The Advanced Cryogenic Evolved Stage (ACES)- A Low-Cost, Low-Risk Approach to Space Exploration Launch

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Space exploration top-level objectives have been defined with the United States first returning to the moon as a precursor to missions to Mars and beyond. System architecture studies are being conducted to develop the overall approach and define requirements for the various system elements, both Earth-to-orbit and in-space. One way of minimizing cost and risk is through the use of proven systems and/or multiple-use elements. Use of a Delta IV second stage derivative as a long duration in-space transportation stage offers cost, reliability, and performance advantages over earth-storable propellants and/or all new stages. The Delta IV second stage mission currently is measured in hours, and the various vehicle and propellant systems have been designed for these durations. In order for the ACES to have sufficient life to be useful as an Earth Departure Stage (EDS), many systems must be modified for long duration missions. One of the highest risk subsystems is the propellant storage Thermal Control System (TCS). The ACES effort concentrated on a lower risk passive TCS, the RL10 engine, and the other subsystems. An active TCS incorporating a cryocoolers was also studied. In addition, a number of computational models were developed to aid in the subsystem studies. The high performance TCS developed under ACES was simulated within the Delta IV thermal model and long-duration mission stage performance assessed. Pratt & Whitney Rocketdyne studied the effects of long-duration missions on the RL10 engine, and found that, with few exceptions which could be dealt with by design, the RL10 was suitable for the long EDS missions. The high performance TCS, when used in orbit, requires a thermodynamic vent system (TVS) and vapor cooled shields (VCS). The MLI/TVS/VCS architecture was optimized using the Boeing Design Sheet^R tool for two configurations. The first was "independent", where both the LH_2 and the LO_2 tanks vented, and the vent gas used in a fuel cell to produce onboard power. The second was "integrated" where only the LH_2 tank vented and the H_2 vent gas used in the LO_2 tank VCS to keep the LO_2 tank vent free. All other affected EDS subsystems were studied for operation over long-duration missions. Most of the subsystems are technology extensions of currently operational and flying hardware with relatively low risk. The higher risk subsystems include the TCS, the engine packaging and performance, the autonomous rendezvous and capture mechanisms, the low heat leak skirt structure and micrometeorite-orbital debris protection. As a result of this study, a design was developed for an EDS suitable for long duration space exploration missions

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Nomenclature

ACES	Advanced Cryogenic Evolved Stage
AR&C	Autonomous Rendezvous and Capture
C&DH	Command and Data Handling
DV	Delta V – velocity increment
EDS	Earth Departure Stage
GN&C	Guidance, Navigation, and Control
H_2, GH_2, LH_2	Hydrogen, Gaseous Hydrogen, Liquid Hydrogen
LAD	Liquid Acquisition Device
LEO	Low Earth Orbit
LOI	Lunar Orbit Injection
MLI	Multilayer Insulation
M/OD	Micrometeorite/ Orbital Debris
O_2, GO_2, LO_2	Oxygen, Gaseous Oxygen, Liquid Oxygen
PAF	Payload Attach Fitting
pph	Pounds per Hour
Q	Heat Flow
RIFCA	Redundant Inertial Flight Control Assembly
SUS	Super Upper Stage
t	Metric Ton
TCS	Thermal Control System
TLI	Translunar Injection
TRL	Technical Readiness Level
TVS	Thermodynamic Vent System
VCS	Vapor Cooled Shield

I. Introduction

SPACE exploration top-level objectives have been defined with the United States first returning to the moon as a precursor to missions to Mars and beyond. System architecture studies are being conducted to develop the overall approach and define requirements for the various system elements, both Earth-to-orbit and in-space. Lunar missions are estimated to require transportation of up to 150 t to lunar orbits. In order to reduce total system cost and risk, reliance on derivative systems will be preferred to all-new systems. Use of a Delta IV second stage derivative as an in-space transportation stage offers cost, reliability, and performance advantages over Earth-storable propellants and/or all new stages.

Cost advantages accrue because the intellectual capital has already been invested: analysis models are complete, the stage design is likely about 80% complete, and existing tooling and processes are in place. The availability of flight-proven hardware and in-hand hardware sources, together with learning curve benefits due to known cost and schedule will also result in low risk (Figure 1-1).

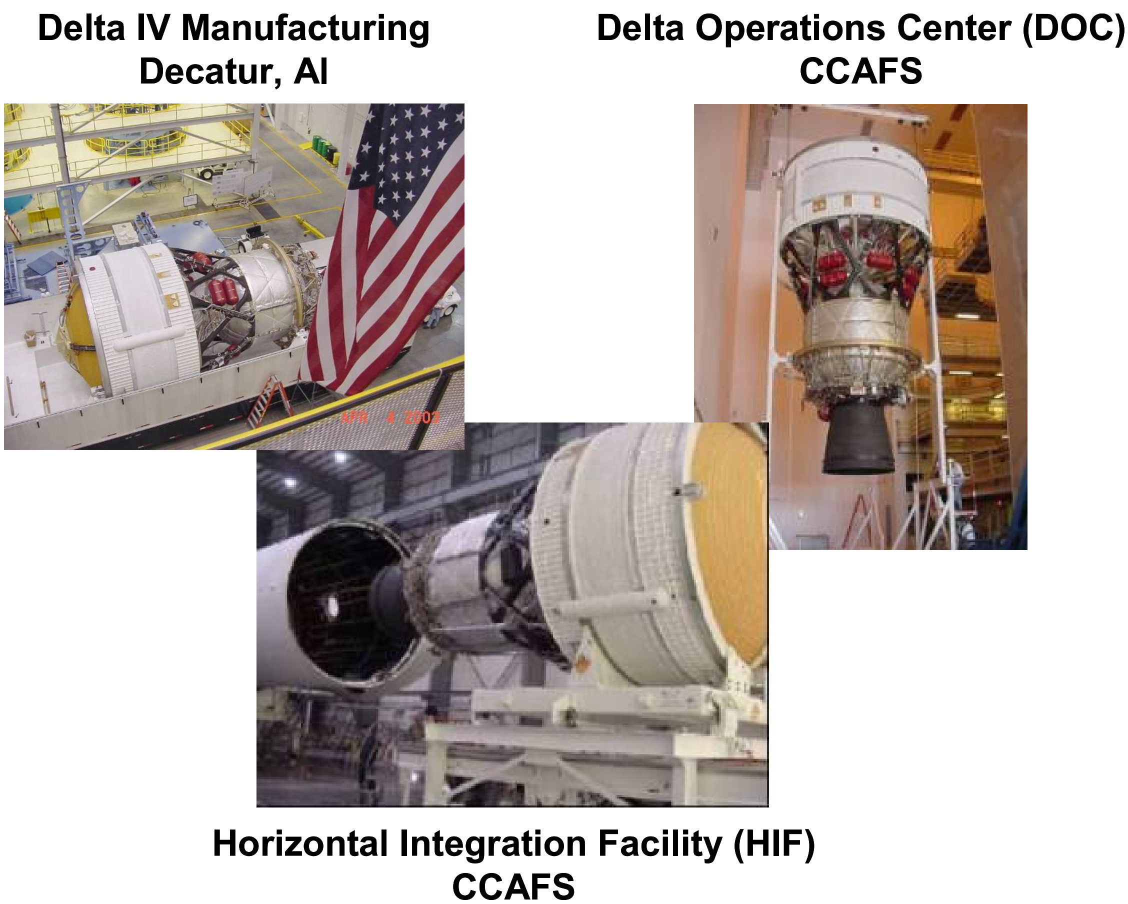

Figure 1-1. Invested Intellectual Capital Results in Low Costs and Risks

A modified Delta IV second stage has the ability to satisfy NASA's requirements for many of the Design Reference Missions shown in Table 1-1. However, the longer duration required for the TLI mission, for example, requiring weeks or months vs. hours for the current stage, requires more advanced techniques to minimize cryogen boiloff, additional power and Attitude Control System capability, and probable modification to other subsystems.

Table 1-1. Design Reference Missions

DRM #	Mission
1	Trans-Lunar Injection (TLI) Only
2	TLI + Lunar Orbit Insertion (LOI)
3	TLI + LOI + Return
4	Interplanetary $C_3 > 0$
5	L1 Insertion

A number of options are available for providing Earth Departure Stage (EDS) functionality. Figure 1-2 shows EDS launch vehicle configuration options. The first option is an Apollo/Saturn approach where the entire lunar payload is carried to orbit on top of the upper stage. After a handful of orbits (1.5 or 2.5 for Apollo) the upper stage performs the TLI burn. The life of the upper stage is short (hours) so extended duration kits are not required. However, for Evolved Expendable Launch Vehicles (existing or derivatives) the payload capability to a TLI trajectory is small. The second option is to launch the EDS as a payload to LEO. Delta IV Heavy derivatives can place as much as 55 t in LEO. In this case a fully fueled EDS is placed in LEO where it waits for the payload to be launched. The payload and EDS then rendezvous and dock with each other, followed by the TLI burn. This scenario requires additional functionality (e.g. AR&C, long-term cryogen storage) but significantly reduces the launch vehicle development cost. The final option considered is one that provides the functionality of the second and EDS stages into one Super Upper Stage (SUS). The SUS places itself into orbit, thus saving the cost associated with 2 separate stages. Trade studies have shown the SUS to be an attractive option.

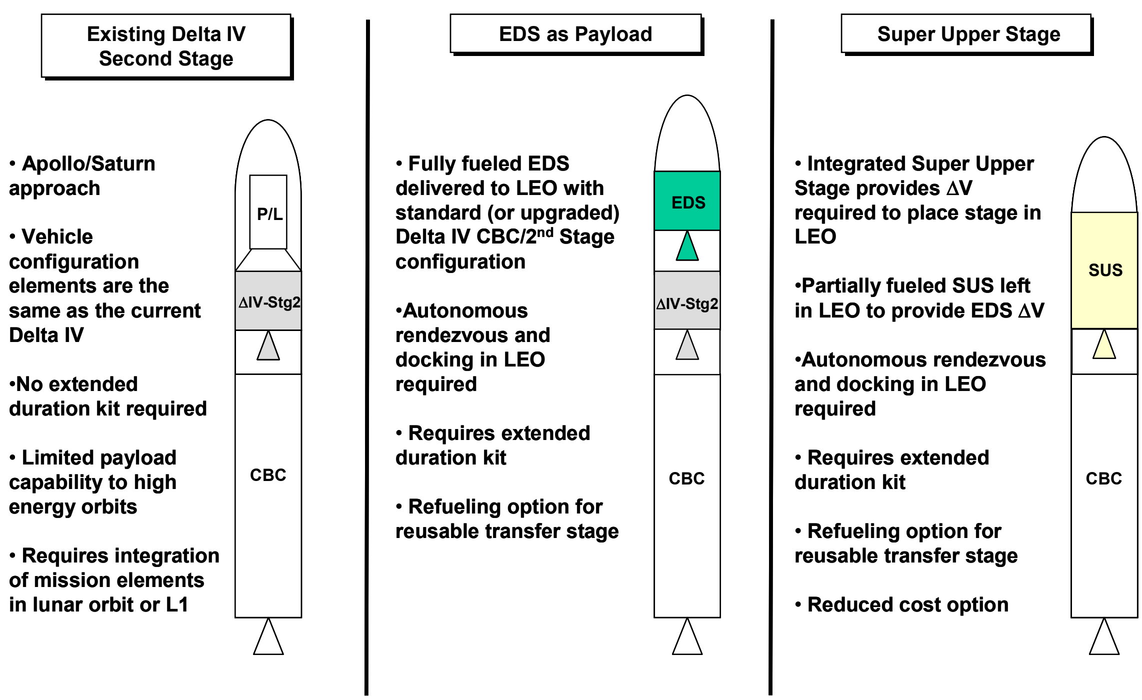

Figure 1-2. EDS Configuration Options

In addition to the thermal control system (TCS) modifications needed to reduce propellant venting and increase stage performance, many other subsystems will be impacted when adapting the Delta IV second stage to be an EDS to perform TLI/LOI missions. Figure 1-3 shows the scope of the vehicle subsystem impacts.

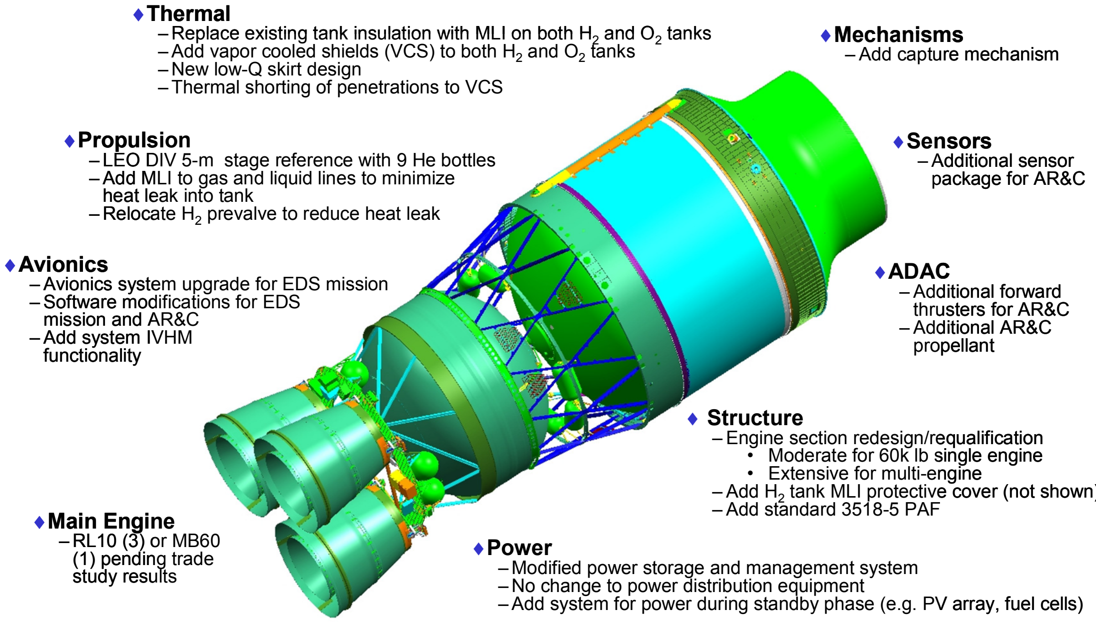

Figure 1-3. EDS Subsystem Impacts

The Thermal Control System (TCS) has high risk, and was selected for detailed analysis under the ACES contract. The ACES effort concentrated on a lower risk passive TCS, the RL10 engine, and the other subsystems shown above. An active TCS incorporating a cryocoolers was also studied.

II. Passive Thermal Control System

The primary modification to the TCS is the tank insulation system. Reducing H_2 and O_2 propellant boiloff to an acceptable level for a multiple-month mission duration requires over 2 orders-of-magnitude improvement in thermal performance. Multi-Layer Insulation (MLI) is the only available insulation that can provide this type of improvement. Existing H_2 and O_2 tank insulation will be replaced by MLI integrated with a Vapor Cooled Shield (VCS). The existing H_2 tank utilizes an aluminum skirt which is a major contributor to the overall heat leak into the propellant tank (see Figure 2-1). An innovative low-Q skirt design was developed to reduce skirt heat leak to an acceptable level. Finally, for these very high performing insulation systems, the propellant lines can be a significant heat leak source. Selected penetrations will be shorted to the VCS to intercept heat that would otherwise enter the propellant tank. Figure 2-1 shows the current stage heat load, along with allocated requirements for long term missions.

Cryogen stages which must vent for long periods in orbit require a Thermodynamic Vent System (TVS) for low-g venting. It is convenient to use the VCS as a TVS to reduce or eliminate heat flow (Q) into the propellant tanks. MLI is required (100 layers is the near-optimum thickness for a 1-year mission) in order to reduce Q to the tanks by ~300 times (Figure 2-2). The VCS, properly placed within the MLI, will further reduce Q by a factor of 3. MLI system performance can be uncertain due to variations in installation; MLI performance from 15 references was used (Figure 2-2). A factor of 2 times the "undisturbed" performance was used which matches the data on demonstrated performance. "Undisturbed" performance is only obtained with calorimeter-type installations with no seams, or with NASA's Multi-purpose Hydrogen Test Bed where the MLI was rolled on without seams. A full-size stage would not be able to economically use this installation method.

Item	SOLAR ORIENTATION Nose Inertial Q (BTU/hr)	Tail Inertial Q (BTU/hr)	Side Inertial Q (BTU/hr)	EDS Performance Model Description	Q (BTU/hr)
H2 Tank	30,494	30,665	38,360		89.9
Forward skirt	5,930	5,151	7,311	35% of misc (allocation)	17.5
Forward dome	11,406	4,541	8,101	allocate by surface area	16.7
Cylinder	4,819	4,805	10,093	allocate by surface area	6.5
Aft dome	2,716	8,051	4,353	allocate by surface area	16.7
Aft skirt	4,941	7,354	7,308	35% of misc (allocation)	17.5
H2 Vent	23	23	23	10% of misc (allocation)	5.0
LH2 fill/drain line	9	9	9	10% of misc (allocation)	5.0
LH2 feedline	650	731	1,162	10% of misc (allocation)	5.0
H2 tank instrumentation	0	0	0	0% of misc (allocation)	0.0
O2 Tank	3,225	4,733	6,806		68.5
Forward ring	1,708	3,024	3,852	35% of misc (allocation)	17.5
Forward dome	136	199	263	allocate by surface area	6.4
Cylinder	126	126	292	allocate by surface area	5.7
Aft dome	138	139	261	allocate by surface area	6.4
Aft ring with "fingers"	1,024	1,150	2,040	35% of misc (allocation)	17.5
O2 Vent	9	9	9	10% of misc (allocation)	5.0
LO2 fill/drain line	8	8	8	10% of misc (allocation)	5.0
LO2 feedline	76	78	81	10% of misc (allocation)	5.0
		Reference			

Figure 2-1. Baseline On-orbit Heat Leak Summary

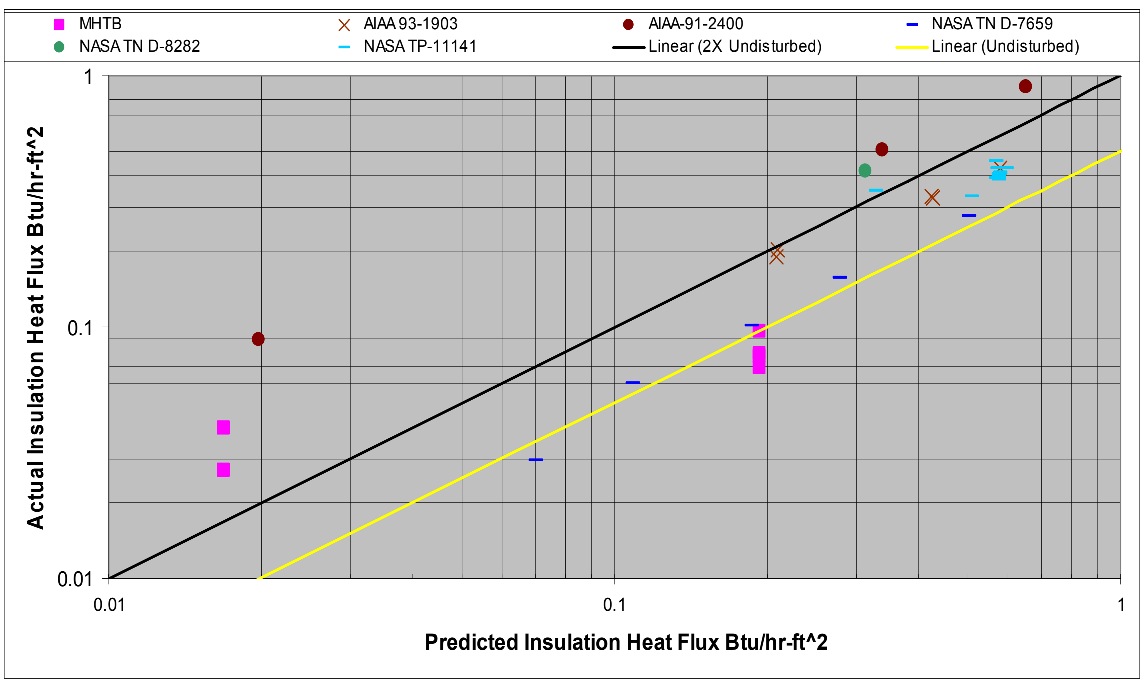

Figure 2-2. MLI Performance: Historical Test Data

Boeing's Design Sheet[R] is a model building and analysis tool which takes input algebraic equations and produces the computational plans and code needed to produce user-specified tradeoff studies. Its main strength is its flexibility: the user can, at run time, change the state of the model variables (dependent or independent) and desired tradeoffs, and Design Sheet automatically produces new tradeoffs without requiring any recoding by the user.

Design Sheet was used as an architectural trade study tool to evaluate performance optimums for VCS position within MLI, for shorting position to the VCS of plumbing and supports, and for venting configuration. The efficiency of Design Sheet allowed hundreds of cases to be run for system optimization.

Initially, allocations were used for the tank skirt (support) and plumbing heat leaks, as shown above in Figure 2-1. Detail designs were then developed. The current skirt is basically uninsulated aluminum, which is replaced by a skirt made of fiberglass epoxy sandwich material, integrated with the MLI blankets and VCS, which reduces the skirt Q to the LH_2 tank from 12,500 Btu/hr to about 9 Btu/hr.

The propellant fluid lines were assumed to be 304 stainless steel. The lines were assumed to be dry, which requires a tank-mounted pre-valve on the engine feedline (the vent and fill/drain lines already use close-coupled valves). The conductive length of the lines through the MLI was assumed to be 10 in., which experience has shown is easily achievable.

Two TCS configurations resulted from the Design Sheet optimization studies, which are described in detail below.

Independent

The independent, or "G", configuration consists of the EDS LH_2 tank and LO_2 tank, each with its own separate internal TVS, VCS integrated with MLI, and low-Q skirts/plumbing. The LH_2 tank vents H_2 through its TVS and VCS, and the LO_2 tank vents O_2 through its TVS and VCS. The vented O_2 and part of the vented H_2 are used in a fuel cell to provide 800 W of on-board power (Figure 2-3). The open loop fuel cell would use the constant venting of the H_2 and O_2 as reactants, with lithium-ion batteries to balance the system.

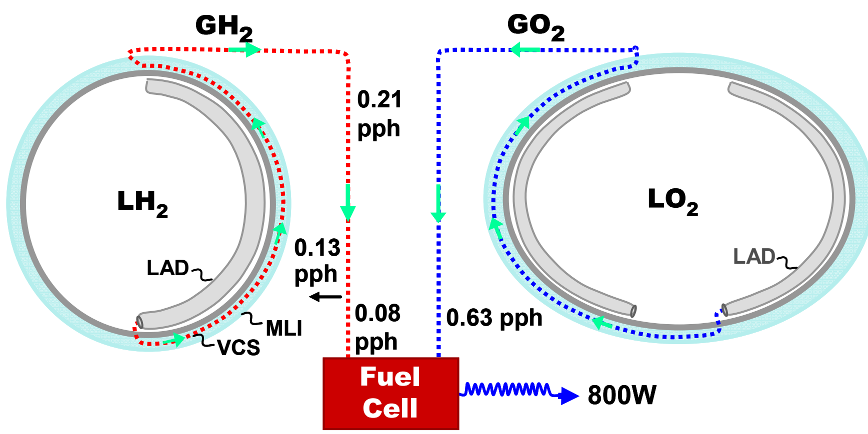

Figure 2-3. Independent TCS Configuration

Integrated

The integrated, or "I", configuration again consists of the EDS LH_2 tank and LO_2 tank, but only the LH_2 tank vents H_2 through its internal TVS and VCS integrated with its MLI. This vented H_2 is routed through a VCS integrated with the MLI around the LO_2 tank and this VCS is kept at ~170°R to intercept all incoming Q and keep the LO_2 tank vent free (Figure 2-4). Since only H_2 is vented, a fuel cell can't be used for power (supplying additional O_2 for fuel cell use is poor from a mass standpoint). Therefore a photovoltaic solar array, non-deployable and body-mounted to the EDS, was assumed for providing EDS power. Again lithium-ion batteries are used to balance the system.

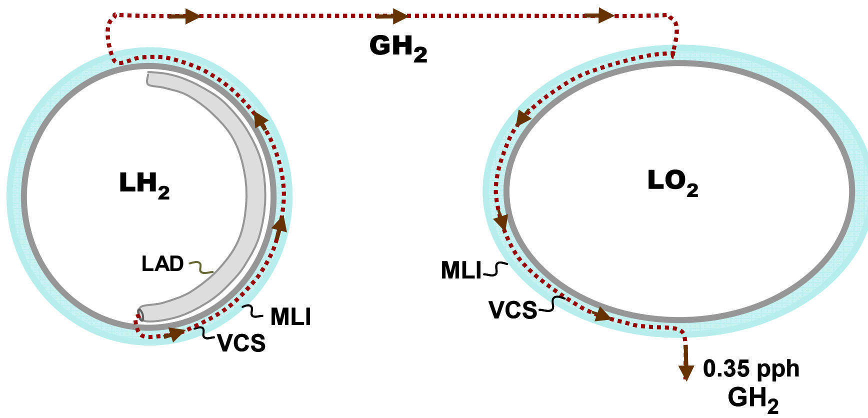

Figure 2-4. Integrated TCS Configuration

<u>VCS Design</u>

The Design Sheet model used for the TCS optimization assumed a constant temperature VCS with the TVS fluid exiting the VCS at the VCS temperature. This conservative assumption was adequate for the TCS optimization studies, but a more rigorous model was desired to determine detailed configuration influences on VCS performance. A SINDA/FLUINT VCS model was developed. The modeling approach was to divide each tank VCS into sections and then nodalize each (representative) section using SINDA, as shown in Figure 2-5 for a series routing scheme for TVS fluid in the VCS. On the left side of Figure 2-5 the VCS has been unwrapped from around the tank to be shown in two dimensions. The domes and cylinders are divided into 4 longitudinal sections, giving a total of 12 sections which are connected in series by the TVS flow line. The fluid processes in the line are modeled using FLUINT. Other flow schemes, such as parallel or manifold flow can also be analyzed using this nodalization method. The 12 panels can be thermally connected along their edges, or not, as desired.

The VCS section was then nodalized in detail using the node definition also shown in Figure 2-5. Each section is comprised of a 9 x 9 matrix of nodes with the center node replaced with 9 nodes for better spatial resolution near the TVS tube. Conductors are defined between adjacent nodes in both the width and length directions (there is only a single node in the thickness direction). The TVS tube nodes are connected to the thermal nodes along the center of the VCS section, as shown in Figure 2-6.

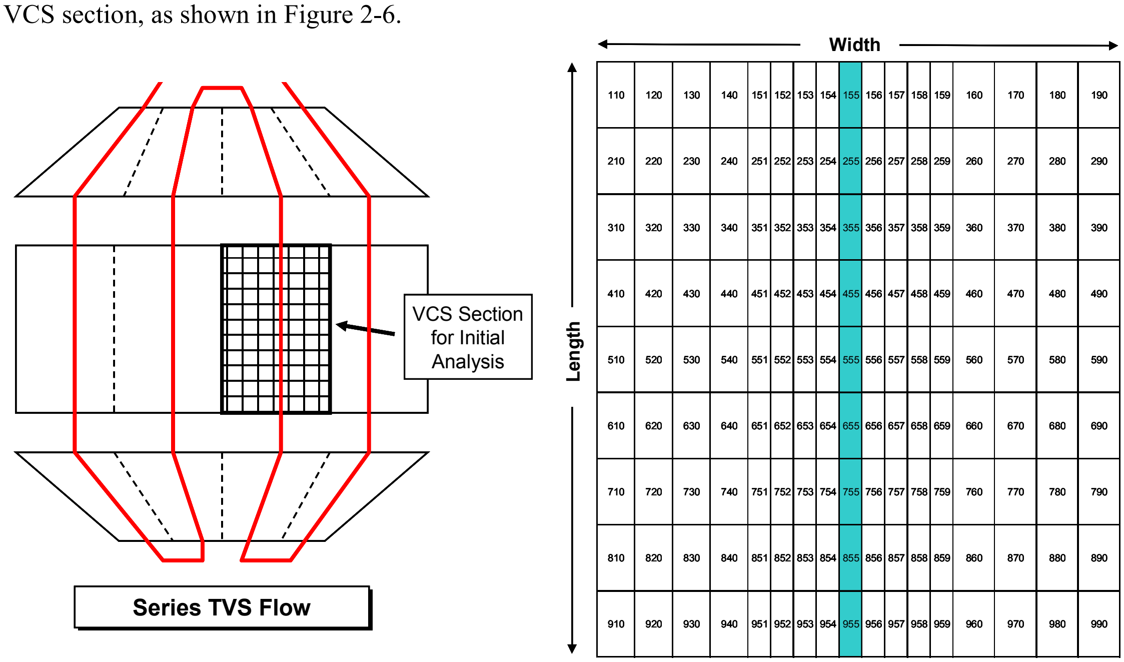

Figure 2-5. VCS Panels, Nodalization, and TVS Routing

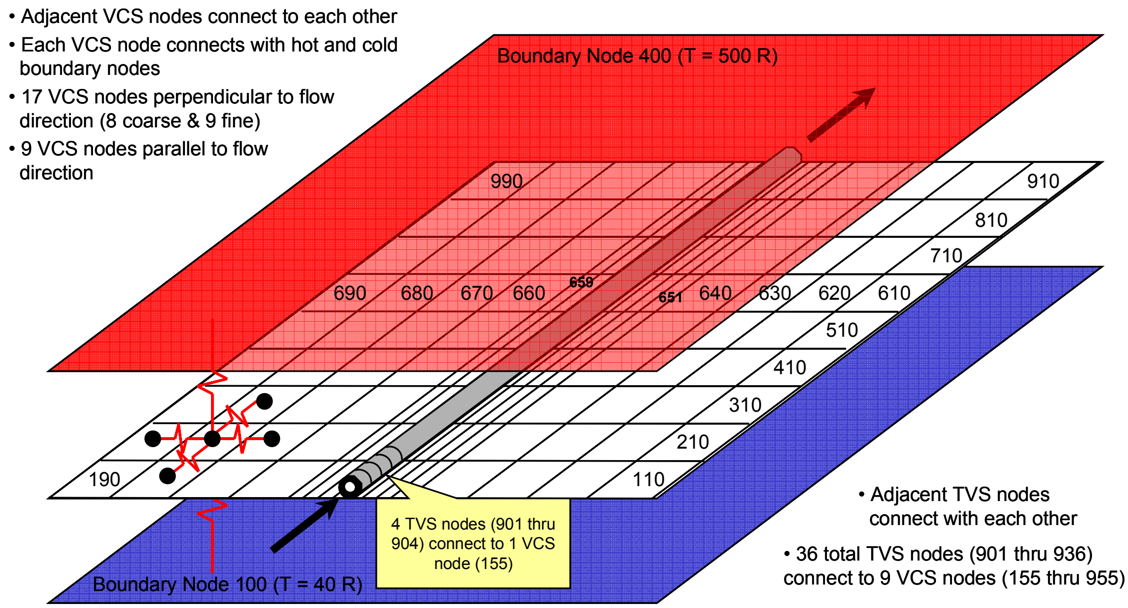

Figure 2-6. Single Panel Isometric View

The SINDA/FLUINT model was used to analyze the VCS performance and optimize the various configuration parameters, with results as shown in Table 2-3. Note that 100% contact between the TVS tube and the VCS is not required; as little as 11% contact (1 out of 9 nodes) only incurred a 7% penalty in thermal performance (heat flow to tank). The 4-pass parallel flow arrangement gave the best Q performance, and in all cases (both G and I, LH_2 and LO_2) the SINDA/FLUINT model gave 25-30% better VCS performance than the Design Sheet model, as expected.

Table 2-3. VCS Configuration Details

Trade Study	Result
VCS Thickness	0.010 inch
Tube Diameter	0.25 x 0.020 inch
Routing	Parallel
Tube Spacing	10 to 15 feet
Tube to VCS Contact	11% contact = 7% Q penalty

Tank Internal Components

The TVS/Liquid Acquisition Device (LAD) configuration internal to the propellant tanks is shown schematically on the left side of Figure 2-7. The LADs provide minimum continuous vent flow (~50-70% of that required) to the VCS. This flow passes through a Joule-Thomson expander which drops its temperature a few degrees and then enters a heat exchanger integrated with the body of the LAD (see right side of Figure 2-7). The LADs also provide intermittent parallel flow of vent fluid to a backup pumped heat exchanger which operates at ~10% duty cycle as needed to control tank pressure.

The LADs consist of a channel (triangular or rectangular) with fine mesh screens on the channel side facing the tank wall (Figure 2-7). Since the cryogens are wetting fluids, they will tend to be wall-bound in low-g, and the screens will tend to remain wetted. The TVS tube passes along the apex of the LAD channel and uses the LAD sides as heat exchanger fins. The cold TVS fluid keeps the LAD contents cold and improves LAD performance.

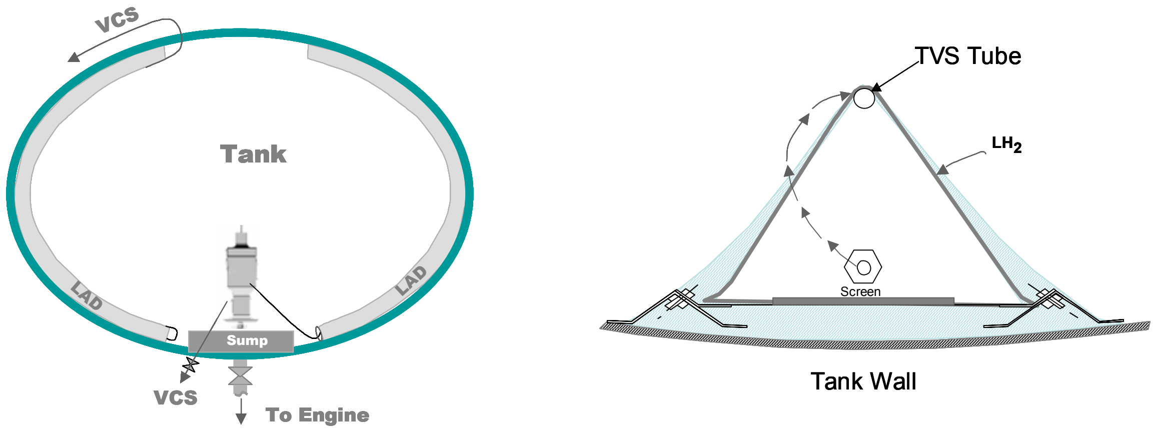

Figure 2-7. Thermodynamic Vent System (TVS) Schematic and LAD Configuration

III. Active Thermal Control Systems

Active TCS concepts were also studied, for comparison to the passive TCS concepts just described. These concepts include active refrigeration (a cryocooler), use of subcooled propellants, and hydrogen para-ortho conversion.

Active TCS/Cryocooler Concept

The same basic TCS arrangement studied for the passive TCS, including MLI, low-Q skirts/plumbing, and cooled shields was used. The shields were cooled by the cryocooler and the low-Q skirts/plumbing were shorted to the shields so that nominally, no heat entered the propellant tanks. A 22K shield surrounded the LH_2 tank, and a 95K shield surrounded the LO_2 tank. The Design Sheet heat flow analysis showed that with 278K external temperature the heat flow to the 22K shield was 6.8 W, and to the 95K shield was 66.5 W.

Because of the size of the loads, and the need to cool to 22K, a two-stage turboBrayton cryocooler was selected. This design was based on the Creare NICMOS cooler that has been flying on the Hubble Space Telescope for the last ~4 years. The turboBrayton cycle uses GHe as the working fluid and this cooled gas can be easily distributed to the loads (i.e. the 22K and 95K shields). The ACES cryocooler configuration, shown in Figure 3-1, has 3 compressors in series and 2 expansion turbines in parallel, one for the 22K load, and one for the 95K load.

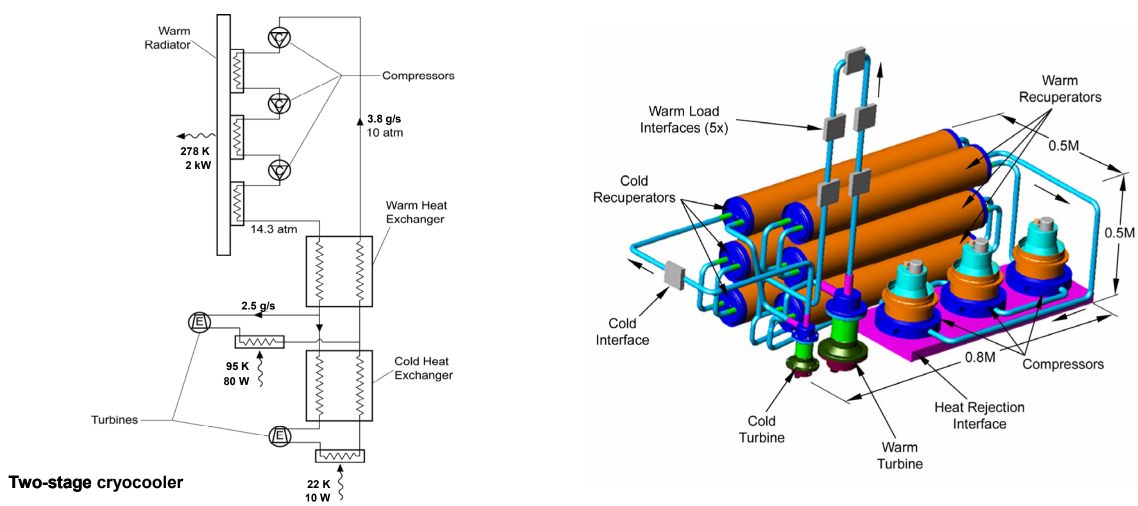

Figure 3-1. ACES Cryocooler Configuration and Layout

With this configuration and a modest scale-up of the NICMOS components, the ACES application provides reasonable margins on the heat loads: 10 W at the 22K load (47% margin), and 80 W at the 95K load (20% margin). The highest margin is provided at the coldest and least certain 22K load. Using the space-qualified technology flying on Hubble, the input power required is 2 kW; this could be reduced to 1.5 kW using advanced technology. The ACES cryocooler layout is also shown in Figure 3-1. The system mass is 60-85 kg depending on recuperator technology, and the volume is ~ 0.2 m^3.

Subcooled Propellants Concept

The concept of using subcooled propellants is based on cooling the cryogens to near their triple point. If the propellants are kept mixed, this allows the onset of venting to be delayed as the liquid sensible heat absorbs the incoming heat leak. Eventually, this incoming Q will warm the cryogens up to their saturation temperature (at nominal tank pressure) and venting will commence as required to maintain nominal tank pressure.

For the LH_2 tank, the temperature of the loaded LH_2 is dropped within a HEX to about 27°R (Psat = 2 psia). The cold bath in the LH_2 HEX is also LH_2 which is pumped on using a vacuum blower to reduce its temperature to ~ 26°R. This would be a large facility HEX which would boil about 15% of the loaded LH_2 quantity in order to subcool the propellant load. The vehicle would have to be continually topped off with subcooled propellant up until launch.

For LO_2 it is more convenient to use a LN_2 bath at atmospheric pressure and ~ 140°R to subcool the LO_2 to ~ 140°R (Psat = 3.2 psia). This is simpler than pumping on the LO_2 with a vacuum blower, as was necessary for the LH_2 tank.

Hydrogen Para-Ortho Conversion

It is well known that H_2 has two molecular forms: ortho, in which the nuclei spin in the same direction; and para, in which the nuclei spin in opposite directions. The equilibrium composition of these forms varies with temperature from ~100% para at 20K to ~25% para/75% ortho at room temperature. There is a difference in energy between ortho and para of 302 Btu/lbm, which is much larger than the 190 Btu/lbm required to vaporize LH_2. When ortho is converted to para, this energy is released; when para is converted to ortho, heat is absorbed from the surroundings to produce a significant cooling effect. This cooling effect manifests itself as an increased effective specific heat (as much as 80% higher at 100°R).

For this study, para-ortho conversion reduces the H_2 vent rate by 15-25% depending on whether TCS option G or I is used. Similar performance gains would be expected for other stage sizes. Thus it is very advantageous to convert the para H_2 to equilibrium ortho whenever the H_2 is used for cooling, as in a VCS. This para-ortho conversion occurs naturally very slowly, and catalysts must be used to speed this conversion. Such catalysts have been thoroughly developed by LH_2 producers who convert ortho to para in discrete steps during the LH_2 liquefaction process in order to avoid the slow release of heat from natural conversion (which would end up boiling off up to 60% of the produced LH_2.). A high performance ruthenium/alumina catalyst was developed under the National AeroSpace Plane program which requires only about 4 lbm catalyst/lbm/sec of vented H_2. For the very low vent flow rates of our vehicle, the catalyst requirements are less than 0.2 gram. Using the actual specific heats for equilibrium conversion, the Design Sheet analysis shows that para-ortho conversion saves 850 lbm of H_2 venting (25% saving) for the 365 day TLI mission using the integrated (I) configuration.

Concept Comparison

The results of the baseline independent (G) and integrated (I) TCS configurations with and without para-ortho conversion, together with subcooled propellants and using a cryocooler are shown in Figure 3-2. Note that for mission durations exceeding 180 days, the cryocooler has minimum mass, and that for the venting options, using the integrated (I) configuration with subcooled propellants and para-ortho conversion has minimum mass penalty.

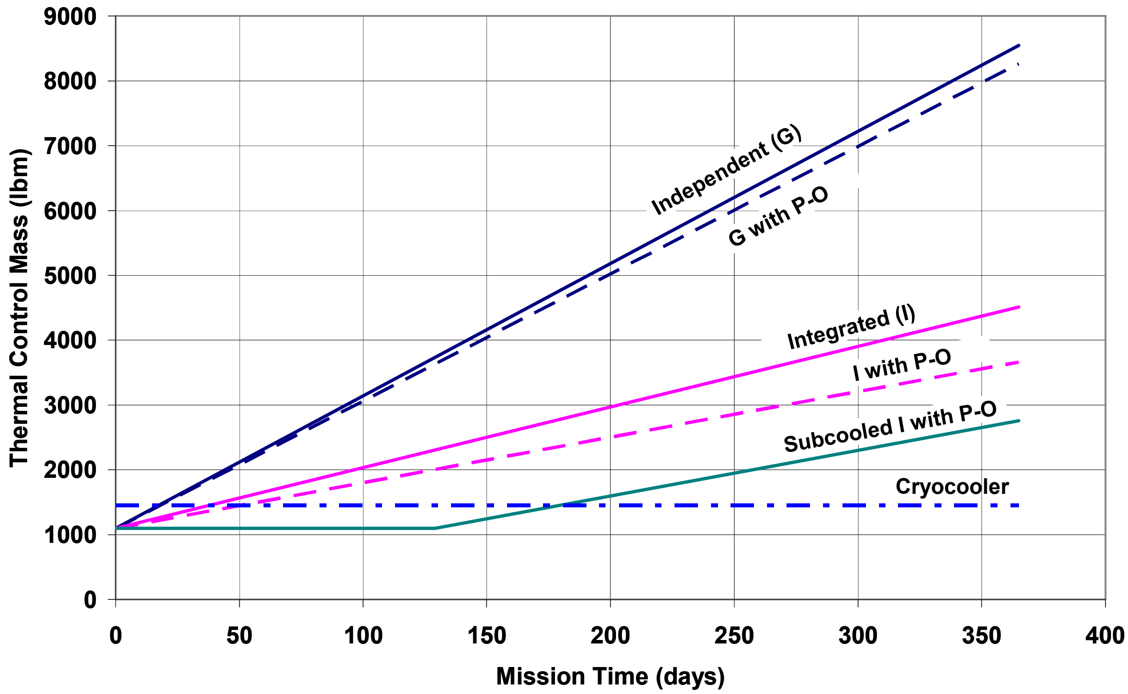

Figure 3-2. Boiloff Comparison Using Active Technologies

IV. Subsystem Impacts

The other major subsystems on the ACES vehicle were assessed to determine the impact of the long-duration EDS mission on subsystem component selection and performance.

Avionics

Most of the communications requirements will be satisfied using existing equipment except for the addition of Ka band and video to facilitate AR&C. The C&DH function will require upgrades to the Delta-proven RIFCA including expanded memory and throughput capacity. These all have low development risk, since they are extensions of existing hardware and software.

GN&C

Again, RIFCA upgrades for vehicle position and attitude will be required. AR&C requirements are all new since the current Delta IV second stage does not require this technology. The needed hardware and software are all similar to that being developed for Orbital Express.

Propulsion

The Main Propulsion System impacts are mainly in the tank internals where the TVS/LADs are substituted for the settling vent system on the current stage. The propulsion section also will be modified for larger or more engines (3 RL10s or 1 MB60) with the requirements for additional feed lines, valves, etc. For 3 RL10s, the engine expansion ratio must be reduced so that the engines will fit within the stage envelope; this reduces Isp by 2 to 7 sec depending on stage diameter.

The RL10 engine impacts were studied in detail by Pratt & Whitney Rocketdyne. It was found the RL10, originally a space-based design, was suitable for use on EDS with only minor design changes. A program for modifying and qualifying the engine was laid out, including nozzle extension design modifications and M/OD protection of the basic engine nozzle.

For the Attitude Control system (ACS), a comprehensive study was done of the thruster layout and number needed to satisfy ACS requirements. Based on this study, a 16-jet configuration placed near the vehicle CG, using 10-degree canted thrusters similar to the Apollo Service Module was selected. This configuration provided the following benefits:

1) lower cost from 16 thrusters, plumbing, components, vs. 20 or 24,
2) adequate fault tolerance; 1-fault tolerant translation and pitch/yaw control and 2-fault tolerant for roll control (additional fault redundancy may be available from other elements, LSAM, CEV),
3) ease of vehicle integration; all jets are in 4 packages mounted in the vicinity of the equipment shelf between the LO_2 and LH_2 tanks,
4) in the event of thruster failure, this configuration would use more propellant, but this penalty is believed small compared to integration mass savings,
5) plume impingement can be handled by canting and by protecting the aeroshell around the LH_2 tank.

This provides the minimum system capable of performing the GN&C and AR&C functions.

Power

Based upon the Power Subsystem trade studies performed, there are currently two configurations which have been identified as optimal for EDS use:

1) A solar array with rechargeable battery (suitable for use with the "integrated" TCS option where only H_2 is vented),
2) A fuel cell operating on H_2 and O_2 boiloff (suitable for use with the "independent" TCS option).

The benefits and issues associated with each configuration are summarized in Table 4-1.

Table 4-1. EDS Power System Concepts

Power System Option	Benefits	Issues
Fuel Cell + Batteries	▪ Dual use of vent fluid ▪ Attractive packaging and integration	▪ Not available for zero-boiloff TCS arrangement ▪ No power available after propellant depletion
Array/Battery	▪ Proven technology ▪ Independent of TCS configuration ▪ No consumables required	▪ Stage integration

Structures

Since micrometeorite/orbital debris (M/OD) protection is not required for the current stage due to its short time in orbit, this will be a major technology development. The LH_2 and LO_2 tanks will be adequately protected by the aeroshield and MLI/VCS surrounding the tanks. The Attitude Control System propellant tanks and gaseous helium bottles will be protected by the addition of Multi-Shock blankets. The main engine nozzle protection scheme has yet to be developed and could represent a significant contributor to M/OD vulnerability. There will also be a trade-off with when a deployable nozzle is deployed. Early deployment eliminates any issues associated with the nozzle deployment mechanism after a long on-orbit standby period. Early nozzle deployment also 1) eliminates any protection that the extendable cone provides to the powerhead, and 2) increases the vulnerability of the extendable exit cone as it is no longer partially hidden behind the stage (assuming the EDS is oriented with the long axis in the flight direction with the engine aft).

The high heat leak skirts on the propellant tanks will be replaced with fiberglass composite skirts to significantly reduce heat leak. The intertank and equipment shelf trusses will also likely be redesigned to accommodate the new skirts. These structures will require extensive development and testing because of their high risk level.

Mechanisms

Separation of the EDS from the launch vehicle is similar to what is in use currently. The development of an EDS-to-EDS docking system is new since no docking system exists today that is easily adaptable to the Delta IV second stage. However, its development is based on the designs and approaches of existing mechanisms. The topology of the recommended designs (Stewart platform topologies) has flight heritage, is well understood, and has been modeled and simulated extensively. The main issue is the increased size (4-5 m) and careful development, simulation, and testing will be required; there is adequate precedent on the Shuttle, International Space Station and Orbital Express programs.

Technical Readiness and Risk

The technical readiness assessment for the EDS subsystems follows the TRL guidelines established by NASA. Most of the TRLs are fairly high (6-9) since many of them are derivations of currently flying items. The main exceptions are the long term storage TCS and the low heat leak skirts. These have been validated in a laboratory or relevant environment (e.g. thermal/vacuum chamber), but have never flown on an operational launch vehicle.

A list of 61 EDS risks was compiled and ranked by risk exposure level; the top risks were 1) 60K engine development, 2) engine section arrangement of 3 RL10s, 3) truss structure to support 3 RL10s, 4) uncertainty in low heat leak support skirt thermal/structural performance, and 5) deployment/capture failure of docking mechanisms.

V. EDS Performance

The vehicle payload performance for a TLI/LOI mission of 300 days was analyzed and is shown in Figure 5-1. As shown, all of the cases, with their high-performance TCS, give better performance than the initial performance baseline. The H_2 vent flow for both the G and I configurations is a direct payload penalty, but is less than that which would cause the engine mixture ratio limit to be reached, causing the precipitous payload drop.

On the other hand, the large O_2 boiloff of case G (and H) is a much larger direct payload penalty and, relative to case I, incurs a payload decrement of nearly 1 metric ton. Although the independent (G) configuration has the desirable quality of venting both H_2 and O_2 which could be used to generate about 800 watts from a fuel cell, 1 t seems a heavy mass penalty to pay to avoid the complexity of installing solar panels on the EDS (which would be needed if the integrated (I) configuration, which vents only H_2, was selected). Since an overall system trade of this venting/power issue has not yet been done, both G and I configurations were retained for future analysis.

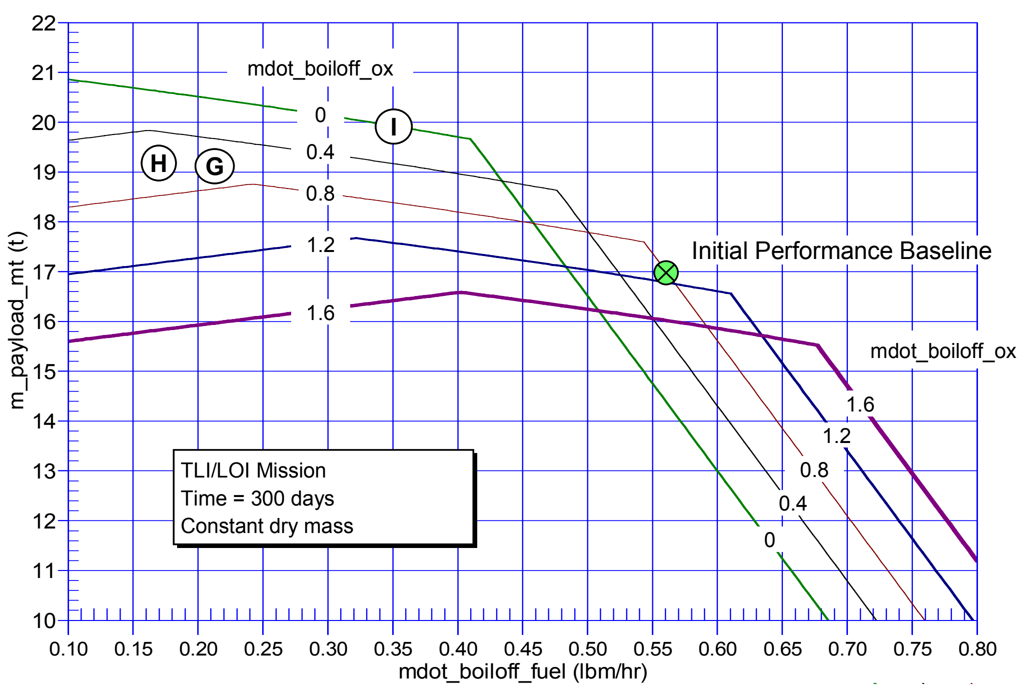

Figure 5-1. Effect of Boiloff on Payload (TLI/LOI-300 days)

VI. Conclusions

A Delta IV second stage derivative vehicle, suitable as an EDS for long-duration TLI/LOI missions, was comprehensively studied. The TCS needed to allow a one-year EDS mission was defined and analyzed. Details of the TCS were designed and Design Sheet and SINDA/FLUINT analyses were developed to determine system performance. The other affected EDS subsystems, such as avionics, GN&C, AR&C, propulsion, power, and structures were studied and technology development needs identified. With reasonable development, an EDS based on the Delta IV second stage could be developed by the 2014 timeframe.

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