While COLDSAT never proceeded to development, the program's three parallel contractor reports and the subsequent NASA summary report provided an in-depth assessment of technology readiness and necessary investments that have competently informed technology investments for 30 years. As the Agency began to formulate the Cryogenic Propellant Storage and Transfer (CRYOSTAT) Mission in 2009–2010, engineers found that many challenges identified in the COLDSAT studies had received little attention since the COLDSAT reports were published.

In the 1990s and early 2000s, interest in U.S.-based CRYOSTAT continued to receive low-level investments due to Agency interest in future placement of cryogenic depots in LEO or LaGrange points to enable resupply of exploration vehicles. This period of low funding and relative calm in the cryogenic fluid management technologists' community was interrupted for a brief time in the Ares-I launch vehicle program when the agency baselined LO_2 and methane (CH_4) for the Orion SM propulsion systems. This decision placed in-space sub-critical cryogenic liquids at the forefront and at high priority. When Ares-I risk and programmatic assessments identified this decision as a cost and schedule driver, NASA reverted to conventional hypergols for the SM and reassigned investments in LO_2/LCH_4 to technology development activities. Initially, the technology development activities were managed by the Propulsion and Cryogenic Advanced Development (PCAD) Project, including investments in cryogenic propellant depot technologies for LO_2 and LCH_4. As the agency shifted away from the Constellation Program and the Ares

Launch Vehicles, new scenarios piqued NASA's interest in Cryogenic Depots as a possible exploration architecture, prompting further depot studies. This depot concept and its future variants would have used small commercial launch vehicles to ferry propellants to an in-space depot, which would then resupply Earth-departure stages and human-rated transfer vehicles for large-scale exploration missions.

The agency's boldest step toward in-space CRYOSTAT stemmed from the Fiscal Year 2009 Budget request, which evolved into the termination of the Constellation Program and, among other things, the initiation of a flagship mission to demonstrate long-duration in-space storage and transfer of cryogenic propellants for a broad range of future stakeholders, such as human and robotic missions involving chemical or even nuclear propulsion. Initially envisioned as a billion-dollar-class mission, funding constraints continued to descope this mission, leading to its renaming as the Cryogenic Propellant Storage and Transfer (CPST) flight demonstration project and its eventual descoping to a ground demonstration of highest-priority ground-testable technologies.

Following the end of the Constellation Program and ramping up of subsequent programs such as the Asteroid Retrieval and Redirect Mission (ARRM), Gateway, and Artemis, the agency saw the need to continue the advancement of in-space servicing technologies. This resulted in several years of research and development of in-space propellant transfer and robotic operations performed on the ISS with the Robotic Refueling Missions (RRM) along with the development of the OSAM-1 mission, formerly Restore-L, which plans to perform a non-cooperative (client not designed for servicing) N_2H_4 propellant servicing operation of the Landsat 7 satellite in low Earth polar orbit in about 2024. The technologies and intellectual property for the operational concepts developed have also been shared with U.S. industries for licensing and potential future use on commercial propellant servicing missions as well as possible applications on Gateway, Human Landing System (HLS), manned Mars Programs, large observatories, military, and ISRU payloads or vehicles.

From 2010 to the release of this report, ongoing efforts have been continually progressing through projects such as the RRM and OSAM-1 along with several hypergolic, cryogenic, and xenon related demonstration tests. Additionally, in parallel, exploration programs, technology programs, and industry partners have continued low-level investments in relevant cryogenic activities, including subscale flight experiments, ground-based liquefaction experiments, radio-frequency mass gauge experiments, and ground-based operational demonstrations with various test articles at MSFC and Glen Research Center (GRC) including tests of the United Launch Alliance's CRYOTE test article at MSFC.

In light of this general historical framework, the following paragraphs will summarize accomplishments in each of these areas in more detail.

2.2 Storable Hypergolic Propellant Systems

Beginning in the 1980s, NASA began to make significant strides toward identifying and overcoming the perceived challenges of transferring hypergolic propellants in space. Table 3 provides highlights of the most relevant known experiments.

Table 3. Highlights of Early Propellant Transfer Experiments

Experiment Name, Platform, and Investigator	Fluid and Configuration	Significance
Orbital Resupply System (ORS) aboard STS-41G (1984) **NASA JSC**	• Loaded with 904 lb_m of N_2H_4. • In Shuttle Payload Bay with Extravehicular Activity (EVA) required to connect simulated tanker to simulated satellite. • Fluid connection was not demated until return to Earth. • GN_2 used as pressurant	• Addressed concerns of overheating N_2H_4 due to ullage compression during tank fill. • Performed 6 tank-to-tank transfers. • Involved 285 minutes under flow conditions. • Proved ullage compression to be more isothermal than adiabatic, relieving N_2H_4 overheating concerns.
Storable Fluid Management Demonstration (SFMD) **Aboard STS-51C (Jan 1985)** **Martin Marietta [ref. 10]**	• Used colored water in clear acrylic tanks for visibility. • Performed in Space Shuttle Orbiter Mid-deck. • Receiver tank contained a surface tension PMD. • Supply tank featured positive expulsion.	• Demonstrated successful filling of a receiver tank in microgravity. • Demonstrated gas-free expulsion of liquid from the supply tank. • Provided observations of fluid motion in microgravity. • Included 9 tests. 2 tests were under acceleration. 1 test involved ullage recompression. • Limitation: Provided no means of orienting liquid away from the tank's vent port.
Fluid Acquisition & Resupply & Expulsion Experiment I (FARE-I) **Aboard STS-53 (Dec 1992)** **MSFC**	• Included Screen-type PMD. • Used SFMD Hardware (clear acrylic tanks). • Used filtered deionized water with wetting agent and food coloring.	• Included 8 tests (5 of which were vented). • Provided observation of screen PMD performance and demonstrated expulsion efficiency. • Provided flight data for model correlation.
FARE-II **Aboard STS-57 (June 1993)** **MSFC**	• Included vane-type PMD • Used filtered deionized water with wetting agent and food coloring	• Provided first visual data for vane PMD under prolonged microgravity. • Provided flight data for model correlation.

2.2.1 Monopropellant N_2H_4 Systems

2.2.1.1 Orbital Refueling system (ORS) – STS-41G

The ORS Flight Demonstration was a low-cost system built by JSC to demonstrate the equipment and procedures necessary to transfer liquid N_2H_4 fuel, free of gas bubbles, on orbit. The underlying objective was to develop, demonstrate and evaluate the tools and procedures necessary to interface with and transfer fuel to an existing satellite.

STS-41G launched on October 5, 1984, and over the course of the mission demonstrated six transfers moving a maximum of 142 kg of N_2H_4 back and forth. Mating and demating of the simulated transfer line connector was accomplished manually by astronauts during an EVA.

One of the major concerns for the operation was the heating of the GN_2 pressurant gas behind the bladder. Because N_2H_4 decomposes exothermically above approximately 200 °F, the transfer process was controlled to limit ullage gas temperatures to 150 °F. Unfortunately, the ORS instrumentation was limited to one temperature sensor for each tank mounted on the sidewall external to the tank, leaving actual ullage gas temperatures unknown. To deal with this setback, temperature and pressure were maintained to desired levels using a PVT calculation during the transfer process. Because of the heating concern and the uncertainties about the pressurant temperature, the largest transfers were completed in two phases separated by a 3-hour cool-down period. The final transfer was performed with the receiving tank being vented of ullage gas during propellant transfer.

Post-test analysis of the transfer data showed that as the transfer progressed, the conditions were much closer to an isothermal process as opposed to an adiabatic process. The on orbit data agreed very well with predictions based on ground test data; in fact a higher than expected degree of heat dissipation occurred, which was similar to the response seen in the ground test. As a result of the test, it was concluded that with the configuration used, tank pressure was preferred over the tank wall temperature measurement as an indicator of ullage gas conditions [ref. 11].

2.2.1.2 Miscellaneous Studies

Responding to agency desires to refuel the Compton Gamma Ray Observatory (GRO) in flight, NASA JSC funded contracts to develop phase B design studies for a N_2H_4 servicing vehicle known as the Orbital Spacecraft Consumables Resupply System (OSCRS) in the 1986 to 1987 timeframe. The contracts awarded to Rockwell and to Martin-Marietta led to two independent design studies published in multi-volume reports.

These studies later informed the OMV project managed by MSFC and executed by TRW. The OMV would have eventually involved “space tug” services as well as propellant and pressurant gas resupply, but the project was cancelled in 1990, due to general budget constraints, lack of stakeholder requirements, and a far-term need date [ref. 12]. At the time of termination, propellant resupply requirements were unclear and the eventual implementation uncertain.

2.2.1.3 Orbital Express

The most significant development in the U.S. capability for transferring monopropellant N_2H_4 in space is the successful Orbital Express Program sponsored by DARPA with NASA as a partner discussed in the next section. Launched March 8, 2007, aboard an Atlas V vehicle as part of the United States Air Force (USAF) Space Test Program (STP), Orbital Express demonstrated automated rendezvous and capture using two satellites, NEXTSat provided by Ball Aerospace and ASTRO provided by Boeing with the fluid transfer systems on both vehicles and the ASTRO propulsion system designed and built by Northrop Grumman Space Technology [ref. 13].

Prominent among Orbital Express’ accomplishments is a very thorough set of N_2H_4 transfer demonstrations using helium as pressurant or a pump to transfer to a cooperative client. Fifteen different scenarios were performed over a flight period exceeding 100 days [ref. 14] (with 13 scenarios considered successful). The demonstrations involved transfers back and forth between

NEXTSat and ASTRO and thus involved both a positive expulsion (diaphragm) tank and a tank featuring a vane-type PMD in both supply tank and receiver tank roles. Remaining highlights of the demonstrations are summarized in Table 4.

Table 4. Highlights of the Orbital Express N_2H_4 Transfer Demonstrations

Feature / Consideration	Relevant Demonstration Highlights
Tank fill fractions	• Varied fill fractions in receiver and supply tanks.
Transfer flow rates	• Varied flow rates by a factor greater than 3.
Flow rate measurement	• Included two liquid and one gas flow sensor that utilized two embedded probes (one heated and one not) to measure the temperature (proportional to the flow rate) of the fluid. • Results were marred by failure of one sensor during launch. • Remaining sensor showed > 10% error at high and low flow rates but only about 5% at mid-range compared to PVT [ref. 14].
Propellant quantity gauging	• Used PVT gauging technique and pump speed integration technique. • PVT method fell short of target accuracies at high fill levels. • Pump speed integration was generally within about 5% of PVT results.
Automated Fluid Coupling	• Used a Moog-provided coupling based on earlier NASA MSFC investments in the "Automated Fluid Interface System" (AFIS) version of the coupling. • Demonstrated 9 mate/demate cycles on orbit.
Ullage compression	• Demonstrated recompression of helium in the PMD tank with no indication of N_2H_4 overheating.
Transfer pump	• Demonstrated pumped N_2H_4 transfer with no significant issues. • Pump very similar to Space Shuttle APU pump. • Pump motor derived from ISS Rotary Fluid Separator [ref. 13].
Catalytic non-propulsive venting and on-board liquid purging	• Following the interface leak check, GHe was vented using the non-propulsive vent. • Liquid purging from Astro to NEXTSat after propellant transfer was performed using non-propulsive vent [ref. 14].

Based on the Orbital Express experience, storage and cooperative transfer of monopropellant N_2H_4 has arguably transitioned from a set of technology risks to a suite of ordinary engineering design development and qualification challenges unique to the mission requirements. Future design activities can leverage some mature, flight-demonstrated technologies but may still encounter challenging mission requirements, just as in any other spacecraft development. Furthermore, nitrogen tetroxide chemical storable systems, although similar, would have unique challenges to overcome such as the extreme corrosiveness demanding a seal-less design for pumps and mechanical connections, potential two phase flow impacts, and iron nitrate precipitation issues.

2.2.2 OSAM-1 (Formerly Restore-L) Mission

On-orbit hypergolic replenishment studies and development testing started at Kennedy Space Center (KSC) in collaboration with Goddard Space Flight Center (GSFC) in 2010 with a focus on GEO satellite servicing concepts and early risk reduction testing on key technologies. Details of the GEO satellite servicing design efforts are further described in Section 2.2.3.3 of this document. These studies eventually evolved into a formal mission called Restore-L, but changed from a GEO to a LEO satellite servicing mission.

In May of 2016, NASA officially began the development of the Restore-L mission, which was later renamed to the OSAM-1 when in 2019 another payload was added to develop the technologies to perform on-orbit assembly of a 3-meter communications antenna and manufacture a 10-meter lightweight composite beam via 3-D printing. The mission successfully completed its PDR in November of 2017 and is planned to complete its CDR in the fall of 2020. The mission also successfully completed the formal NASA funding milestone known as KDP-C (Key Decision Point C). The Propellant Transfer Subsystem (PTS) level engineering CDR was completed in July 2019 [ref. 15]. This mission is the first attempt to develop the technologies to service a non-cooperative spacecraft on-orbit. The current mission baseline has Landsat 7 as the candidate client [refs. 16 and 17]. Along with developing key technologies to enable NASA's future deep space missions, OSAM-1 hosts a technology-sharing forum annually with industry and other government organizations in an effort to jumpstart the commercial U.S. satellite servicing industry.

OSAM-1 is designed to use robotic tools to access the ground servicing fill and drain valves (never meant to be accessed after launch), mate to one of the servicing valves, perform a precision leak test using helium, flow a predetermined amount of N_2H_4 propellant via a pressure-driven transfer (with modeling analysis based on the STS-41G testing for start, stop, and cooling cycle timing required), remove propellant from the interface to trace vapor levels, and safely disconnect from the client. Each step of this process has challenges that have been studied for years while incorporating lessons learned from historic programs.

Several unique challenges exist with the design of a propellant transfer system for a client not designed to be refueled. One that has not been previously discussed is the connecting device between the servicer and the non-cooperative client. The device was designed to ensure that it is cross-cutting to potential future oxidizer servicing missions, so a seal-less design became a baseline requirement, meaning that all fluid interfaces for the device had to be welded, yet have means to accommodate some rotational twist on its path to the client fill and drain valve (FDV) interface via the robotic arm. On OSAM-1, this device is the Hose SubAssembly (HSA).

The HSA utilizes an inner metal convolute hose with several layers of heaters, pass-through wiring, insulation, and a coated metal outer conduit to meet the requirement for unlimited cold (shadow) and hot (sun) exposure, fluid temperature control, along with radiation and space charge dissipation. The hose connects to a custom designed tool that is "flown" to the client interface using a robotic arm controlled by ground operators. The inner hose is a unique annealed stainless steel, 100% radiographed metal convoluted design with proven qualification test results to enable twist and torque cycles that far exceed mission requirements while still meeting 5x burst levels to operating pressure [ref. 18]. Also, the HSA requires a storage mechanism, so a complex set of roller drives, a motor, and uniquely shaped box were created. The hose and storage device were named the Hose Management Assembly (HMA). The HMA has undergone multiple design iterations and Engineering Test Unit (ETU) environmental tests to ensure it will meet flight requirements [ref 19].

The verification of a leak-tight interface with the client prior to servicing is a key technology required for propellant servicing, as it was on Orbital Express. Several tests were performed on similar servicing valves including long-term exposure to valves on decommissioned Peacekeeper stages, exposure of new valves to radiation, and bench top testing with intentionally damaged valve seals. Following all this testing, it was determined that while there was still a residual risk of leakage, the likelihood was low for the Landsat 7 heritage FDV that contained Teflon and ethylene propylene rubber (EPR) internal softgoods [ref. 20].

Leveraging the historic data from Orbital Express and STS-41G, the client ullage compression was carefully studied to ensure the qualification temperature limit of the 20+ year old N_2H_4 diaphragm client tank was not exceeded. A detailed thermodynamic and heat transfer propellant transfer model that can be switched between an adiabatic and isothermal process was created to characterize the physics and was successfully anchored against a ground test setup using water for the isothermal case [ref. 21].

The propellant transfer operation also utilizes a turbine flow meter upstream of the transfer hose that was derived from a similar ground unit. The flow meter has been rigorously qualification tested to verify it meets all the environmental requirements for on-orbit life while still maintaining a high accuracy. Calibration testing revealed approximately 1% error at varying temperature and viscosity ranges verified after all major environmental tests [ref. 22].

Since the client interface was not meant to be touched following launch, a post-servicing circulation purge of the trapped liquid volume at the interface is not possible (as it was during Orbital Express and ground servicing using a hand-held aspirator). A technique using a catalytic vent along with system heaters was devised to reduce the propellant quantity to trace vapor amounts over a period of several hours. This was verified by analysis and testing using water as a simulant and analyzed for a N_2H_4 use application [refs. 23 and 24].

The OSAM-1 PTS was also designed to be replenished on-orbit. The system includes two Cooperative Service Valves (CSVs) that enable on-orbit servicing. The CSV was developed internally at the NASA GSFC Exploration and In-Space Services (ExIS) Division to replace standard fill/drain valves. The valves utilize internal seals and, therefore, do not require caps or wires to achieve two-fault tolerant sealing, a driving range safety requirement. Removal of these components makes on-orbit fueling significantly less complex. The valves feature other enabling characteristics as well, like integrated heaters and structural interfaces to react on-orbit loads. While there is no plan to utilize these valves on OSAM-1 on orbit, their inclusion in the mission demonstrates the feasibility of such a concept. They are planned to be utilized during ground servicing of OSAM-1.

In summary, the OSAM-1 mission will greatly advance many propellant servicing and other technologies that will be drawn upon for years to come. Spinoff technologies are already being incorporated into spacecraft currently under development, such as the Roman Space Telescope (RST), to enable easier on-orbit serviceability.

2.2.3 Bipropellant Storable Hypergolic Propulsion Systems

Conventional hypergolic bipropellant systems typically rely on a N_2H_4 derivative fuel and a N_2O_4 formulation as oxidizer, also referred to as nitrogen tetroxide (NTO). In the U.S. space program, the fuel is typically MMH (or $N_2H_3CH_3$ or $CH_3(NH)NH_2$), and the oxidizer is a solution of N_2O_4 containing some percentage of nitric oxide (chemical formula NO) as a stress corrosion cracking inhibitor. In the Russian Space Program, including the Russian elements of the ISS, the fuel is (or $N_2H_2(CH_4)_2$), which fell into disuse in the U.S. due to supplier issues following the Apollo era. The Russian oxidizer is comparable to the U.S. formulation of NTO.

Fortunately, MMH, and UDMH are sufficiently similar to N_2H_4 that the Orbital Express and similar accomplishments have very strong relevance to the components and general design of systems that would transfer these propellants in space. On the other hand, NTO brings new challenges to the overall design of transfer systems, as enumerated below:

1. While N_2H_4 transfer systems can be simplified using positive expulsion tanks (particularly polymeric diaphragm tanks common in the U.S. space program), N_2O_4 is significantly more chemically aggressive toward polymers, limiting diaphragm tanks to a short (i.e., anecdotally a 2-week) exposure time before the polymer fails. NOTE: The Russian propulsive elements aboard ISS solve this problem through the use of metal bellows tanks with a high residual volume that is well suited to a long-life on station with frequent resupply missions but poorly suited to other applications that need high expulsion efficiency and have only a small number of planned resupply missions.[3] Metallic diaphragm tanks could be an option for providing positive expulsion capabilities for MMH and NTO tanks, and NASA MSFC and KSC have gained some ground test experience with military-heritage metal diaphragm tanks and custom built titanium bellows tanks that could withstand the corrosive NTO environment. This option, however, is mostly relevant to propellant *supply* tanks, since conventional metal diaphragms are typically designed for one-time expulsion but not refilling the tanks (i.e., diaphragm reversal) or multiple cycles of expulsion and refill.
2. Since conventional NTO tanks in the U.S. space program employ PMDs rather than positive expulsion devices, interaction between the NTO solution and the ullage must be considered. The NO dissolved in the NTO has a high partial pressure and preferentially enters the gas phase when tank pressure decreases. As receiver tanks are vented in preparation for transfer, it is possible that NO can become depleted over time, increasing the risk of stress corrosion cracking of titanium tanks and other components. Structural failure of a propellant tank is a critical issue threatening loss of crew and loss of mission.
3. Propellant ejection may be required to meet mission requirements and the risk of ejecting liquid propellant overboard during 1) receiver tank venting (to decrease pressure prior to tank resupply), 2) to clear propellant from the interface connection, or 3) during fluid coupling mate/demate cycles threatens optical surfaces, solar arrays, and EVA suits and masks.
    - Since NTO is far more corrosive than N_2H_4 and its derivatives (MMH and UDMH), care needs to be taken when ejecting or venting is performed.
    - There is also risk of material incompatibilities with N_2H_4. The OSAM-1 mission initiated a test program to study the material impacts in the event N_2H_4 liquid or vapor came into contact with spacecraft surfaces when released. The results showed that many materials including polyimide insulation blankets and polycarbonate camera lenses would be significantly damaged if releases were not decomposed using a monopropellant thruster like that used on Orbital Express

[3] It is important to grasp this point before proposing to apply the successful Russian ISS approach to other mission scenarios. Although the Russian bellows tanks are qualified for deeper expulsion cycles, the tanks are managed to maintain a 50% residual volume to prolong bellows life. In calculating lifetime expulsion efficiency, this initial 50% inefficiency occurs only once with future refill operations enjoying 100% expulsion efficiency. Hence, the lifetime average expulsion efficiency approaches 100% as the number of refills approaches infinity. This approach is very well suited for the ISS application, but is very costly for satellite servicing or resupply of human spacecraft on distant exploration missions involving ISRU propellants.

and OSAM-1 or directed away from the spacecraft using a non-propulsive vent if mission geometries allow.

- The ISS manages this risk by locating the Russian propulsive elements at the aft end of the ISS away from solar arrays, windows, and star trackers and defining the aft end as an EVA keepout zone. Smaller or more tightly integrated spacecraft may not be able to use this approach and will likely need to design very low-spill automated fluid couplings and develop a robust approach for liquid-gas separation during receiver tank venting.

4. Another risk arises from NTO's tendency to deposit sometimes-unstable solid nitrates onto valve sealing surfaces over time. As solid crystals, these nitrate deposits can prevent seals from fully seating; as nitrate compounds, these deposits can even damage seals or components through micro-explosions when energy is added (e.g., the kinetic energy of a fast acting solenoid valve impacting the nitrate on the seal surface). These nitrates, because of their low solubility in NTO, eventually saturate the NTO/nitrate solution. Subsequently, solid particles of iron nitrate may form in the propellant, either remaining suspended in the solution or precipitating out. Over the years, numerous failures have occurred in manned spaceflight and satellite programs, both on the flight vehicles and on the ground, because of this phenomenon. Typical types of failures associated with nitrate contamination include: plugging of orifices and injectors, leaking or non-functioning seals and a shift in fuel/oxidizer ratio referred to as "flow decay" (flow decay is defined as an impaired flow within the NTO system caused by the deposition or accumulation of iron nitrates in quantities sufficient to degrade or prevent propulsion system operation). Iron nitrate formation was studied in detail by the GSFC-KSC team during the early development of the OSAM-1 mission. The following findings highlight some potential mitigation approaches for minimizing iron nitrate formation.
   - <u>On-orbit flushing [refs. 25, 26, 27]</u>: Iron nitrate has a very low solubility in NTO oxidizers. Because of this, NTO is not effective at dissolving nitrates once formed, and flowing unsaturated oxidizer over contaminated components within the system will not efficiently remove nitrates. Even in programs like Milstar where the tanks and plumbing are made of Ti 6Al-4V to mitigate iron production, thruster valves and transition joints still contain stainless steel and the production of nitrates remained an item of concern. The best solution for an active system appears to be active flushing to remove NTO with higher concentration of nitrates and replace it with NTO of lower concentration before precipitation has a chance to occur. Periodic on-orbit firing/flushing is now a standard procedure in many spacecraft, but the frequency of flushing varies based on analytical predictions, architecture and risk tolerance.
   - <u>Surface treatments [refs. 28 and 29]</u>: Passivation, pickling, and electro-polishing are all effective in increasing corrosion resistance. Passivation has been shown to have a very significant effect on flushing frequency; however, passivation by N_2O_4 alone has been shown to be ineffective. Passivation by pickling in a mixture of nitric acid (HNO_3) and Hydrofluoric Acid (HF) followed by immersion in HNO_3 has been shown to be more effective than passivation in HNO_3 alone. In a comparison of surface treatments including passivation with pickling and passivation with electropolishing, the electropolished surface showed the greatest

decrease in corrosion rate over 300 days using mixed oxides of nitrogen (MON)-1.

- Control of NTO Chemistry [ref. 30]: There are several aspects of the chemistry of the NTO/MON that can be manipulated to effect the corrosion rate and solubility of iron nitrate in the propellant.
    - In general, it is recommended that NTO be procured as dry as possible.
    - There appears to be an "optimum" water content of approximately 0.05% by weight that gives a maximum number of days between flushing. This is true because there is less corrosion with water contents less than 0.05% by weight, but this is outweighed by the lower solubility of iron, causing the iron to precipitate out of the propellant sooner.
    - Industry understanding is that NO does reduce the corrosion rate (and increase the concentration of iron necessary to saturate the propellant), leading some satellite companies to change the specification for satellite propellant to MON-10 or MON-25.
    - There is likely an intermediate value of NO and water (H_2O) percentage that produces the best trade-off between corrosion rate and solubility, but the precise "sweet spot" has yet to be found.
    - Low initial iron content of oxidizer is essential.
    - Some specific recommendations from the Milstar Program:
        - Limit initial iron content of oxidizer to less than 0.03 ppm.
        - Confirm water content of NTO to be stored for extended periods is less than 0.04% by weight.
        - Flight propulsion system NTO water content is recommended to be between 0.04 and 0.10% by weight.
- Temperature control [ref. 31]: Temperature within the system should be limited to a 9-°F temperature band and the temperature of the propellant should be maintained above 60 °F. If performing a propellant transfer, the temperature of the propellant delivered to the client should be matched as closely as possible to the propellant already onboard the client to prevent precipitation of nitrates resulting from a "cold shock."
    - At warmer temperatures, corrosion occurs significantly faster, but iron solubility is higher. If you have warmer propellant going into a cool tank, iron may precipitate from the warmer NTO into the client. Conversely, if cooler propellant from the supply tank is used, it will have less iron content but the temperature shock of introducing it into a warmer client may cause iron nitrate to precipitate out of the client's propellant. A good trade *might be* to use MON-10 as the standard propellant and keep the temperature on the higher side, say 74 °F, but this is not a specific recommendation
- Bleed fill: When loading propellants, the use of a bleed fill as opposed to an evacuation and hard fill may introduce less water into the system.

- Correct selection and configuration of valves [ref. 32]:
    - Series valves can be a problem for nitrates. The problem occurs because a pressure drop across the first valve can cause propellant saturated with high concentrations of nitrate to precipitate and then the nitrate that came out of solution can interfere with the performance of the second valve. It is recommended to minimize the number of valves in series and bear in mind that the larger the pressure drop is across a valve, the greater the potential for problems.
    - Direct acting valves should be used in preference to pilot operated valves since they are simpler, with fewer "dead volumes" having high surface area to volume ratio, and they operate at a higher temperature, which *may* be useful in dissolving iron nitrate.
    - The use of non-magnetic versus magnetic valves is preferred since iron nitrate is paramagnetic and will be attracted to the magnetic field.
- Minimize isolated volumes of propellant: Isolation of wetted components (particularly high surface area to volume ratio components) within the system ensures that nitrate concentrations are allowed to build by limiting diffusion of the corrosion products. Design and operational scenarios should seek to minimize the number of isolated volumes of propellant within the system.
- Optimal line sizing: Stainless steel lines, particularly those that will be wetted for extended periods of time, should be of the largest diameter practical to minimize surface area to volume ratio of the system.
- Titanium filters [ref. 33]: It is recommended to manufacture all filters out of titanium to reduce the localized corrosion rate of these high surface area to volume components.
- Purge interfaces: It is recommended that any wetted interfaces that are exposed to the environment be purged with dry nitrogen to the extent possible during ground processing.
- Full scale testing [ref. 34]: Full scale testing over an extended period with a flight-like system is advisable for complex, high-risk, high-value systems.

In the context of these challenges, the following subparagraphs will highlight the contributions of the three known programs that have wrestled most with the challenges of transferring conventional bipropellants in space. The first is the U.S. Propulsion Module, a MSFC development program terminated between Delta PDR and CDR. The second is the ISS Russian Segment, which has been fully operational aboard station for nearly 18 years, not to mention the previous use of analogous systems in the Russian Mir Program. The third is a research and development collaboration effort between GSFC and KSC related to a GEO satellite servicing mission of N_2H_4, MMH, and NTO.

2.2.3.1 The United States Propulsion Module (USPM) Program

Section 2.1 provided a brief discussion of the USPM in its historical context. This section will provide a brief overview of the carefully developed design features as they existed following Delta PDR before project redirection and subsequent cancellation. Although USPM did not fly, the design trades and rigorous treatment of technical risks and safety risks provide a significant case study in the challenges of an in-space propellant resupply system using MMH and NTO. The final months of the USPM Program also provided lessons from the Agency's first known human-rated development project wrestling in-depth with how high-pressure, high transfer efficiency helium would be resupplied in space.

As a backup system for the Russian elements, the USPM would have launched in the Space Shuttle Orbiter payload bay with repurposed Orbiter OMS/RCS propellant tanks loaded with NTO and MMH. The USPM would be attached to a docking port on a node at the forward end of the ISS, since the aft end (i.e., the ideal location for reboost thrusters) had been reserved for the Russian elements. The USPM would also block a docking port; therefore, it would have to provide a new location for Shuttle docking and crew access to the ISS through an integrated tunnel. The propulsion module would be left on orbit for a period of several years (15 as a goal) with no plans for return to earth for maintenance. It would be delivered to the ISS with as much propellant as the launch manifest and tank volumes would allow and would be resupplied with liquid NTO and MMH through a set of fluid couplings with design heritage to the MSFC AFIS coupling prototypes.[4] The Orbiter fleet would be modified to connect aft and forward RCS systems and to allow transfer of excess OMS/RCS propellants from the Orbiter to the USPM when docked to ISS. During ISS reboost operations, the USPM would be required to provide propulsion for an ISS 180-degree yaw maneuver to place the propulsion module's thrusters at the aft end of ISS prior to and following reboost. Using Aerojet R-42 thrusters (200 lb_f at vacuum), the reboost would have required constant firing for a period of hours, with possible coast periods.

The USPM development effort shed light on interesting technology gaps and design challenges. While many were solved by the time of the Delta PDR, the following risks and concerns were still challenging designers at the time of termination:

1. As the design matured, it became apparent that the mission-critical nature of the automated fluid coupling would require levels of redundancy and/or design robustness that would drive cost and schedule unacceptably. As a result, the NASA/Boeing team decided to eliminate drive motors and automation, designing the system to operate from an internal hand-crank to be operated by a crew member. Future designers will want to consider human rating and mission-success requirements early in the design of automated fluid coupling systems such as the CSV developed for OSAM-1 and licensed to American industries discussed in Section 2.2.2.
2. Venting of PMD tanks in microgravity prior to refilling presented two significant challenges:
    a. As explained in Section 2.1, the frequent venting of the oxidizer tanks would have led to NO depletion in the NTO tanks, heightening the risk of stress corrosion cracking. Designers planned to solve this problem by either (1) adding NO to the propellant tanks in flight (which could have caused NO gas evolution in liquid fed to the thrusters, risking combustion instability) or (2) reducing or eliminating overboard ullage venting by recovering and recycling ullage gases (a complex subsystem that had not yet been designed).
    b. Ullage venting, even under low-G accelerations, could allow liquid propellant to migrate toward the vent, undesirably expelling propellant. As explained in Section

[4] The reader will recall that the AFIS design provided the starting point for the Orbital Express automated fluid coupling, which flew about 7 years after NASA terminated USPM development.

2.1, expelling liquid could lead to negative impacts on the propellant inventory and damage to optical surfaces, solar arrays, and EVA equipment. Thus, future designers will seriously want to consider firmly settling propellants with thruster firings during preparation for transfer operations, if propellant budgets can allow this. A more elegant solution would be the development of an effective liquid/gas separation device that can reliably retain liquid inside the tank while withstanding the variable delta pressures associated with the venting operation. Work on such a device was pursued during the early and mid-2000s timeframe at Texas A&M by Dr. Fred Best and a graduate student, Mr. Bich Nguyen, who had worked as a designer on the USPM Program at Boeing.

3. Transfer gauging accuracies and quantity gauging in the receiver tanks represent another issue. The risk of overfilling tanks and the constraint of fitting the transfer into 8-hour astronaut work shifts drove the need for accurate knowledge of the quantity of propellant in the receiver tanks as the transfer progressed. These flight tanks would not have the luxuries of ground loading procedures involving fill-to-spill approaches, flow metering, or tank weighing as discussed in Section 1.2.2. Internal capacitance probes had proven troublesome in the Space Shuttle Program and would only work when propellants were settled. With the flight-capable designs at the time, this situation left the designers with the option of pausing for tank equilibration and use of the PVT method or using some kind of flow integration scheme. Since the transfer operation would be driven by pressurizing the supply tanks and since the schedule and funding did not allow for development of electric transfer pump, pump speed integration was ruled out.[5] Following the USPM Program, an oxidizer pump was developed for potential use on a GEO servicing mission as discussed in Section 2.2.3.3. At the time of the USPM development, the design of a reliable flight-rated, long-life flow meter was deemed too risky, eliminating the desirable option of flow rate integration; however, major strides have been made on hypergolic propellant flow meter technology as discussed in Section 2.2.2.
4. Ullage heating due to compression during tank fill operations continued as a concern, particularly due to the astronauts' 8-hour shift requirement. While the ORS experiment mentioned in Table 3 mitigated the concern for N_2H_4 tanks, concern for the NTO tanks was heightened by the oxidizer's high vapor pressure and low boiling point, making it a more sensitive set of thermal interactions. Given the lack of convective heat transfer in the vacuum environment, this concern was on the watch list for potential violation of operational timelines as discussed in Section 2.2.2.
5. The automated fluid coupling presented three primary technical concerns, one of which had been solved before the project was terminated. First, while confidence in the coupling design itself was high, application of human-rating rigor to the electric drive motors that would mate and retract the coupling halves was problematic. The mission-critical nature of the transfer operation drove the designers to increase motor size and add redundant motors to the extent that a more elegant (if primitive) design solution was chosen: An astronaut would mate and demate the coupling halves using a mechanical drive system with a hand crank located inside the habitable volume. Second, the risk of

[5] The reader will recall that Orbital Express later developed and demonstrated a pump for hydrazine derived from Shuttle and ISS pumps, using a pump-speed integration scheme that fell short of gauging accuracy requirements (see Section 2.1). Application of this pump heritage to NTO systems remains a design challenge worthy of advanced development or technology development funding.

nitrate formation on the flight half seals threatened to lead to leakage the whole time the couplings were mated as discussed in Section 2.2.3. This phenomenon is well known in long-life spacecraft thrusters, but relying on a coupling seal for vital propellant resupply placed this nitrate formation on valve seals in a new spotlight.[6] Third, given the variation in fluoropolymer formulations over the years (known variations exist, affecting seal life) and given the long duration over which the couplings' flight halves would have been in space with no maintenance, the design team had valid concerns about seal life. Future designers facing long mission life requirements should consider early testing of candidate seal materials, since some heritage polymeric seal materials may not be available or feasible in the particular environment. Testing of some alternative materials is discussed in Section 2.2.3.3.

6. Finally, a major risk facing the USPM at the time of program termination was the transfer inefficiency of helium resupply since a flight-qualified (or reasonably flight-qualifiable for long term use in the space radiation environment) helium compressor was not available. Pressure-driven transfers would have been extremely inefficient, since the supply tank and receiver tanks had similar pressure ratings and the difference in driving pressures could only force a small amount into the receiver tank. Hence, the baseline option was a resupply pallet with helium tanks that would be connected to the USPM. Structural interfaces and launch requirements would have driven this pallet to require about 2000 lbm of dry up-mass and would only have transferred about 100 to 200 lbm of helium. Future designers will want to consider approaches to a more elegant solution that increases mass efficiency. An obvious option would be the development of a flightworthy helium compressor and a thermal design that effectively addresses the challenge of receiver tank cool-down pauses when charging the tank in a vacuum (non-convective) environment. This new development would not only benefit a potential future in-space GHe transfer, but would also likely be evolvable to other pneumatic-type transfers such as GN_2 and Xenon. Unique material properties such as soft good expansion with Xenon use application would need to be carefully studied for each application.

Admittedly, different mission requirements could eliminate some challenges that USPM faced. Missions that allow completely autonomous operation could escape the constraints of an 8-hour work shift, unless other mission requirements such as sunlight exposure (or lack thereof) drive certain timing constraints. Shorter mission durations could mitigate concerns about flight-half coupling seal life. A generous propellant budget and ample thruster life could enable settling of propellants, eliminating the gas-liquid separator concern. A mission that involves transfer in a gravity field on the surface of the moon or another planet could have the same benefit. The numbered list above simply illustrates the nature of the technology gaps and design challenges that one very serious, high-priority development program encountered. It is offered here to sensitize future design teams to the need for early questioning of driving requirements and thorough, early-as-possible assessment of how a proposed transfer system will be operated.

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[6] It is noteworthy that this concern on the Russian segment of ISS is dwarfed by the routine spilling of a large quantity of propellants (anecdotally on the order of 2 liters per mate/demate). The AFIS coupling was designed from its earliest stages with an emphasis on low spill volumes. At the date of this writing, no known testing has been done, however, to evaluate the effects of nitrate formation on AFIS seal integrity over long periods of time.

2.2.3.2 The ISS Russian Propulsive Elements

Basic information about the design of the Russian propulsive elements and the propellant transfer systems is available in the public domain, and that information will be summarized here.[7] Since the details of the Zarya, Zvezda, and Progress modules are proprietary, NASA has compiled a separate internal document, "Russian Propulsion & Propellant Resupply Systems for the ISS: Technical Description" (JSC-27979) that is available on a restricted, need-to-know basis from the Propulsion and Power Division (EP) at NASA JSC.

Public domain information indicates that all three elements (Progress, Zvezda, and Zarya) are able to store UDMH and NTO. Although Zarya was initially used to provide thrust for reboost, it is only used for propellant storage now. Zvezda stores UDMH in four 200-L (52.8-gal) propellant tanks, two for each commodity. Zvezda uses nitrogen to pressurize the propellant tanks. Use of highly soluble nitrogen is possible due to the use of positive expulsion (metal bellows) tanks aboard the Zvezda and Zarya modules. These bellows provide a solid barrier between pressurant and propellant, preventing nitrogen dissolution in the liquid and potentially causing destructive combustion instabilities downstream. The Progress module contains diaphragm tanks that are only capable of supplying (not receiving) propellant, suggesting the use of metallic diaphragms, since polymeric diaphragms are nearly always reversible.

The overall success of the Russian segment is well established. As of May 2019, the Russian segment has provided 68 Progress resupply missions and has transferred nearly 39,000 kg of propellants. The European Automated Transfer Vehicle (ATV), using the Russian Androgynous Peripheral Attach System (APAS) system, has additionally transferred over 4,000 kg of propellant (see Figure 1).

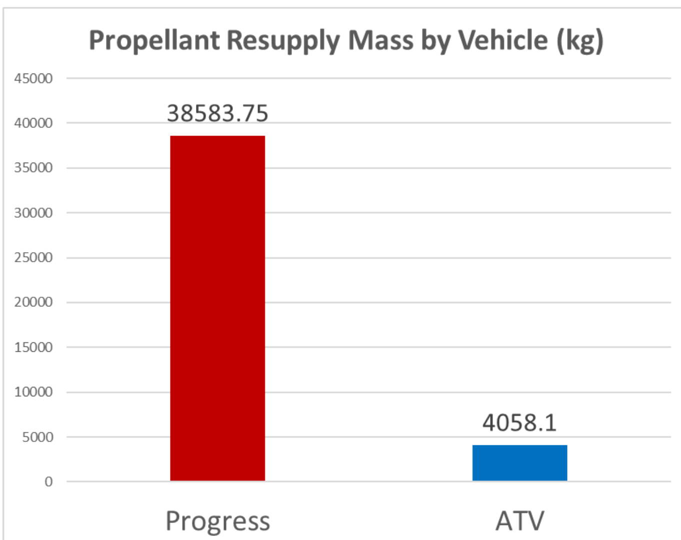

Figure 1. Propellant Resupply Mass to ISS by Vehicle

Informal communication with the cognizant JSC Propulsion engineer indicates that the Russian propellant transfer system has had a few on-orbit anomalies and concerns that do not align with goals of the challenges of reliable long-life propellant resupply systems. The compressors did not

[7] For public domain information, see, for example, page 54 in "Reference Guide to the International Space Station," published by NASA, dated September, 2015. Accessed March 30, 2020 at the following web address: https://www.nasa.gov/sites/default/files/atoms/files/np-2015-05-022-jsc-iss-guide-2015-update-111015-508c.pdf

work as expected on a few occasions. On a few occasions, valves were incorrectly configured, resulting in propellants going to other than their intended destinations. One Progress module had thruster failures during rendezvous that were attributed to nitrogen bubbles in the line due to a cracked diaphragm in the propellant tank. And, finally, echoing the USPM concerns about preventing overfilling the receiver tanks, there have been concerns about overfilling the metal bellows.

As cautioned in Section 2.1, the success of in-space propellant transfer among the Russian elements, while vital to ISS and remarkable in its own right, should not be mistaken for a universal approach that is suitable for all mission scenarios. Overboard spills, reliance on Russian metal bellows suppliers, and fatigue-life issues restricting bellows limited cycle operation are potentially problematic in other applications.

2.2.3.3 Research and Development for Geosynchronous Orbit Propellant Servicing Mission

Prior to the OSAM-1 mission from 2010 through 2016, a multi-center team worked to develop architecture, operational concepts, and technologies to enable the servicing of multiple geosynchronous satellites with MMH, NTO, or N_2H_4. Since NTO is the most challenging commodity for an in-space transfer, several components were studied to enable a NTO servicing with plans to adapt them to fuel at a later date. The components that were developed included seal-less transfer pumps, an all titanium bellows accumulator tank, a quick disconnect to use on existing FDVs, and an array of flow meters.

The pumps proved to be the most challenging to develop. A commercial company leveraged their design from a USAF Space-Based Infrared System (SBIRS) contract to meet the new set of requirements such as seal-less, positive displacement, magnetically driven, and usage for NTO service. The first design iteration was about the size of a roll of quarters and proved the concept; however, it showed performance degradation after a couple dozen hours of runtime with NTO. The second iteration was significantly improved and did not show any appreciable performance degradation after 70+ hours of runtime with NTO [ref. 35]. Following the development and testing, the pump design was developed through the Small Business Innovation Research (SBIR) Program, to evaluate the pump’s capability to feed propellant to a bi-propellant thruster. Further improvements were made to the design and a successful hotfire test was performed [ref. 36].

Following the second design iteration of the pumps, a propellant transfer assembly (PTA) was designed and tested with a NTO simulant fluid (Freon-113) and then with NTO. The test included a remote robotic mating, to a simulated client FDV. NTO was then transferred across this interface [ref. 37]. The system included the pumps, an all titanium bellows accumulator to enable the manipulation of system pressure, and a flow meter among other standard propellant system components like valves, pressure transducers, etc. Prior to the PTA testing, many of the components were tested individually, but it was unknown how they would perform when all working together within a transfer system. Many different scenarios were tested which resulted in an accumulation of lessons that were used for the development of the OSAM-1 mission.

Another key component, the bellows accumulator, was determined to be an integral part of the PTA since it provided features such as liquid pressure manipulation, small propellant transfers (approximately 1 L), pump startup and shutdown transient pressure fluctuation absorption, priming surge pressure absorption, and thermal pressure transient absorption. A commercial company was contracted to design an all titanium alloy bellows accumulator with a 2-ply bellows assembly to prevent leakage due to fatigue cracking. The commercial company successfully built and provided three models to be used for testing with Freon-113 and NTO. System level testing with the PTA was successful and specifically the cycle life of the bellows was far exceeded without any issues [ref. 38].

Another effort undertaken for this proposed geosynchronous satellite servicing research and development mission was to carefully study and test key materials. The material compatibility and mechanical degradation due to propellant exposure would need a deep understanding to allow for their usage with N_2H_4, MMH, and NTO. Over a three and a half year period, 23 separate materials were tested, using a modified version of the NASA STD 6001 Test 15 method. Table 5 lists all the materials tested and their summarized results. These material tests were critical in the development of several key components in the PTA. Further assembly level testing is highly recommended if any of these materials are chosen for use on future missions.

Three key materials that are typically used in commercially available propellant servicing valves were also studied in detail. The area of interest was specifically how they would hold up after decades of propellant exposure and then subsequent cycling on-orbit. Accelerated aging was performed followed by basic mechanical property testing such as a tensile strength tests to study how the materials would hold their original shape when cycled within a servicing valve after many years of exposure to the space environment. The testing showed that there was minor strength reductions for the fuel materials; however, the NTO materials showed enough degradation that further study is required [ref. 39].

Table 5. Materials Tested for Geosynchronous Propellant Servicing Component Research and Development

Material	Type	Preliminary Results
Daikin M-111 (mod. polytetrafluoroethylene (PTFE))	Non-Metal	Passed
Daikin M-112 (mod. PTFE)	Non-Metal	Passed
Graphite AG21PT	Non-Metal	Passed
Silicon Carbide (SiC)	Non-Metal	Passed
Zirconia	Non-Metal	Passed
Graphite grade 114	Non-Metal	Failed
Samarium Cobalt (SmCo)	Non-Metal	Failed
Chemraz 514 (perfluoroelastomer (FFKM))**	Polymer	Passed
Chemraz 584 (FFKM)**	Polymer	Passed
Simriz 486 (FFKM)**	Polymer	Passed
EPR 515-80 (MMH)	Polymer	Passed
EPR 1267-80 (MMH)	Polymer	Passed
E0-515-80 (N_2H_4 & MMH)	Polymer	Passed
455 Stainless Steel	Metal	Passed
420H Stainless Steel	Metal	Passed
416 Stainless Steel	Metal	Passed
Pyrowear 675	Metal	Passed
Nickel	Metal	Failed
Nickel (N_2H_4)	Metal	Failed
Nickel (MMH)	Metal	Passed
302 Stainless Steel (N_2H_4)	Metal	Passed
440C Stainless Steel	Metal	Passed

*All tests are with NTO unless noted

**Conditionally passed (swelling similar to other FFKM; Kalrez 1045)

2.2.4 Key Design Trade-Offs and Lessons Learned for Storable Propellant Transfer Systems

As the foregoing examples have indicated, the technical risks facing any proposed in-space storable propellant transfer system are intertwined with a complex set of mission requirements, architecture decisions, and operational approaches. When propellant transfer experts have briefed study teams and program decision makers, the intricacy has produced frustration and bewilderment. To help a new cadre of engineers prepare for these complex conversations in the future, this section will identify a set of questions that express the key interdependencies. The objective of these questions is to prompt discussion of areas where simpler designs can decrease technical, cost, and schedule risks for the project from the outset. As with all sets of networked interdependencies, this list should be tailored to consider the unique situation each project will face.

1. Is there an option to treat the propellant and pressurant tanks (or the entire propulsion system) as an on-orbit replacement unit (ORU) so that resupply can be accomplished through ORU changeout rather than fluid transfer? Although, for certain cases, this approach could deviate from the normal good design practice of main propulsion storage tanks being at or near the center of gravity of the spacecraft. If possible, even for low mass commodities such as pressurants, is a grappling arm or similar resource available to enable this changeout? This approach has limited usefulness, but using it would altogether avoid several of the risks and design challenges of a fluid transfer system, if applicable to a particular mission.
2. If the project cannot use an ORU change-out approach for propellant resupply, the chief engineer should immediately charter a lean, expert, multi-disciplinary team led by a propulsion expert an involving thermal, power, ground operations, flight operations, and flight safety to develop and maintain detailed operational approaches for the proposed propellant transfer system as the design concept matures. Operational impacts of the physical, chemical, and thermal properties of the liquid propellants and gaseous pressurants have been a constant source of challenges and previously undiscovered risks. This “propellant transfer working group” should remain intact throughout the development phase and should assist during the transition to operations.
3. If a bipropellant system is proposed, could the project accept the weight and performance penalty of using a monopropellant N_2H_4 system instead to simplify transfer and claim strong heritage to the Orbital Express and OSAM-1 missions rather than incurring the risk mitigations likely required for a system using NTO oxidizer?
4. If the project is willing to baseline monopropellant N_2H_4 and the use of positive expulsion tanks (e.g., polymeric diaphragm in the receiver tank), can the project fund early advanced development to qualify a highly reliable and appropriate nitrogen or helium compressor device to avoid overboard dumping of this resource? The ISS Program should be consulted for heritage designs of oxygen and nitrogen compressors used for environmental control and life support.
5. If receiver tanks must be vented overboard (i.e., there is no ullage recovery and recompression capability on board), pressurant gas will likely be depleted over time, introducing additional challenges. The project will need to choose among several options: (a) initially oversizing the pressurant tanks to accommodate the entire mission life or (b) investing in a capture system with a flight-qualified (and likely new) compressor design

that would benefit from advanced development funding or (c) planning for a pressurant tank change-out capability requiring a grappling arm or similar device or (d) living with the timeline pauses and general inefficiency of a pressure equalization approach (i.e., recharging a partially expended tank from a higher pressure tank until pressures equalize and flow stops, leaving a large residual in the supply tank).

6. If a bipropellant system must be used, implying the use of PMD tanks at least on the oxidizer side, can the project afford to settle the propellants (by firing thrusters) during receiver tank venting? If not, the project should invest early in a liquid/gas phase separation device to protect against overboard venting of propellants (especially NTO). NOTE: The use of positive expulsion devices or appropriately designed PMDs should enable the post-venting propellant transfer operations to proceed without the need for settling. In fact, settling during transfer would likely be a waste of propellant. Some missions, however, may choose to carry propellant in reserve to enable settling during transfer in the remote possibility of PMD or bladder/diaphragm failure.
7. What constraints, if any, exist on the timeline available for propellant transfer operations? Does the operation have to fit within a single astronaut work shift? If there are constraints, can these be mitigated by shift work or other means to assure ample time is available for receiver tank cool down, tank equilibration for PVT gauging readings, etc.?
8. Since overfilling the receiver tank could lead to spills, overpressures, ruptures, etc., can the project invest early in device(s) that would enable better transfer gauging, such as electric pumps (for pump-speed integration calculations) or highly reliable flow meter development (for flow rate integration)?
9. Can the project tolerate conventional system-level gauging errors on the order of 5 or 6%? If not, the engineering staff should insist on early advanced development funding (or better yet, advocate for maturation and tailoring of advanced gauging methods such as the NASA GRC radio frequency mass gauge (RFMG)). Note: OSAM-1 has qualified a turbine flow meter for on-orbit use with a single-phase incompressible fluid (N_2H_4 or hydrazine family storables) that will be demonstrated for use at its qualified accuracy of <1% error for the mission.
10. Does the project have a multi-year mission life requirement coupled with a requirement for multiple propellant resupply cycles with in-space radiation exposure? If so, the engineering staff should immediately begin investigating potential seal life issues or a seal-less design for the automated fluid couplings, especially the flight half of the coupling. This evaluation should include consideration of robust seal material availability for formulations with a history of long-duration exposure to the projected environment (vacuum, Martian atmosphere, radiation, etc.). This evaluation should also consider the likelihood and impact of nitrate formation on the seal surface and the impacts of leakage on adjacent hardware. Also, reference OSAM-1 fill and drain valve seal interface testing with deliberate contamination showing leak rates [refs. 40 and 41].
11. Is the scale of the proposed system in line with existing component designs, especially the automated fluid couplings? Consideration should be given to leverage from past projects such as Orbital Express and OSAM-1 CSV (https://technology.nasa.gov/patent/GSC-TOPS-170) If not, the project should strongly consider investing early in advanced development of critical and long-lead components for the propellant transfer system. Also, note that even typical heritage components such

as pressure transducers, latch valves, etc. can become major long leads of 2 to 3 years if any unique requirements require one-off qualification testing (or approval of qualification by similarity). Government procurements can be minor secondary tasks for most major aerospace component providers who make the majority of their profits from larger customer, bulk buy orders, which typically gain priority.

12. Is a fully autonomous transfer required? If so, the project should consider investing in advanced development of the drive motors and drive mechanisms required to meet safety and mission assurance requirements. There is no known human-rated or high-value-asset design heritage for the drive motor assembly at the time of this writing. Motor redundancy could be required, inflicting significant mass and perhaps power penalties.

2.3 Cryogenic Propellant Systems

2.3.1 General Highlights for Cryogenic Oxygen, Methane, and Hydrogen Transfer Systems

NASA has invested in and implemented cryogenic fluid management (CFM) technologies, since its earliest days. These investments enabled the use of LH_2 and LO_2 on Atlas and Saturn V and the use of LO_2 and LH_2 for the Shuttle Main Propulsion System. Apollo flew with a boil-off rate of 30% per day, so that the S-IVB stage was only able to maintain its residual propellants for a few hours in LEO. More recent upper stages have succeeded in increasing that number of hours, but long-duration storage of subcritical cryogenic liquid propellants has only been demonstrated on the ground. The only flight of a newtonian cryogen was RRM3, which stored LCH_4 for 4+ months on ISS. Cryogenic component technologies have been demonstrated on the ground in partially integrated systems using prototype and some flight-relevant hardware. For most of NASA's history, these CFM investments in long-term storage and transfer have been conducted within NASA's small but highly expert CFM technologist community under technology maturation projects.

Superfluid helium was transferred between tanks on the Superfluid Helium On-Orbit Transfer experiment on STS-57 in June 1993 [ref 42]. However, many of the technologies demonstrated on the flight were tailored to superfluid helium's non-Newtonian fluid properties.

2.3.2 Cryogenic Propellant Storage & Transfer (CPST)

The CRYOSTAT or CPST Project early in the last decade represented a major shift re-emphasis by NASA in the development of a flight demonstration unit to provide an integrated on-orbit characterization and demonstration of long-term cryogenic storage and transfer. Although cancelled due to budget constraints and converted into a ground test named Evolvable Cryogenics Project (eCryo), the blending of technologist and flight design expertise produced highly significant progress in preparing the cryogenic transfer technologies for flight infusion.

2.3.3 Robotic Refueling Missions

The agency's most recent cryogenic payload named the RRM3 hosted a cryogenic dewar with 42 liters of methane launched to ISS to increase the understanding of cryogenic fluids in the microgravity environment as NASA prepares to embark on exploration missions. The RRM3 payload was designed to demonstrate the zero boil-off storage of methane maintained to 92 to 100K, transfer of LCH_4 to a secondary dewar via no vent fill, and freezing the methane. The RRM3 payload was launched on Commercial Resupply Services (CRS)-16 on December 5, 2018 (following 40 successful days of ground storage). On April 11, 2019, after over 4 months of zero boil-off operation on ISS, a payload electrical issue inhibited the use of the dewar cryocooler to maintain the methane at cryogenic temperatures and the methane was safety vented overboard. During the operations on orbit, the eCryo project successfully demonstrated the Radio Frequency Mass Gauge utilizing the RRM3 cryogenic dewar [ref. 43].

2.3.4 Most Significant Accomplishments

In the mid-1980s, the list of experience and technology gaps related to long-duration in-space storage and transfer of cryogenic propellants was formidable. Since that time, a series of NASA investments have closed most of these gaps. The following list provides a sampling of the most significant accomplishments.

- Development
    - PMD screens appropriate for LH_2, LO_2, and LCH_4
        - Needs in-flight demonstration to expand beyond 1-g test results to demonstrate robustness of design, integrated flow rates, provide correlation to predictive modeling tools, etc.
- Ground Testing and Demonstration
    - Helium pressurization for hydrogen, methane, and oxygen (with quiescent, settled propellants)
    - Thermodynamic Vent System (TVS)
    - Tube-On-Tank broad area cooling (BAC) with LH_2, LCH_4, and LN_2 (as a simulant for LO_2)
    - No-vent tank fill (needs demonstration in zero-G)
    - Transfer-line chill down (needs demonstration in zero-G)
    - Propellant tank chill-down procedures with LH_2, LCH_4, and LO_2
    - Autogenous pressurization approaches for cryogenic liquids
    - Vapor cooling of a structural skirt with hydrogen and nitrogen boil-off vapor
    - Low-conductivity flight-relevant tank mounting struts that increase thermal isolation of the tank.
    - High performance multilayer insulation systems for performance with
    - Settled propellant gauge Early prototype liquefaction systems simulating the condensation and storage of LO_2 and LCH_4 produced by an ISRU propellant plant
    - With cryogens and suborbital demonstration (with referee fluids) of RFMG
- Flight Demonstrations
    - Autogenous pressurization using a wick and heater technique was demonstrated on RRM3
        - Orbital use of wick heating caused bubbly flow when the wick over heated locally and caused boiling in the surrounding liquid.
    - Integrated Multi-Layer Insulation (IMLI) was successfully flown on the Green Propellant Infusion Mission (GPIM) and RRM3.
        - Insufficient data was gleaned to determine performance.

- Cryogenic fluid mass gauging techniques demonstrated in a microgravity environment through use of a RFMG.
    - Gauging uncertainty was 2% at 2 standard deviations at the single fill level RRM3 achieved.

These accomplishments have spanned decades under fluctuation and usually insufficient funding streams, yet these payoffs have placed NASA on the threshold of fielding new, higher-performing in-space cryogenic propulsion systems coupled with an in-space resupply capability, if the few remaining system-level gaps can be closed.

When viewed in context of this broad set of accomplishments, it is clear that the few remaining challenges for implementing a cryogenic in-space propulsion systems and transfer systems represent merely the final steps of a long journey toward implementing these high-payoff mission capabilities.

2.3.5 Remaining Challenges for In-Space Cryogenic Propellant Transfer

Present and future NASA projects are beneficiaries of decades of cryogenic technology maturation supporting in-space cryogenic storage, transfer, and their complementary long-duration propulsion systems [ref. 44]. Table 6 is from the CFM technology roadmap study, circa 2018. This list is a subset of space cryogenic and cryogenic fluid management technologies that are most likely to contribute to the success of the NASA and commercial exploration Lunar architectures in the near term, and ultimately crewed Mars exploration.

Table 6. Remaining Technology Gaps for Initial In-Space Cryogenic Propellant Storage and Transfer Capability

#	Technologies	Current TRL	Gravity Dependent (Y/N)	Path to TRL6	"Cross Cutting" or "Fluid Specific"
1	Low Conductivity Structures	6	No	Ground Test	Cross Cutting
2	High Vacuum Multi-Layer Insulation (MLI)	5	No	Ground Test	Cross Cutting
3	Tube-in-Shield Broad Area Cooling	5	No	Ground Test	Cross Cutting
4	Valves Actuators & Components	5	No	Ground Test	Cross Cutting
5	Vapor Cooling	5	No	Ground Test	Fluid Specific
6	Helium Pressurization	5	Yes	Flight Demo	Cross Cutting
7	Line Chill down	5	Yes	Flight Demo	Cross Cutting
8	Pump Based Mixing	5	Yes	Flight Demo	Cross Cutting
9	Thermodynamic Vent System (TVS)	5	Yes	Flight Demo	Cross Cutting
10	Tube-on-Tanks Broad Area Cooling	5	Yes	Flight Demo	Cross Cutting
11	Unsettled Liquid Mass Gauging	6	Yes	Flight Demo	Cross Cutting
12	Liquid Acquisition Devices	5	Yes	Flight Demo	Fluid Specific
13	Advanced External Insulation	3	No	Ground Test	Can be Both
14	Automated Cryo-Couplers	4	No	Ground Test	Cross Cutting
15	Cryogenic Thermal Coating	4	No	Ground Test	Cross Cutting
16	High Capacity, High Efficient Cryocoolers 90K	3	No	Ground Test	Cross Cutting
17	Soft Vacuum Insulation	3	No	Ground Test	Cross Cutting
18	Structural Heat Load Reduction	3	No	Ground Test	Cross Cutting

#	Technologies	Current TRL	Gravity Dependent (Y/N)	Path to TRL6	“Cross Cutting” or "Fluid Specific"
19	Propellant Tank Chill down	4	Yes	Flight Demo	Cross Cutting
20	Transfer Operation	4	Yes	Flight Demo	Cross Cutting
21	High Capacity, High Efficient Cryocooler 20K	3	No	Ground Test	Fluid Specific
22	Liquefaction Operations (MAV & ISRU)	4	No	Ground Test	Fluid Specific
23	Para to Ortho Cooling	4	No	Ground Test	Fluid Specific
24	Propellant Densification	4	No	Ground Test	Fluid Specific
25	Autogenous Pressurization	4	Yes	Flight Demo	Fluid Specific

The table is provided as guidance and should be subject to critical evaluation for each situation. The table is also not exhaustive in terms of all CFM technology gaps. There are other technologies worthy of investigation and investment, as they could enhance performance or robustness for next-generation systems.

Fielding initial capabilities for cryogenic storage and transfer will, however, require judicious requirements definition and appropriate early risk mitigation to deploy these capabilities at an acceptable level of risk.

3.0 Electric Propulsion Systems (Xenon)

In the decades-long consideration of various in-space propellant transfer scenarios, the transfer of Xenon for electric propulsion systems is a relatively recent discussion. One option is to store the Xenon as a supercritical gas and allow pressure equalization between the supply tank and the receiver tank, possibly followed by use of a compressor to push additional Xenon into the receiver tank. One of the main issues in the transfer of Xenon, as with any high-pressure gas or pressurant is the dissipation compressive heating from the receiver tank while adding heat to the supply side as the pressure decreases. Additionally, Xenon has unique fluid compatibility issues with multiple materials. Gilligan and Tomsik (both NASA GRC) provide details of Xenon tank pressurization in a NASA Technical Memorandum [ref. 45].

Future deep-space missions will likely utilize large quantities of extremely high purity xenon, and the capability for high efficiency in-space refueling of such spacecraft is highly desirable. The rate of xenon transfer relies on the ability to model the xenon fluid mechanics in the microgravity environment over a wide range of temperatures and pressures and the ability to reject heat to determine an acceptable (scalable) and safe flow rate. An on-orbit demonstration of xenon transfer would validate the 1-g findings and microgravity predictions with respect to the fluid and thermodynamic models.

4.0 Closing Remarks

Following this lengthy discussion of the range of envisioned in-space propellant transfer scenarios, a few summary remarks may help to distill the important points of this overview.

It is clear that work in recent years has brought propellant transfer much closer to flight readiness than was the case when our present senior engineering team members were beginning their careers. The Orbital Express flight demonstrations brought monopropellant N_2H_4 fluid transfer within the reach of new programs needing that capability, although there is still room for next-generation improvements in propellant flow measurement and gauging accuracies along with robotics and servicing tools for specific clients. Early development work for the OSAM-1 project and the cancelled USPM Program brought the needs and risks of a storable bipropellant system into clear focus. Core needs such as liquid-gas separators for tank venting, more accurate next-generation gauging methods, reliable, long life, leak tight, qualified transfer couplers for oxidizer, and helium compressors are now clearly identified and ready for technology maturation investments or advanced development funding from flight development programs. Decades of investment in relevant CFM technologies have brought them to the point of readiness for advanced development of non-gravity-dependent components and for an integrated system-level flight demonstration. System demontration in the relevant flight environment will mature the full suite of technologies to technology readiness level (TRL) 6 and validate analytical design tools, opening the door to deployment of initial in-space cryogenic transfer. This initial capability represents another step toward a future system of systems that gains the efficiencies of ISRU propellant production.

This paper has attempted to capture relevant knowledge, insights, and lessons learned (along with reference to detailed backup materials) to assist future NASA engineering teams and decision makers to maximize previous investments and experiences and to make pragmatic compromises to enhance the likelihood of success within the cost and schedule constraints of future missions.

This paper has also advocated the continued drive to move boldly forward in the pursuit of new in-space propellant resupply capabilities, with a walk-before-we-run philosophy. Hence, several sections have mentioned "initial capability" (i.e., walking) without pursuing every possible enhancing technology (i.e., those things that could help us run). When these are adequately distinguished, it is the sense of the authors that initial capabilities are now within reach, if decision makers are well informed of actual risks and pragmatic compromises are made to eliminate unnecessarily risky design features when a simpler, perhaps lower-performance option may be more achievable.

Despite this emphasis on pragmatism, it must be stated that continued investments in next-generation enhancing technologies should still be pursued, particularly where these improvements are aimed at increasing the performance or robustness of future systems. Some clear examples include improving gauging accuracies, eliminating waste due to venting, capturing and recompressing pressurant gases, and developing high-efficiency multi-use positive displacement tanks for NTO.

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