Direct Earth/Moon Cargo Delivery

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ABSTRACT

As the space exploration initiative matures, the fundamental need for effective and efficient heavy-cargo logistical delivery means across space will become increasingly apparent. Accepting human lunar exploration as the initial step toward more ambitious missions to follow, early on it is evident that a number of different cargo-carrying vehicles will incrementally operate across the mission profile, requiring several in-flight payload transfer operations. When later stages of long-duration exploration and lunar-base build-up are underway, a more effective logistical arm may be needed. A leading-candidate approach is the “direct delivery” of Earth-launched cargo to the Moon. Here, by way of example, a single large vehicle to be described is operated across the total mission profile with no intermediate cargo transfers. This two-stage system consists of a solid-rocket Boost Stage and a cryogenic liquid-rocket Space Stage. In addition to multi-ton discretionary payloads being so delivered, a comparably sized collateral payload arrives at the Moon: the Space Stage transporter itself. This landed stage would be specially designed to be highly useful, in various ways, to the awaiting human exploration teams. It would be used directly, as a shelter, habitat, storage facility and, indirectly, as for pre-planned disassembly for materials of construction, power systems hardware, instrumentation equipment, and the like. In the longer term, given in situ produced cryogenic propellants, an intact Space Stage might return certain high-value cargo to Earth. As discussed, acquiring these unique direct-delivery benefits would productively challenge the propulsion and structural-design state-of-the-art, no doubt stimulating a host of innovative design ideas. Several examples are included here. Were such an Earth/Moon cargo transporter to be redirected to ubiquitous ETO service, an impressive super-heavy-lift capability would be made available.

I BACKGROUND OF THIS PAPER

The authors’ company, SAIC, continues to actively participate in systems analysis and mission-concept formulation and review aspects of NASA’s human and robotics space exploration initiative. SAIC is one of several contractors participating in study contracts awarded by NASA addressing the multi-spiral exploration mission design and those technological requirements involved. Special studies addressing the constituent enabling and enhancing technologies associated with the initiative, and their suggested prioritization from a programmatic synthesis standpoint, have been conducted by the company for several involved NASA centers.

Having been encouraged, in conducting this work, to think innovatively and to examine out-of-the-box mission and system ideas and concepts, several of us in the spring of 2004, in taking the longer view of space exploration, examined a “special corner” of the potential exploration mission design space, one apparently not being heeded by other researchers at that point. This attracted our attention and the subject was casually worked over about three months’ time. This novel mission approach focused on the far-term lunar cargo delivery challenge, with emphasis on meeting large-mass, highly diversified logistical requirements as might be established for more complex, time-extended lunar exploration ventures, including lunar base development.

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The basic idea resulting from this line of investigation was the transporting of relatively massive and diversified payloads on a single transportation vehicle flying directly from the Earth to the Moon. We have termed this the “Earth/Moon Direct” cargo delivery mission approach. It is in rather sharp contrast to the more-conventional multi-step incremental-cargo delivery approach, with its need for a multiplicity of vehicle types and several in-flight cargo-transfer operations. The “direct” delivery approach requires but one vehicle type, not more, and it eliminates the need for multiple in-space cargo transfers from one vehicle type to another. These special attributes signify numerous opportunities to reduce mission complexity, increasing reliability and possibly astronaut safety, and to very likely reduce life-cycle costs in the long run.

But these potential benefits come at a very substantial engineering challenge, particularly as it might be viewed in a conventional-thinking mindset environment. First off, the focus on large-mass cargo, in view of the high mission energetics involved, would require a very large transportation vehicle, one with quite ambitious rocket-system performance and structural efficiencies. Achieving these unusually high-level technical characteristics, while keeping the vehicle size (mass) in bounds, would call for some highly innovative design measures. Such, at least in-part, had been introduced into the system concept as it was being developed.

Several internal company technical reviews of this limited conceptual-design level effort took place within this period, with quite helpful suggestions being made by SAIC technical-staff members. For example, the initial sizing of the vehicle was deemed to overstretch acceptance credibility among any outside target audience, were the approach to be taken beyond company bounds. In response, the vehicle’s gross takeoff weight was halved to its present size, without a substantial loss of the original suite of benefits offered. Still, the general feeling of those involved in these reviews, as well as local management staff, was that the Direct Earth/Moon cargo delivery concept probably departed too far from the more conventional thinking of that time to merit its being brought into the planning mainstream of NASA and its exploration study contractors, including SAIC. Consequently, several initially considered presentation opportunities, including several prospective conference technical-paper submissions, were forgone, and this planning activity was quietly concluded.

Some time later, the thought occurred to the authors that, still, the propulsion and structural system technological challenges illuminated by this approach, and the several innovative conceptual-design level technical responses so induced, particularly in propulsion, might well be of interest to the technical community. This led to the present paper being offered to, and with its acceptance, to be presented at the upcoming Joint Propulsion Conference.

II INTRODUCTION

A. Description of the Direct Earth/Moon Cargo Delivery Approach

As with human terrestrial exploration ventures that go back into historical time, future space exploration missions to be pursued by astronaut-class voyagers to the Moon and beyond, will require adequate and timely logistical support. This means that an effective Earth-originated cargo-carrying space transportation means must be provided, along with requisite human-rated carriers. A relevant space transportation guideline promulgated under the nation’s present human and robotic space exploration initiative being spearheaded by NASA is that, to the extent practical, the human and cargo transport means should be separately developed and operated, but in a fully coordinated manner. This paper addresses only the cargo-transportation component.

The conceptual design approach to be presented here mainly addresses later, rather than earlier phases of the long-term space exploration effort to be pursued. In the NASA-planning parlance, this would begin with Spiral 3, involving long-duration human stays on the Moon, and later phases of the exploratory campaign. This is where a variety of relatively large-mass cargo elements will need to be delivered to planetary destinations of interest. While the subject approach is considered broadly applicable to long-term efforts beyond Spiral 3, e.g., human Mars fly-by and landing missions, it is the cargo-logistical needs of multi-month human missions on the Moon that is mainly to be addressed in introducing the paper’s theme of Direct Earth/Moon Cargo Delivery.

This cargo-transportation approach contrasts sharply with the conventionally accepted multi-step mode of transferring a logistical payload from its Earth launch to its being safely landed on the surface of the Moon. Here, a
number of launch- and in-space vehicle types are involved, and several in-flight cargo transfers must be conducted among them. Specifically, following an Earth launch to low Earth orbit (LEO), the cargo is to be transferred to an in-space vehicle for the journey from LEO to a Low Lunar Orbit (LLO, or in some cases, to an intermediate Lagrangian point). Following a further payload transfer in LLO to a lunar descent vehicle, the cargo finally reaches its destination on the lunar surface.

Overall, this flight sequence requires a minimum of three different vehicle types and involves at least two in-space cargo transfers. These transfer operations may require astronaut support activities, or otherwise may be automated or use robotic services. It is clear that a fairly complex suite of flight equipment and cargo-handling means will be required to safely and effectively conduct this mode of logistical service to the Moon. Yet, since it may require only modest-sized vehicles, and may use existing-technology cargo-transfer means, this general operating scenario will likely be selected for at least the opening phases of the space exploration initiative.

In contrast to this conventional multiple-vehicle, and repetitive in-flight cargo transfer operations logistical-delivery approach, the subject direct-delivery scheme involves only one vehicle and requires no inflight cargo transfers. Speculatively, where relatively large cargo masses are required to be delivered to the surface of the Moon, say, in a matter of several days time, this direct-delivery transportation approach – to be described in conceptual-level detail – could eventually simplify, and even accelerate the on-going overall space exploration campaign, while reducing costs and augmenting mission capabilities. Looking ahead to the long-term future, it could feasibly be extended tologically support a projected follow-on “Lunar Exploitation Phase” of astronomical development. A specific embodiment of the direct Earth/Moon cargo delivery approach is described in the next sections, with emphasis on its propulsion-system make-up. This delivery vehicle concept is made up of a Boost Stage and a Space Stage system. Further, there are two distinct types of payloads to be landed on the Moon, as discussed next.

B. The Two Classes of Delivered Payloads

1. Discretionary Payloads – These are the conventionally-viewed payloads to be delivered to the lunar surface by the landed Space Stage. These cargo items may range over a wide variety of materiel types including propellants, other cryogens (e.g., nitrogen, helium), water, food, a wide range of personnel-support items, energy generation and utilization devices, communications systems, materials of construction, transportation devices, and so on. These would be generally logistical items that are required, and will have been appropriately requested for delivery to the Moon, to support developing advanced exploration and basing efforts.

2. Collateral Payload – This is a unique category of payload, one that is inherent in the direct-delivery mode of operation. The landed Space Stage, with its discretionary payload now unloaded, now becomes a source of additional logistical materiel. Being put to use directly as an intact personnel habitat or storage facility (e.g., use of the pressure-tight, voluminous oxygen tank as specially outfitted with floor members, etc.), or as partly or fully disassembled as a preplanned source of construction materials, special equipment (e.g., computer, communications equipment), electrical power systems, wiring, and the like, the Space Stage, as collateral payload, considerably augments the discretionary payload it delivers. However, a significant number of stage hardware items will not likely be useful on the Moon, e.g., thrust chambers, turbopumps, and certain NG&C components such as an inertial navigation unit. Hence the collateral payload is logistically “less efficient” than its counterpart discretionary payload.

3. A More Distant Alternative: Preserving One or More Space Stages for Relaunch for Payload Return to Earth – In any detailed consideration of the collateral payload role of the Space Stage, as just noted, it becomes clear that a significant fraction of the stage’s constituent hardware will likely have no particular lunar-utilization applicability. These previously, vital-to-flight items would include the stage’s basic rocket engine components such as the turbopump and thrust-chamber assemblies (the propellant delivery lines might be accepted as being useful in lunar service). There might also be certain high-value navigation, guidance and control items that would have only limited, or no particular service applicability on the Moon. This observation leads to the following idea.

With the potential development of in situ cryogenic propellants (e.g., from lunar ice deposits at the poles), the idea of preserving one or more Space Stages in a state of near flight readiness develops. The long-term opportunity arises to now return to Earth certain high-value, but non-usable (on the Moon) hardware for reuse in later-to-be-flown, relaunched Space Stages. Looking ahead to this possibility, it might be pointed out that these
dense hardware items would most probably be carried in the aft plug-nozzle compartment (to be described), and returned to the Earth’s surface through a conventional reentry/parachute recovery process. The comparatively bulky returning Space Stage would likely be then expended in a reentry burn up.

III DIRECT DELIVERY MISSION DESCRIPTION

A. Direct Flight Feasibility and General Flight Profile Approach to be Followed

1. Feasibility of Direct Flight to the Moon – It has been done before. Launch vehicles have departed the Earth and their payloads have been safely landed on the Moon. This has been accomplished without any intermediate inflight rendezvous and payload transfer operations. In other words it has been done in a direct-flight mode. The U.S. Surveyor project of the 1960s is a leading case-in-point, as is the contemporary Soviet Luna 16 and 24 spacecraft, which performed the more difficult sample-return (to Earth) mission.

While the Apollo program used the non-direct lunar-orbit rendezvous approach, so notably espoused by NASA Langley Center’s John Houbolt, the alternative approach of direct Earth/Moon flight was seriously considered in NASA’s early planning phase of what was to become the Apollo program. For instance, a highly-illustrated technical paper by NASA Headquarters’ Milton Rosen and Carl Schwenk entitled, “A Rocket for Manned Lunar Exploration” was predicated on direct flight. It described a Nova-class, multi-stage launch vehicle (its first stage was equipped with eight F-1 engines; Saturn 5 had five) that could land two astronauts on the Moon, and then return them to a safe Apollo-style safe splashdown back on Earth. So, from the standpoint of both unmanned spacecraft that have performed direct flights to the Moon, and in the studies of larger manned round-trip flights, the feasibility of direct Earth/Moon flights has been convincingly demonstrated.

2. General Direct Flight Approach to be Used – As will next be described, a large two-stage vehicle will be used in the exemplary embodiment of the direct Earth/Moon cargo delivery system approach to be covered in this paper. This massive vehicle (with about twice the takeoff weight of the Space Shuttle) consists of a solid-propellant rocket “boost stage” and a cryogenic liquid-propellant rocket powered “space stage.”

This large two-stage rocket-powered vehicle will be operated along the following mission profile from its Earth-launch to its landing on the Moon: The vehicle is vertically launched under Boost Stage thrust, pitching over into an easterly trajectory from an assumed Cape Canaveral launch site, and then accelerates through the atmosphere into space conditions. At about 6000 ft/s (1829 m/s) boost thrust is terminated, and the Space Stage separated and ignited for its first-burn acceleration to low-orbit insertion, followed by a brief circularization burn establishing the Stage in its parking orbit. Properly timed, the Space Stage performs its third burn to accelerate it to just over Earth-escape speed to place it into the proper trans-lunar flight condition. On arrival at the Moon a fourth burn places the Space Stage in a low lunar orbit aligned with an entry point for the final (fifth) descent burn to a soft landing on the surface, at the cargo-delivery destination site.

3. Mission Delta-Vs -- The characteristic flight-speed increment for this mission profile, as used in this study, are given below in terms of the three major segments of the overall profile:

Delta-V Requirements:

<table>
<thead>
<tr>
<th>Segment</th>
<th>m/s</th>
<th>ft/s</th>
</tr>
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<tbody>
<tr>
<td>Earth-to-orbit (to LEO)</td>
<td>7,632</td>
<td>25,040</td>
</tr>
<tr>
<td>LEO to Low Lunar Orbit (LLO)</td>
<td>3,915</td>
<td>12,845</td>
</tr>
<tr>
<td>LLO to Lunar Surface</td>
<td>1,895</td>
<td>6,217</td>
</tr>
<tr>
<td>Total: Earth to Lunar Surface</td>
<td>13,442</td>
<td>44,102</td>
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</tbody>
</table>

American Institute of Aeronautics and Astronautics
A. Design & Performance Characteristics

1. Leading Characteristics – The overall two-stage Earth/Moon Direct mission vehicle has a takeoff gross weight (TOGW) of 10,000 klbm (or ten million pounds), about double that of the Space Shuttle and about seventy-percent greater than the Saturn 5 Apollo launch vehicle. At 280 ft in length, it is some 70+ ft shorter than Saturn 5. The vehicle is sketched in Figure 1.

The Boost Stage would be made up of five modified Space Shuttle solid-propellant rocket units (SRBs), or equivalents. These are upgraded five-segment units with the recovery gear removed (no recovery/reuse is planned). As an observed option, the complex and heavy hydraulically operated gimbaled nozzle thrust vector control (TVC) system is removed and replaced with a conventional, lighter-weight fixed-nozzle installation. A separate smaller solid rocket, mounted for transverse-oriented rotation about its longitudinal axis, is added to provide TVC and other auxiliary functions. This approach was devised by the Allison Division back in the early 1960s and was referred to as “Alltrol.” Each of the five SRBs is equipped with an aerodynamically-stabilizing fin to which the Alltrol rocket unit is attached on its trailing edge. The general Boost Stage layout is reflected in the vehicle sketches of Figure 2.

The Boost Stage has the following overall characteristics:

- TOGW: 5 x 1,500 klbm = 7,500 klbm (seven and a half million pounds)
- Thrust: 5 x 3,000 klbf = 15,000 klbf (fifteen million pounds)
- Thrust/Weight at Launch: 10,000 klbm/15,000 klbf = 1.5 (half a g takeoff acceleration)

Its key performance and structural parameters, related as specific impulse, Isp, and propellant mass fraction, PMF, are: Isp = 275 sec, SRB PMF = 0.90, System PMF = 0.675 (includes the Boost Stage initial mass plus the Space Stage all-up mass, but as ratioed only to the solid-propellant mass).

The Space Stage would be an all-new large hydrogen/oxygen 40 ft (12.2 m) diameter stage. It basically consists of an aft-located oblate-spheroidal liquid oxygen tank, the main structural element of the stage, an extended-cylindrical forward-mounted

Figure 1: Overall layout sketch of the Earth/Moon direct cargo transporter system
liquid hydrogen container whose forward dome forms the nose of the vehicle (no forward-mounted payloads). It is powered by ten dual-thrust-chamber hydrogen/oxygen rocket engines whose diametrically opposite-mounted thrust-chambers form an annular arrangement of twenty units. A “skirted plug cluster” arrangement provides for further expansion of the individual thrust chamber plumes. Their exhaust streams are fed into the final ultra-high expansion ratio plug nozzle surrounded by an expandable conical outer nozzle (the outer skirt is initially in a cylindrical configuration at the stage diameter). Pitch/Yaw TVC is provided by differential throttling of the thrust chamber pairs. Roll control is provided by turbine-exhaust swivel-nozzle units. Alternatively, full three-axis attitude control can be provided by differential throttling, but this approach is not pursued here. The Space Stage design is shown in Figure 3.
The Space Stage has the following leading weight, thrust and performance-governing characteristics:

- Weight at staging = 2,500 klbm (two and a half million pounds)
- Thrust: 20 x 150 klbf = 3,000 klbf (three million pounds)
- Thrust/Weight = 3,000/2,500 = 1.2 (two-tenths of a g initial acceleration)
- Nominal values: Isp = 465 sec, PMF = 0.950 (propellant mass fraction)
- Backed-off values: Isp = 455 sec, PMF = 0.940 (inert weight margin of 11-percent)

2. Boost Stage Staging Speed Determination -- Based on the vehicle characteristics as given above, it is possible to estimate the staging speed achieved at Boost Stage propellant exhaustion. Subtracting this from the full ETO delta-V required yields the speed increment that must be provided by the Space Stage, up to the point of orbit insertion into the LEO parking orbit.

 Inputs are:

<p>| | | |</p>
<table>
<thead>
<tr>
<th></th>
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<tbody>
<tr>
<td>Boost Stage Gross Weight</td>
<td>7,500 klbm</td>
<td></td>
</tr>
<tr>
<td>Space Stage Gross Weight</td>
<td>2,500 klbm</td>
<td></td>
</tr>
<tr>
<td>System TOGW</td>
<td>10,000 klbm</td>
<td></td>
</tr>
<tr>
<td>Boost Stage Propellant Weight</td>
<td>6,750 klbm @ PMF = 0.90</td>
<td></td>
</tr>
<tr>
<td>Boost Stage Inert Weight</td>
<td>750 klbm @ PMF = 0.90</td>
<td></td>
</tr>
</tbody>
</table>

Which gives:

<p>| | |</p>
<table>
<thead>
<tr>
<th></th>
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</thead>
<tbody>
<tr>
<td>System Inert Weight</td>
<td>2,500 + 750 = 3,250 klbm</td>
</tr>
<tr>
<td>System Gross Weight</td>
<td>7,500 + 2,500 = 10,000 klbm</td>
</tr>
<tr>
<td>System PMF</td>
<td>6,750/10,000 = 0.675</td>
</tr>
<tr>
<td>System Mass Ratio</td>
<td>10,000/3,250 = 3.0769</td>
</tr>
</tbody>
</table>

(Calculation :)

For \( M_0/M_1 = 3.0769 \), \( \ln M_0/M_1 = 1.1239 \) – Using the ideal rocket equation:

\[
V_{\text{ideal}} = g \times \text{Isp} \times \ln M_0/M_1 = 32.2 \times 275 \times 1.1239 = 9,952 \text{ ft/s}
\]

Presuming all drag & gravity losses of ~4,000 ft/s are taken in the boost phase,

\[
V_{\text{actual}} = 9,952 - 4,000 = 5,952 \text{ ft/s} \text{ staging speed}
\]

Staging speed to LEO-insertion velocity increment: 25,040 – 5,952 = 19,088 ft/s to be made up by the Space Stage burns to LEO. Adding the other two delta-V increments gives the Space Stage total mission speed required: 19,088 + 12,845 + 6,217 = 38,150 ft/s (11,628 m/s).

3. Space Stage Discretionary & Collateral Payload Determination – Knowing the Space Stage total mission speed required, and having the two sets of specific impulse and propellant mass fractions (nominal and “backed off” values), it is possible to calculate the range of Stage discretionary payload capabilities. Delivered (discretionary) payload masses are the differences between the Space Stage propellants available (but not fully needed) as determined from the Isp/PMF values, and the propellants actually required to achieve the stated overall delta-V mission value. Four such discretionary payload values will be determined in view of the “4x4 matrix” comprising the, two-each, Isp and PMF values involved.

Space Stage Discretionary Payload Determination (based on the ideal rocket equation)

\[
V_{\text{ideal}} = g \times \text{Isp} \times \ln M_0/M_1, \quad \ln M_0/M_1 = V_{\text{ideal}}/g \times \text{Isp}
\]

Specific Impulse = 465 s

\[
\ln M_0/M_1 = 38,150/32.2 \times 465 = 2.5479
\]

\[
M_0/M_1 = 12.780
\]

With \( M_0 = 2,500 \text{ klbm}, M_1 = 195,618 \text{ lbm} \)

\[
M_{\text{prop-rqd}} = 2,500,000 - 195,618 = 2,304,382 \text{ lbm}
\]

Specific Impulse = 455 s
\[ \ln \frac{M_0}{M_1} = \frac{38,150}{32.2} \times 455 = 2.6039 \]
\[ \frac{M_0}{M_1} = 13.516 \]
With \( M_0 = 2,500 \text{ klbm} \), \( M_1 = 184,966 \text{ lbm} \)
\[ M_{\text{prop-reqd}} = 2,500,000 - 184,966 = 2,315,034 \text{ lbm} \]

As noted, delivered payloads are the differences between Space Stage “virtual” propellants available (but not fully needed) and propellants required to achieve the delta-V mission schedule.

Propellants available:
@ PMF = 0.950
\[ 0.950 \times 2,500,000 = 2,375,000 \text{ lbm} \]
@ PMF = 0.940
\[ 0.940 \times 2,500,000 = 2,350,000 \text{ lbm} \]

Resulting Discretionary Payloads Delivered to the Lunar Surface:

<table>
<thead>
<tr>
<th>Isp</th>
<th>PMF</th>
<th>Payload</th>
<th>Discretionary Payload</th>
</tr>
</thead>
<tbody>
<tr>
<td>465</td>
<td>0.950</td>
<td>[0.950, 465s]</td>
<td>[0.950, 455s]</td>
</tr>
<tr>
<td>465</td>
<td>0.950</td>
<td>2,375,000 lbm</td>
<td>2,315,034 lbm</td>
</tr>
<tr>
<td>465</td>
<td>0.940</td>
<td>2,304,382 lbm</td>
<td>2,315,034 lbm</td>
</tr>
<tr>
<td>455</td>
<td>0.950</td>
<td>2,304,382 lbm</td>
<td>2,315,034 lbm</td>
</tr>
<tr>
<td>455</td>
<td>0.940</td>
<td>2,304,382 lbm</td>
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<tr>
<td>455</td>
<td>0.940</td>
<td>2,304,382 lbm</td>
<td>2,315,034 lbm</td>
</tr>
</tbody>
</table>

Landed Space Stage (less cargo) as Collateral Payload

The inert mass of the Space Stage, with its discretionary payload removed, comprises the collateral payload (see discussion):

For the Isp = 465s, PMF = 0.950 case, 0.050 x 2,500 klbm = 125 klbm or 56.8 mT
\[ 125.0 - 70.6 = 54.4 \text{ klbm or 24.7 mT} \]
For the Isp = 455s, PMF = 0.950 case, 0.050 x 2,500 klbm = 125 klbm or 56.8 mT
\[ 125.0 - 60.0 = 65.0 \text{ klbm or 29.5 mT} \]
For the Isp = 465s, PMF = 0.940 case, 0.060 x 2,500 klbm = 150 klbm or 68.2 mT
\[ 150.0 - 45.6 = 104.4 \text{ klbm or 47.5 mT} \]
For the Isp = 455s, PMF = 0.940 case, 0.060 x 2,500 klbm = 150 klbm or 68.2 mT
\[ 150.0 - 35.0 = 115.0 \text{ klbm or 52.3 mT} \]

The collateral payload is seen to increase markedly as the stage Isp and PMF factors become less favorable – just the opposite of the trend with discretionary payloads. The extremes show a doubling of collateral payloads from about 25 to just over 50 mT.

Total Payload: Discretionary plus Collateral Payload

For the Isp = 465s cases:
PMF = 0.950 \[ 24.7 + 32.1 = 56.8 \text{ mT} \]
PMF = 0.940 \[ 29.5 + 20.7 = 50.2 \text{ mT} \]
For the Isp = 455s cases:
PMF = 0.950 \[ 47.5 + 27.2 = 74.7 \text{ mT} \]
PMF = 0.940 \[ 52.3 + 15.9 = 68.2 \text{ mT} \]

Ranging from the highest to the lowest performance characteristic sets (465s, 0.950 to 455s, 0.940), but this time in a different sequence, the total payload, that is, the discretionary plus the collateral payload, varies over a range of from about 50 to 75 mT.
### Discretionary Payload Fractions (Discretionary Payload Mass/Landed Space Stage Mass)

For the Isp = 465s cases:
- PMF = 0.950 32.1/56.8 = 0.565
- PMF = 0.940 20.7/50.2 = 0.412

For the Isp = 455s cases:
- PMF = 0.950 27.2/74.7 = 0.364
- PMF = 0.940 15.9/68.2 = 0.233

The general finding here is that the conventionally-stated dry-mass payload fractions range from about 23 to 57 percent. The lower range would appear to be acceptable from a general design feasibility standpoint. But the higher range might be considered problematical from a practical vehicle design standpoint.

### B. Direct-Flight Vehicle as a Super-Heavy-Lift ETO Payload Transporter

The baseline vehicle concept, as described earlier, is sized for 16-32 mT direct Earth/Moon discretionary payload delivery. Reoriented to alternative missions (no doubt requiring redesign measures), this system could provide an impressive payload-lift capability when operated in Earth-to-orbit (ETO) service, with its significantly reduced delta-V requirement. Assuming a conservatively-designed vehicle with less optimistic Space Stage values (PMF = 0.940, Isp = 455 s), for a nominal ETO mission flight-velocity increment to LEO of 25,000 ft/s (7620 m/s), the discretionary payload estimate is 240 mT (529 klbm). This is of the order of the on-orbit mass that is sometimes called out for the several Mars missions under consideration in the exploration initiative. But, as a precautionary note, this payload mass is some 78-percent of the total Space Stage mass entering LEO, i.e., with its mission propellants expended, so this indicated ETO delivery capability may be highly problematical. (No studies conducted.)

### V SPACE STAGE PROPULSION SYSTEM

As covered earlier, the Earth/Moon direct cargo delivery vehicle considered here is made up of two stages, referred to as the Boost Stage (solid-propellant rocket elements) and the Space Stage (hydrogen/oxygen rocket propulsion). Propulsion-wise, the Boost Stage is essentially a conventional design, needing no further discussion. This section provides an initial technical description of one, less-than-conventional version of a candidate propulsion system for the Space Stage, with brief mention of how it might be developed. Figure 4 presents the selected propulsion system on a simplified schematic basis.

#### Design Characteristics (ratings, sources)
- Propellants: Liquid hydrogen/liquid oxygen
- Thrust: 3 million lbf (vacuum)
- Specific impulse (vacuum): 465 s (target), 455 s (reduced)
- Starts/Restarts: Unlimited – Five nominally required for the Earth/Moon flight profile
- Thrust vector magnitude control [engine throttling range: 100 – 20 percent (5:1)]
- Thrust vector moment control (TVC): Pitch/Yaw planes – via differential thrust
- Roll plane – pivoting turbine exhaust nozzles
- Sources of major engine hardware – Existing/past rocket engine developments:
  - Turbopump assemblies (TPAs): Vulcain 2 (SNECMA/SEP) or J-2S (Rocketdyne)
  - Thrust chamber assemblies (TCAs): J-2S (Rocketdyne)
  - Precision propellant throttling valves (EMA-operated): XRS2200 (Rocketdyne, linear aerospike engine)

#### Design Features (nominally selected)
- “Engine Element” consists of two cross-diameter TCAs fed by a single TPA
- The two TCAs are located opposite one another at the full stage body diameter
- Multiple body-fixed (non-gimbaled) thrust chamber assemblies (TCAs), 20
- Regeneratively-cooled, 150 klbf nominal-operating (200 klbf rated), 1500 psia Pc
- Centrally located turbopump assemblies (TPAs), 10
- Gas generator cycle, with exhaust afterburning, rated for 300 klbf, >1500 psia propellant delivery service
- Two step exhaust expansion to a very high nozzle area ratio (~500:1):
  1) TCA nozzle expansion (~50:1), 2) Common expandable nozzle extension (x10)
Figure 4: Space Stage propulsion system overall multiple-engine layout plus a simplified schematic diagram of one engine (of ten)
Thrust of each 1-TPA/2-TCA engine set: 300 klbf (nominal)
Number of engine sets: 10 giving a total thrust of 3 million lbf (10 x 300 klbf)

Operational Features

Basic multi-engine approach: Each single-TPA/twin-TCA engine element is independently operated. Precision propellant flow-control valves are fitted to each TCA and TPA. In effect, the propulsion system is comprised of ten such engines, with each non-interactive engine being separately controlled in terms of start/restart, thrust-level setting, O/F setting, differential thrust output for pitch/yaw control, and turbine exhaust pivoting for roll control.

Thrust vector magnitude control (overall throttling level): Overall system thrust level is set by the combination of engine sets being operated, 10 down to 2, and the individual engine-set throttle setting, 100 to 20 percent of rated thrust. This gives an overall system thrust-level range of from 3,000 klbf (10 x 300 x 1.0) to 120 klbf (2 x 300 x 0.2) for an overall throttling ratio of 25:1. Minimum thrust would be just 4-percent of maximum thrust. These extreme settings would correspond, respectively, to the initial start-up in sub-orbital (to LEO) flight of the fully-loaded stage at 2,500 klbm (stage initial thrust/weight ratio of 1.2), to be eventually followed by the lunar-surface descent and landing of the nearly empty stage (thrust/weight ratio of near unity at a near-propellant-expended landing mass of ~125 klbm).

Thrust vector moment control (traditional TVC): The stage disturbing moments to be controlled should be low level (absence of atmospheric-flight disturbances) and caused mainly by center of gravity (c.g.) shifts due to propellant motion, and TCA misalignments, both to be minimized (e.g., with propellant baffles, TCA precision alignment means).

Pitch/Yaw Control -- The pitch/yaw disturbances are to be countered by differential thrust operation of each engine set’s pair of TCAs. For maximum moment-generation at a constant-thrust output, the two TCA thrust settings would shift from a balanced 150/150 klbf, to a corrective-moment generating 175/125 klbf, with 175 being the maximum rated TCA condition (with proof testing at 200 klbf for additional margin). For a 40-ft diameter stage (nominal sizing) this would yield a maximum correcting moment of about one-million lb-ft from just one engine set. Assuming this engine set is located within the plane of the disturbance, the remaining 9 sets would contribute sizable, but somewhat lower correcting moments (shorter operating arms). Actually this would be more like 7 active engines, since two sets would then essentially fall along the “neutral axis” of the disturbing moment being cancelled, and thereby made largely ineffective in terms of moment-generation. The pitch/yaw differential-thrust approach would appear completely adequate for controlling the stage, but detailed analyses should be pursued, were design work to be entered into.

Roll Control – Roll disturbances would mainly derive from TCA misalignments, and should likely be orders of magnitude less than the pitch/yaw disturbing moments. The use of pivoting turbine-exhaust nozzles, as used in the RS-68 engine in powering the Delta 4 EELV is anticipated to be an adequate roll-control means. There is an alternative approach using the “third axis” capability of the differential-throttling approach that would provide an adequate range of control moments about the stage roll axis, and would allow the swivel-nozzle complication to be deleted (but this option is not to be gone into here).

Engine-Out Provisions – In the case of an engine failure, or an unsatisfactory operating condition being experienced (or predicted by its EHM system), the affected engine set can, by design, be individually safely shut down, or reduced to a lower operating condition. The remaining nine (9) engines can then continue operation, providing a minimum of 90-percent of the pre-shutdown thrust level. It should be recalled that in-space acceleration operations have a “fairly flat” sensitivity to the vehicle thrust/weight ratios being operated.

In an engine-out condition during later phases of flight, when the overall thrust level has been reduced to limit stage g-loading, with one (or more) engine sets shut down, the thrust level of the remaining operating engines can be correspondingly increased to restore the planned stage acceleration profile, if this is desired. As the stage mission profile is executed through several “burns,” selected engine sets will be shut down (to reduce acceleration-loading on the reduced-mass vehicle). As such, these units can serve in reserve, to be restarted to compensate for any subsequent engine-out shutdowns to be experienced. Generally speaking, this overall independent ten-engine arrangement appears fairly immune to loss-of-engine caused problems.

American Institute of Aeronautics and Astronautics
A. Configuring the Space Stage Propulsion System

The Space Stage propulsion system has been described in general terms as being made up of ten independent, but operationally coordinated engine sets, as shown in Figure 4. Each hydrogen/oxygen engine would operate at a nominal 300 klbf thrust level, with a throttling capability down to 60 klbf (a 5:1 ratio, or throttling to 20-percent of full thrust). Each engine consists of a single turbopump assembly complete with its drive gas-generator and controls, plus two body-fixed thrust chamber assemblies equipped with precision-operated propellant valves. Each of these thrust chambers is placed at diametrically opposite positions at the outer rim of the 40-ft diameter liquid oxygen tank. This signifies that, with the turbopump assemblies located centrally at the bottom of the tank, that there will be some 20 extended high-pressure hydrogen and oxygen feed lines, each having a length of the order of 20 feet.

Figure 4 presents a simplified layout for one engine, and for the full propulsion system, showing these extended feed lines, as well as the basic layout of the turbopump assembly, as well as its propellant supply means. The liquid oxygen supply is directly available to the liquid oxygen pump from the tank sump at the bottom of the lower tank dome. But the liquid hydrogen must be fed, over some length of passage, from the tank’s annular sump area, formed higher up at the intersection of the cylindrical liquid hydrogen tank and the equatorial band around the oxygen tank. See Figure 1 and 3 for this overall layout. The low-pressure feed lines leading to the hydrogen pumps will then pass along the bottom of the oxygen tank in radial fashion leading to the centrally located turbopumps, two opposite-running lines feeding each one of the ten liquid hydrogen pumps. The high-pressure hydrogen feed lines must then be routed back along this same path to the individual thrust chambers.

A concern with such long low- and high-pressure cryogenic propellant feed lines is their thermal design. Questions arise: How are these lines to be properly chilled-down to avoid vapor pockets and ensure a well-controlled engine start? What are the thermal insulation requirements to eliminate any such vapor pockets and to minimize propellant boil-off losses? And, how can the added inert weight burden of these extended lines be minimized?

In response to these propellant supply related issues, the following design approaches are being suggested:

1. The 20 high-pressure liquid oxygen lines leading from the turbopump outlets to the thrust chambers will serve as load-carrying, integral radial rib elements as installed in the lower dome of the liquid oxygen tank. They will be externally insulated, as will the entire dome. This will ensure that, whenever the tank is loaded with the cryogenic oxidizer, these lines will be effectively chilled down and in readiness for an engine start and run condition. The inert mass of these delivery lines, having been “put to work” as structural members, will now be accounted for as weight constituents of the tank, rather than as engine items. Also, any possible line leakage will result in the escaping high-pressure liquid oxygen being safely returned to the tank.

2. The 20 well-insulated low-pressure hydrogen lines delivering cryogenic fuel from the annular sump area of the hydrogen tank to the turbopumps will be routed just below the liquid oxygen dome-rib piping. This will permit these hydrogen lines to be physically supported by short “hanger elements” from support fittings attached to these structurally-secured liquid oxygen lines.

3. The remaining set of propellant lines are the 20 high-pressure hydrogen lines leading from the turbopump outlets to the thrust chambers. These are to be routed directly through the low-pressure hydrogen lines just described. With liquid hydrogen present in these low-pressure lines leading to the pumps, the high-pressure lines within them will be inherently chilled down for a proper engine start and run. In the possible event of leakage from these lines, any escaping hydrogen would be safely captured by the surrounding low-pressure lines.

It is pertinent to observe that, regarding the avoidance of vapor formation in these liquid hydrogen lines, at pump-out pressures of 200 psia and higher, the hydrogen is at supercritical conditions where there will be an absence of two-phase liquid/vapor conditions. Only a single-phase “cryogenic fluid” will then exist.

These propellant feed-lines features are represented in the simplified sketches of Figure 5.
B. Space Stage Propulsion System Development -- General Approach

1. Principal Focus of the Development Process -- The main propulsion system development focus will be on the single-turbopump/dual-thrust-chamber “engine set” element, ten of which make up the main hardware items of the overall system. A secondary focus will be on the physically large expandable common exit nozzle, and its associated central plug-nozzle elements. Together, these items comprise the very high exit nozzle area-ratio (at ~500:1) of the final skirted-plug assembly. This contributes to the high overall system specific impulse levels to be achieved. The initial exhaust expansion is conducted within the twenty thrust chambers, each having a 50:1 nozzle “feeding” this common final nozzle assembly.

2. Use of Existing Subsystems Provides Development Cost Benefits -- Important to note, this development process can start with two flight-proven major hardware items, namely the Vulcain 2 (or J-2S) TPAs and TCAs. The expandable common nozzle has a significant technology development and demonstration background as well (Aerojet conducted experimental proof-of-concept work on this concept for the Air Force back in the 1960s). The presumed availability of developmental and operational experience with these existing-hardware-status subsystems should lead to a sharp reduction in development cost and time, as well as significantly reduced technical and programmatic risks and uncertainties. The general development theme being pursued here can be summarized as follows: It is to use conventional (and available) hardware component designs, but installed in an unconventional installation arrangement, one offering a number of design and operating benefits.

3. Development Steps to be Considered

1. Overall propulsion system design and analysis to be performed when Space Stage quantitative requirements and design features are made available.

2. Upon confirmation of the (above related) design approach, a technical information compilation of all propulsion system constituent major items is to be documented (e.g., TPA, TCA, common nozzle, precision-control valving, propellant lines).

3. Design-optimization needs will be established for modifying the existing TPA and TCA hardware item designs, and for designing the common skirted-plug nozzle extension.

Figure 5: Space Stage cryogenic propellant feed line features
4. Initiate engineering modifications of existing hardware items in preparation for integration into overall propulsion system; produce prototypical hardware ready for hot-firing tests.

5. Assemble an engineering prototype engine set (TPA + 2 TCAs) and conduct exploratory static-firing tests over the full operating range (rated thrust, O/F, throttling-range).

6. Design, construct and ground test (at sea-level & vacuum conditions) a complete subscale propulsion system to demonstrate system performance and operability (e.g., specific impulse performance, thrust vector control, engine-out).

7. Develop and demonstrate fabrication means for the expandable common nozzle extension. Critically review the 1960s AF/Aerojet work cited. Fabricate and test subscale versions. Develop tooling for full-scale units.

8. Assemble several pre-production engine sets and carry out a full qualification test program on these engine sets. Integrate acceptance-level hardware in consonance with ongoing Space Stage development steps.

9. As coordinated with the overall Space Stage production program, initiate full production of the propulsion system and deliver production hardware into the vehicle integration process.

10. Provide full propulsion system engineering support for initial ground and flight testing of Space Stage; follow up with production hardware logistical support and flight-support operations.

C. Critical Assessment of the Stated Space Stage Propulsion System Specific Impulse Levels

The Space Stage’s hydrogen/oxygen propulsion system is made up of an unconventional body-fixed (non-gimbaled) multiple thrust chamber assembly fed high-pressure propellants from centrally located turbopump assemblies (as shown in Figure 4). The turbines are driven by closely-integrated auxiliary gas-generators, using fuel-rich hydrogen/oxygen combustion products. Post-turbine flows are to be afterburned (reheated) to increase system performance as discussed below. Following partial expansion of the high-temperature exhausts in the 50:1 thrust-chamber nozzles, these gases are further expanded (accelerated) in a large common “skirted plug” nozzle extension, to a very high nozzle area ratio (of ~500:1).

Its target specific impulse level of 465s was set as equivalent to the highest value attained by hydrogen/oxygen engine types in flight service today. This marker is 466.5s attained by the Pratt & Whitney RL10-B2 engine used for upper stage propulsion on the Boeing Delta IV EELV. This vehicle has successfully flown a number of times. The RL10 engine is a 40-year veteran development and has many different type variants. This engine operates on a unique expander-cycle system, in which the turbopump turbine is driven by heated, high-pressure hydrogen, first passing through the thrust-chamber regenerative cooling jacket. It has a space-extendable, carbon/carbon nozzle with a 284:1 expansion area ratio.5

A leading limitation of conventional gas-generator cycle rocket engines is that, being open-cycle systems, the separately-exhausted turbine exhaust, while but a small fraction of the total engine exhaust flow, does not contribute thrust effectively, and thus somewhat penalizes engine specific impulse levels. This penalty is not experienced in the expander cycle, or in staged-combustion engines like the SSME. These are both closed-cycle systems and have no separately-exhausted turbine exhausts. The SSME-class engines (there are Russian and Japanese counterparts) provide vacuum specific impulse levels of 450+ s, as did earlier RL-10 expander-cycle variants (those without the large nozzle of the B2 engine).

As noted, the open-cycle gas generator engines typically provide somewhat lower specific impulse levels than the closed-cycle ones. But, for a number of developmental and operational advantages, earlier rocket engines in the U.S. were almost exclusively gas-generator systems. The hydrogen/oxygen J-2 engine, a gas generator engine rated at 200 klbf-thrust and 420s Isp was a mainstay powerplant used in the Apollo program’s Saturn-series of launch vehicles. A contemporary cryogenic gas-generator engine, the European Vulcain 2 300 klbf-thrust engine is used in the Ariane 5 launch vehicle, being rated at 434s (vacuum) with a 50:1 exit nozzle.

For a number of technical reasons, not to be gone into here in any depth, the subject Space Stage design suggested here uses the gas-generator cycle for its ten single-turbopump/twin-thrust-chamber engines, with significant physical
separation between the turbopumps and the thrust chambers. This geometric layout scheme is judged to be basically incompatible with closed-cycle engine arrangements. Two separate design approaches have been incorporated in this unconventional system specifically to compensate for the noted specific impulse “drawback” of the selected open-loop cycle: 1) the very high area-ratio (final) nozzle, discussed earlier, and 2) afterburning, or reheating the local low-temperature, fuel-rich turbine exhaust by oxygen addition.

The first, the ultra-high area-ratio nozzle, is an established approach for elevating specific impulse performance, e.g., as in the RL10-B2 engine mentioned above. The second, turbine exhaust afterburning, remains undemonstrated, to our knowledge. It is surmised that previous engine designs had restrictions precluding the selection of this performance-enhancement approach. A possible third systems-level approach, considering the stage’s unique light-weight liquid hydrogen tank, a design that could be extended in volume with little weight penalty, is – by increasing the hydrogen loaded -- to reduce the engine operating O/F mixture ratio from the conventional levels of 6:1+, to an O/F of ~5, and possibly lower. As recalled, in vacuum, high nozzle area-ratio conditions favor such a move toward peaking specific impulse.

Going to the bottom line purpose of this discussion of performance enhancement measures, the stated target specific impulse of 465s for the Space Stage may well be optimistic, regardless of the “special steps” to raise performance just mentioned. For that reason, it is prudent to examine the consequences of reduced specific impulse values, the one included in this paper, being 10s lower at 455s, a performance reduction of about 2.2-percent. This was referred to as a “backed off” value. The payload impacts of this reduction in performance have been reflected earlier.

VI SUMMARY

A distinctly “less conventional” space exploration logistical cargo transportation system design approach has been suggested in this paper. It is based on a single-vehicle, direct-flight basis, without the need for multi-vehicle cargo-transfer operations enroute, the conventional scheme judged likely to be pursued in initial space exploration phases. The specific example presented is referred to as direct Earth/Moon cargo delivery. This class of mission support might well be appropriate for later, rather than earlier phases of lunar exploration. Notably, this direct-delivery mode is in sharp contrast to the three-step cargo-carrying process usually considered: Earth-to LEO, LEO to LLO, and LLO to the lunar surface. Three different vehicle types and two in-flight cargo transfers from one vehicle type to another would be required for this more conventional delivery mode.

The direct-delivery approach calls for only one vehicle type and no cargo transfers enroute to a lunar-surface delivery. However, as envisioned here, this one vehicle type would be both quite large (about double the launch mass of the Space Shuttle) and would require very high propulsion specific impulse performance levels (solid-rocket: 275s, cryogenic liquid rocket: 455-465s), as well as extremely favorable stage propellant mass fractions (solid: 0.90, liquid: 0.94-0.95). These values admittedly challenge today’s state of the art in space-rocket propulsion and space-system structural design.

The exemplary Earth/Moon direct-delivery transport vehicle concept that is brought forward as having these ambitious qualities is a two-stage vehicle made up of a solid-propellant rocket “Boost Stage” plus a cryogenic liquid-rocket powered “Space Stage.” While the former is taken to be a five-unit array of modified 5-segment Shuttle SRBs (or equivalent), the latter is a large (40-ft, 12.2-m diameter) new-design cryogenic stage with a number of rather novel propulsion and structural features. The Boost Stage is expended at 6000 ft/s with the Space Stage separating under power, then accelerating towards its orbital insertion point. Operating always in the vacuum of space, it continues in executing its multi-burn mission profile to LEO, then to LLO, and on down to the surface of the Moon, achieving a cumulative delta-V of 38,150 ft/s (11,628 m/s), along the way.

A discretionary payload of 32 to 16 mT is landed (based on nominal and reduced stage performance-governing parameters as noted above). But, in addition, the landed Space Stage itself is considered “collateral payload” at 25 and 52 mT, respectively. This title follows because a major fraction of the stage’s mass, by “foresight of design,” can now be used as a source of diversified materiel on the Moon, or more directly, e.g., as a habitat. Admittedly, a certain significant mass fraction of the Space Stage’s componenry would obviously not be of direct use on the Moon. This would include basic propulsion and GN&C elements, e.g., thrust chambers, turbopumps, inertial guidance units, landing radars. The longer-range plan here is, with the prospective availability of in situ produced cryogenic propellants, to relaunch a Space Stage, one held in flight-readiness status over the years, for a
return flight to Earth carrying such high-value hardware back for eventual reuse on subsequent stages, those to be possibly flown in the program’s out years.

An interesting lunar-delivered payload mass trend has been observed as the two main Space Stage “performance parameters” were varied over their chosen range: Isp = 465/455s and PMF = 0.950/0.940. While the discretionary payloads decreased over the full range of decreasing performance “goodness” (from 32 to 16 mT), as would be expected, the collateral payload mass, that of the landed Space Stage less its delivered discretionary payload, actually increased with these performance-factor reductions (from 25 to just over 50 mT), a doubling of collateral payloads.

*Total* payload masses, the sum of the discretionary and collateral payloads (admittedly, somewhat of an “apples and oranges” agglomeration) are found to grow about 50-percent in going from the 465s/0.940 case to the 455s/0.950 case, the resulting values being about 50 and 75 mT.

Finally, it was of interest to estimate the direct-delivery vehicle systems’ potential for conducting the ubiquitous ETO mission in delivering large cargo loads to LEO (as would likely be required for a human Mars mission). As a rough estimate, unsupported by any design work or in-depth systems analysis, the ETO discretionary payload was found to be increased over an order of magnitude above its lunar-surface delivery values. This estimate, with backed-off stage performance parameters, was a figure of 240 mT (529 klbm). Because this is as much as 78-percent of the loaded stage’s arrival mass on orbit, this value should be viewed as problematic, clearly needing further analysis.

### VII CONCLUDING REMARKS

A “different” engineering approach for transporting relatively massive cargo to space exploration destinations has been illustrated in this paper. This approach is one potentially of interest for later-phase lunar missions involving very long-duration crew exploration and operating-base development. The direct Earth/Moon cargo delivery mission described is representative of this departure from the more conventional multi-vehicle, multi-step process, one requiring several in-flight cargo transfer operations. The subject direct-delivery approach calls for only one vehicle type and has no need for cargo-transfers. These system simplification benefits come, however, at the cost of requiring a new, very large vehicle, of a design that would significantly challenge the state-of-the-art in propulsion and structural design. The essence of such a system concept has been illustrated and discussed.

Innovative design approaches have been incorporated into this direct-flight vehicle concept, and particularly into its “Space Stage” element, in the direction of achieving the requisite high specific impulse performance and propellant mass fractions. This includes the stage’s unconventional engine-out-tolerant, multiple hydrogen/oxygen rocket engine configuration. It features differential-throttling TVC, a deep throttling operation as required for lunar descent and landing, and an ultra-high area-ratio nozzle extension. Target propellant mass fraction attainments call for the unique tankage and propellant feed-line designs, and innovative payload stowage arrangements, as expressed in the stage design. The aft liquid oxygen tank is the stage’s main structural item with its direct fixed-chamber thrust-input arrangement, and its common bulkhead shared with the forward hydrogen tank. Dense payloads are carried aft, below this tank, mainly in tension, while non-dense, thermally robust cargo is contained directly within the specially extended, forward-mounted lightweight liquid hydrogen tank. This sharp departure from conventional design practice, if demonstrated to be feasible, avoids the need for the usual heavy and complex, forward-carried shrouded-payload volume.

As mainly projected for the prospective longer-term and likely heavier-cargo based logistical needs of an ongoing national (and international) human space exploration effort, the unconventional space transportation ideas expressed here may have some future relevance. If so, associated pre-development technology development and demonstration project activities may be placed on the agenda of those who will be active in tomorrow’s space transportation and propulsion systems R&D communities.
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