# HUMAN MARS MISSION

## SEP Architecture
## Crew Taxi Propulsion Stage Study and Design
## And
## Technology for Reaction & Control System

# FINAL REPORT

REF: Order Number H-28653D
(Part I)

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Submitted to:
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# CREW TAXI PROPULSION STAGE STUDY AND DESIGN
## INTRODUCTION

The Mars exploration is a candidate pathway to expand human presence and useful activities in the solar system. There are several propulsion system options being considered to place the Mars payload on its interplanetary transfer trajectory. One propulsion option is the use of Solar Electric Propulsion (SEP) to spiral out with the Mars payload from an initial Low Earth Orbit (LEO) to an elliptical High Earth Orbit (HEO).

This report, presented in annotated facing page format, describes the work completed on the design of a crew taxi propulsion stage used in conjunction with the SEP. Transportation system/mission analysis topics covered in this report include sub-system analysis, trajectory profile description, mass performance and crew taxi stage sizing, stage configuration, stage cost, and Trans-Mars Injection (TMI) launch window.

The high efficiency of SEP is used to provide the major part of the TMI propulsion maneuver. Orbital energy is continuously added over a period of approximately twelve months. The SEP and Mars payload follow a spiral trajectory from an initial LEO to a final elliptical HEO. A small chemical stage is then used to provide the final part of the TMI. The now unloaded SEP returns to LEO to repeat another spiral trajectory with payload to HEO.

The spiral phase of the SEP's trajectory takes several months to reach HEO, thus significantly increasing the exposure time of the crew to zero-gravity. In order to minimize the long zero-gravity effects, a high thrust chemical stage delivers the crew to the SEP's HEO. The crew rendezvous with the Mars payload in HEO. After a checkout period the Mars payload with the crew is injected onto a Trans-Mars Trajectory by a small chemical stage.

# *Human Mars Mission*
# *SEP Architecture*
# *Crew Taxi Propulsion Stage Study and Design*
## Study and Design Team

| | |
|---|---|
| Mission Analysis | L. Kos/A. Young |
| Thermal | R. Alexander |
| Structures | A. Philips |
| Shuttle/Launch Vehicle Performance | J. Hays |
| Crew Taxi Stage Sizing/Performance | V. Dauro |
| Propulsion Systems | B. Price |
| Operations | R. Christenson |
| Avionics (CD&H, etc) | R. White |
| Power | R. White |
| Configuration | M. Gerry |
| GN&C (RCS, AR&C) | V.Dauro |
| Mass Properties | B. Brothers |
| <u>Programmatic Support</u> | |
| Cost | M. Naderi |
| Schedule | R. Shepard |

# CREW TAXI PROPULSION STAGE STUDY AND DESIGN
## STUDY/DESIGN OBJECTIVES

There are four significant study/design objectives listed here which provided the prime focus in working the Crew Taxi Propulsion Stage Study and Design task. The configuration design objective was to design an efficient and effective stage to deliver the crew to SEP's high elliptical orbit from low earth orbit. Supporting mission analysis data was developed to determine mission requirements and performance. With support of other NASA centers and industry, material was developed on the crew taxi launch window, X-38 mass properties, and how the crew taxi will be flown on the Shuttle.

# *Human Mars Mission*
# *SEP Architecture*
# *Crew Taxi Propulsion Stage Study and Design*

## Study Objectives

- Develop crew taxi configuration design
- Develop supporting mission analysis data
- Integrate supporting centers' crew taxi data
- Address how crew taxi will be flown on the shuttle

# CREW TAXI PROPULSION STAGE STUDY AND DESIGN REQUIREMENTS

The Crew Taxi Propulsion Stage will be designed to meet the requirements listed here in a safe and reliable way. The stage propellant will be transferred from the Shuttle's ET after the Shuttle has achieved a safe altitude of 160,000 ft.

# *Human Mars Mission*
# *SEP Architecture*
# *Crew Taxi Propulsion Stage Study and Design*

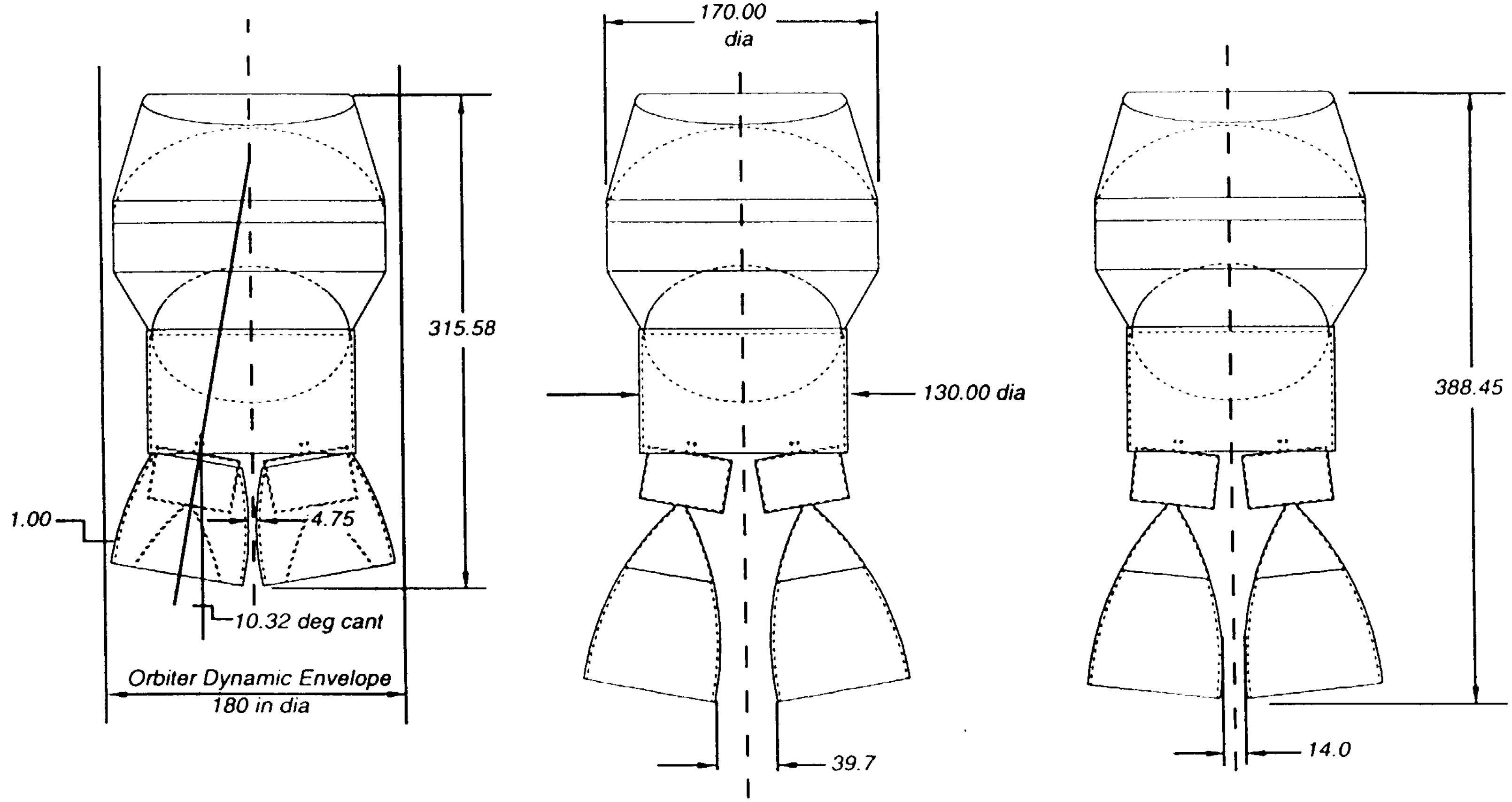
## Requirements

- Accommodate a crew module with a crew of 6
- Deliver the crew & module to the SEP highly elliptical Earth orbit ( 800km X 70,764km )
- Perform terminal phase rendezvous maneuver
- De-orbit the crew module for Earth entry
- Crew taxi propulsion stage final disposition will be expended in Earth's atmosphere
- Propellant transfer from ET at safe altitude

# $LOX/LH_2$ Crew Taxi Propulsion Stage Concept

A conceptual design for a $LOX/LH_2$ crew taxi propulsion stage (CTP S) is shown on the facing page. The stage is designed to be accommodated by the STS Orbiter and therefore was limited to a diameter of 170 inches. Two RL-10 type engines with retractable nozzles are utilized. The nozzles are canted 10.32 degrees to facilitate center-of-gravity tracking. The stowed length is 315.6 inches (engine bells retracted) and the deployed length is 388.5 inches.

# HMM Crew Taxi
# Engine Placement




Stowed In Orbiter

Nozzles Deployed

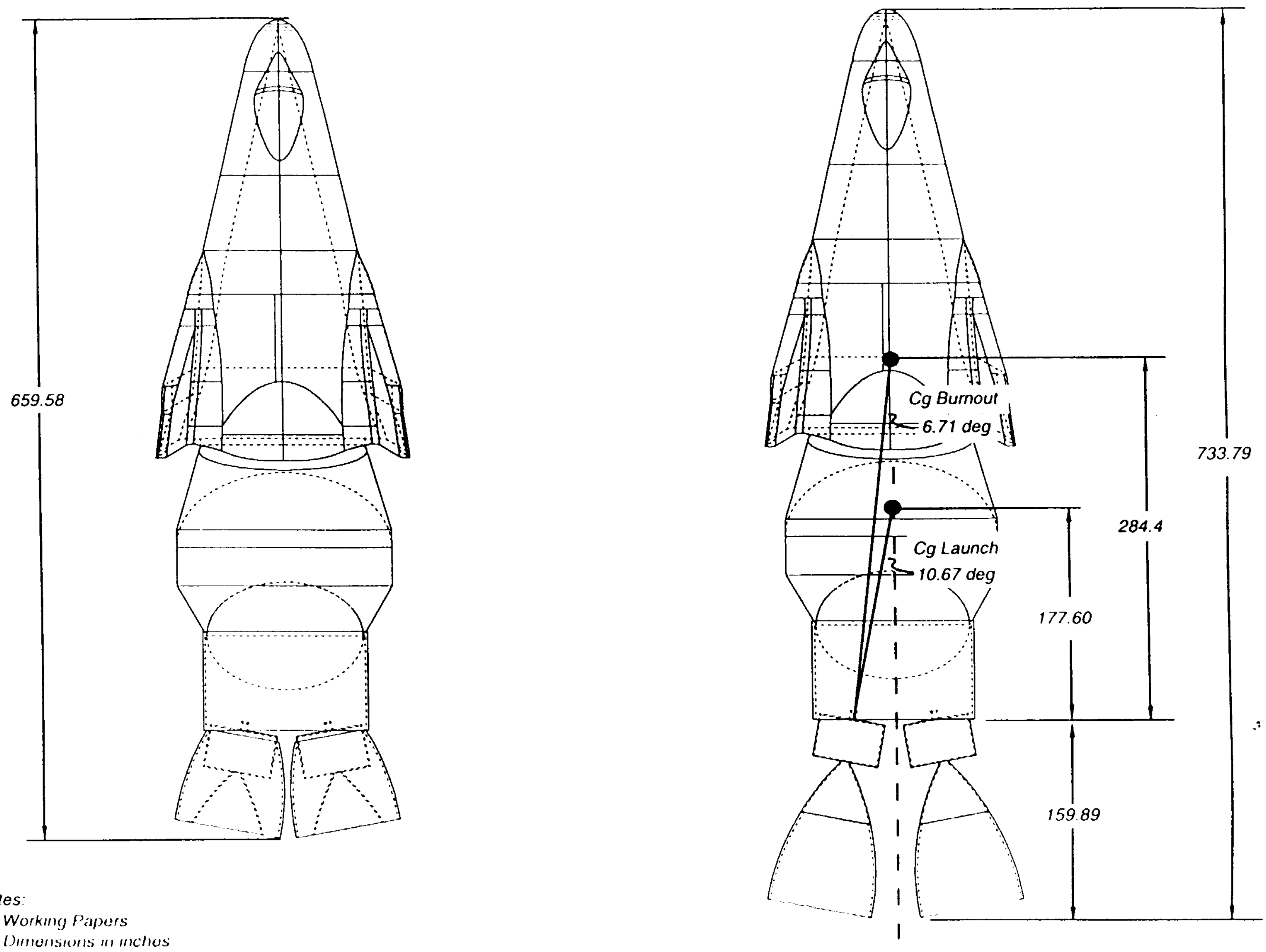
Nozzles Rotated
4 deg inboard

Notes:
(1) Working Papers
(2) Dimensions in inches

# Crew Taxi & CT Propulsion Stage Configuration

The facing page shows the X38-derived crew taxi (CT) integrated with the crew taxi propulsion stage. The overall length is 659.6 inches with engine bells retracted and 733.8 inches with engine bells deployed. The center-of-gravity at launch is located 177.6 inches forward of the engine gimbal point. As propellant is consumed the center-of-gravity moves forward until burnout in which it is located 284.4 inches forward of the engine gimbal point. The engines gimbal 4 degrees to accommodate tracking.

# HMM Crew Taxi

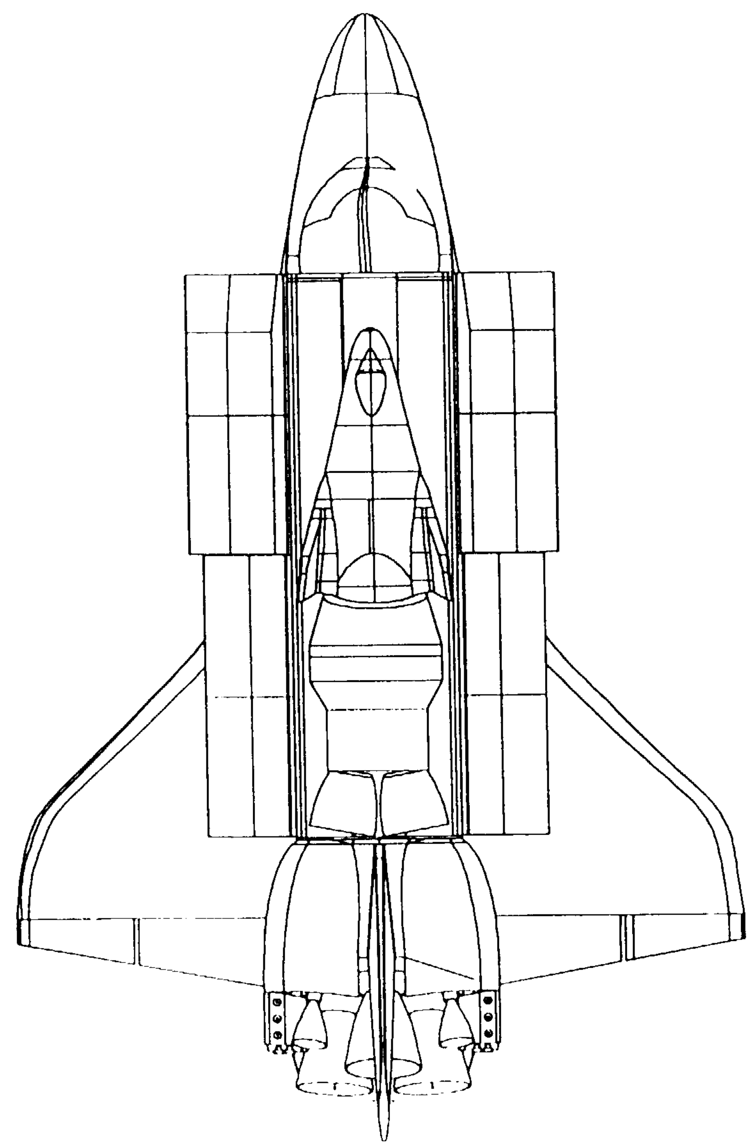



Notes:
(1) Working Papers
(2) Dimensions in inches

# Orbiter, Crew Taxi, & CT Prop Stage Concept

The crew taxi and its propulsion stage is shown integrated into the Orbiter payload bay. The payload bay is 15 feet in diameter and 60 feet in length. The airlock interface will be part of the mid-deck to accommodate this payload configuration.

# HMM Crew Taxi
# Launch Configuration



SEP ARCHITECTURE
CREW TAXI LOX/LH2 STAGE PLUS X-38

Cargo bay length margin with EVA access, X-38 and taxi stage is 0.5 ft; this information is shown on the left-hand side of the figure. The total weight at liftoff of the above cargo bay items is 32,717 lbs. The Shuttle capability to the crew taxi LEO is 36,950 lbs for OV-102 and 43,950 lbs for OV103/4/5. The payload margin for OV-102 is 4,233 lbs and for OV103/4/5 is 11,233 lbs.

# SEP Architecture
## Crew Taxi Lox/LH2 Stage plus X-38

| Total Length in Orbiter Cargo Bay (ft) | Total Weight at Liftoff (lbs) | Shuttle Capability to 200 x 278 km 51.6 Inclination (lbs)* | Shuttle Landing Weight Limit (lbs)** |
|---|---|---|---|
| 3.0 EVA Access | 7,000 ASE | 36,950 (OV-102) | 49,000 (OV-102) |
| 32.0 X-38 | 7,970 Taxi Stage (Dry) | 43,950 (OV-103/4/5) | 56,000 (OV-103/4/5) |
| 24.5 Taxi Stage | 16,424 X-38 | | |
| 59.5 Total Length | 1,323 Crew | | |
| 0.5 Margin | 32,717 Total Weight | | |
| 60.0 Cargo Bay Length | | | |

*with 29,845 lbs Propellant to fill Taxi Stage

**Data from:
Tom Dickerson
TD31

Archie Young
Alpha Technology
6/9/99

# CREW TAXI STAGE HMM SEP DRM v4.0a WEIGHT BREAKOUT

The crew taxi weight breakout into the component level of each subsystem is given on this chart. Futher information is contained for some items as to the number of each item or in some cases the item may consists of several smaller items which are listed by weight and number in the microsoft excel macro cell where each weight is computed and shown. This information may be seen using the computer generated microsoft file that is available. The first weight shown is the total dry mass of the vehicle (including residuals and rcs propellants) which was used along with the total usable propellants (shown next) to check the performance of the vehicle with the other stages to insure the mission could be completed successfully. The misc. / margin shown next tells how the substem weights have increased or decreased since the performance computations were made. The major systems are shown with components which add up to give the total for that system. The HMM team member who derived the component weights is shown out to the right side of the chart. A contingency of 10 percent is added for all the dry weights(excluding the RL10 engines) which should be adequate at this time since all the equipment and structures were sized and estimated from exsisting materials and technology. The stage mass fraction is included at the bottom of the chart

HMM DRM v4.0a CREW TAXI STAGE

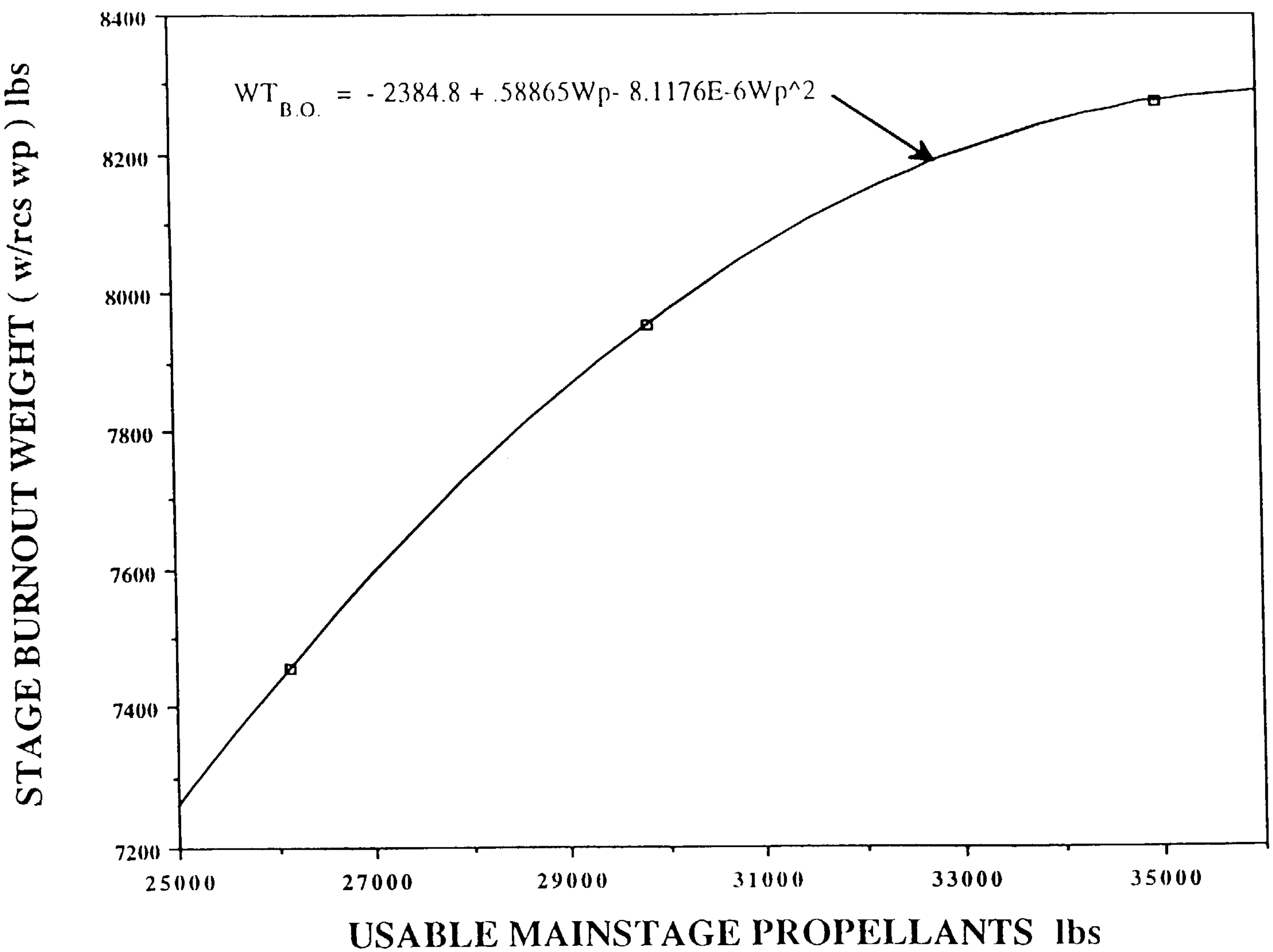
| | | | | | |
|---|---|---|---|---|---|
| CREW TAXI S | Dry Mass w/rcs prop (mt) | | | 3.848 | Perm Koss |
| | LOx / $LH_2$ Propellant Mass (mt) | | | 13.538 | Perm Archie |
| | Misc. / Margin | | | **0.016** | |
| LOX/LH2 | Propulsion System | | | 0.689 | |
| | pressurant tanks(5) | | | 0.095 | Prop Brian |
| | valves,filters,regulars & misc | | | 0.098 | Prop Brian |
| ISP=466 sec | RL10B-2 engines(2) | | | 0.463 | Prop Brian |
| | RCS thrusters(24) | | | 0.033 | Prop Brian |
| | Boil-off prior to start | | | 0.000 | |
| | RCS Propellant | | | 0.147 | Perm Vince |
| | Residuals inc He | | | 0.224 | Centaur/Brian |
| | Thermal | | | 0.207 | |
| | lox mli | | | 0.015 | Therm Reggie |
| | lh2 mli | | | 0.142 | Therm Reggie |
| | inter thermal control | | | 0.050 | Therm Reggie |
| | Main engines gimbal system(TVC) | | | 0.070 | sys Bot Reggie |
| | GN&C and AR&C | | | | |
| | Avionics | | | 0.075 | Avionics Ron |
| | remote data units(2) | | | 0.018 | Avionics Ron |
| | data bus coupler(18) | | | 0.003 | Avionics Ron |
| | tvc controller(4) | | | 0.018 | Avionics Ron |
| | tvc actuators(4) | | | 0.036 | Avionics Ron |
| | Power | | | 0.404 | Power Ron |
| | power distr unit(3) | | | 0.135 | Power Ron |
| | battery(6) | | | 0.162 | Power Ron |
| | battery(3) | | | 0.027 | Power Ron |
| | wiring | | | 0.080 | Power Ron |
| | Tankage | | | 0.859 | STR AL |
| | lh2 tank (mt) | | | 0.396 | Str AL |
| | lox tank (mt) | | | 0.463 | Str AL |
| | Structures | | | 0.885 | Str AL |
| | fwd skirt (mt) | | | 0.228 | Str AL/Koss |
| | aft skirt(mt) | | | 0.175 | Str AL |
| | thrust structure(mt) | | | 0.267 | Str AL |
| | stage misc (mt) | | | 0.214 | Str AL |
| | Contingency (10%) | | | 0.273 | Sys Bobby |
| | | | | 0.779 | |

conv kg lb = 2.204622622

## CREW TAXI PROPULSION STAGE SCALING EQUATION

This chart was generated to produce a scaling equation for the crew taxi propulsion stage. The data was generated and plotted to cover the expected usable propellant range the vehicle would require which was from 26,000 to 35,000 pounds.The data was curved fitted and produced the weight at burnout of the crew taxi stage which is 0.58865 times the weight of usable propellants minus 8.1176E-06 times the weight of usable propellants squared minus 2384.8 pounds.This data was provided to the performance analysis team members to size the crew taxi stage.

# CREW TAXI PROPULSION STAGE




Brothers 7/8/99

# SEP Crew Taxi Propulsion Stage - Structures

The structural design of the crew taxi is similar to the Centaur G'. The larger hydrogen tank on top has a conical section which necks down to the smaller oxygen tank on the bottom. The bulkheads between the two tanks are nested, with the one on the hydrogen tank side being inverted. Unlike the Centaur G' however, the stage is not pressure stabilized. It was designed using isogrid stiffeners where needed.

The stage was sized using UpStart, an in house developed upper stage sizing program written in Visual Basic. UpStart is a component of the LVA (Launch Vehicle Analysis) computer program. For the construction materials, a psuedo-isotropic lay-up of graphite epoxy IM7/8552 was used. This material is hydrogen compatible. For liquid oxygen compatibility, a spray deposited internal coating was assumed for the oxygen tank.

# SEP Crew Taxi Propulsion Stage - Structures

- Structural design similar to Centaur G'
  - Nested domes
  - Isogrid stiffening
    - Not pressure stabilized
- Sized using UpStart, an in house developed upper stage structural sizing program
  - Graphite epoxy IM7/8552

| SEP Crew Taxi Propulsion Stage - Structural Weights | |
|---|---|
| Forward Skirt | 128.4 kg |
| LH2 Tank | 396.2 kg |
| LOX Tank | 463.2 kg |
| Aft Skirt | 175.2 kg |
| Thrust Structure | 267.0 kg |
| Misc | 214.5 kg |
| | 1644.4 kg |

# SEP Crew Taxi Propulsion Stage - Thermal

The thermal control system for the SEP crew taxi propulsion stage is based on the thermal control system designs of the Centaur G' and some other upper stage studies. These designs utilized passive insulation for the cryogenic tanks.

The $LH_2$ tanks for the crew taxi stage uses a combination of 3.81 cm thick foam insulation for pre-launch thermal control and twenty layers of multi-layer insulation (MLI) for on-orbit thermal control. On-orbit thermal control is necessary to minimize the amount of propellant boil-off prior to propellant usage. The number of MLI layers was optimized based on total thermal system mass which is the sum of the MLI mass and the associated boil-off. The $LO_2$ tanks require only twenty layers of MLI to minimize boil-off on-orbit. Foam insulation in not required for pre-launch conditioning on the $LO_2$ tank. The two propellant tanks have nested bulkheads requiring 1.91cm of a fiberglass mat insulation between the bulkheads to minimize the heat transfer between the tanks.

The thermal control for the other components consists of foam insulation on the propellant lines, heaters and insulation for RCS propellants, and MLI to insulate the avionics.

Shown on the facing page is the mass estimate of the external thermal control system which consists of cryogenic tank insulation, and a mass estimate for the internal thermal control which consists of the insulation, and heaters for the propellant lines and avionics.

# *SEP Crew Taxi Propulsion Stage - Thermal*

- Results
  - Based on Centaur G´, Advanced Shuttle Upper Stage (ASUS) Study, and Boeing High Energy Upper Stage (HEUS) Designs
    - $LH_2$ Tank
      - Combination of 3.81 Cm Thick Foam Insulation for Pre-launch Thermal Control and ~ 20 Layers of MLI for On-obit Thermal Control
    - $LO_2$ Tank
      - Only MLI (20 Layers)
    - Insulation Between Nested Bulkheads
      - Fiberglass Mat Insulation
    - Internal Thermal Control
      - Foam Insulation For Lines, Heaters, Avionics Insulation, Etc.

  - Mass Estimate

| Thermal Control System Mass (kg) | |
|---|---|
| External | 157.4 |
| Internal | 49.9 |
| Total | 207.3 |

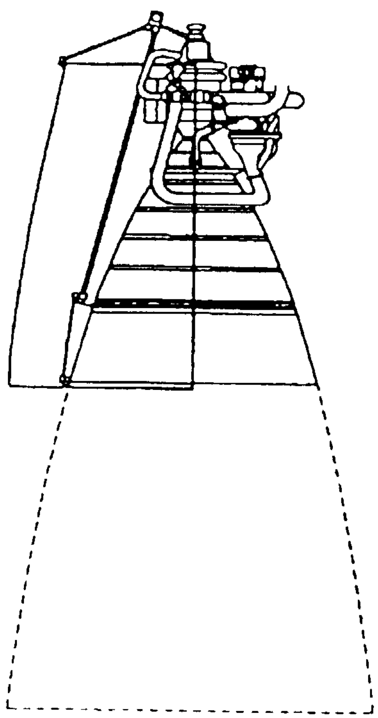
## HUMAN MARS MISSION: SEP V4.0A - 2011/2014

### RL10B-2 UPPER STAGE ENGINE

Summarized are the performance and physical characteristics of the RL10B-2 engine. The performance characteristics are nominal values. The engine employs a translatable nozzle extension. The engine does not have throttling capability but can perform multiple restarts. Depending on the coast time and the thermal environment, restart times can range between 2 and 15 seconds. At most, pump chill down time can be as long as 188 seconds. Propellants are $LOX/LH_2$.

# Mars Chemical Stage Engines

## RL10B–2 Upper Stage



### RL10B–2 Characteristics:

| | | |
|---|---|---|
| Vacuum Thrust, lbf | 24,750 $lb_f$ | 110,100 N |
| Vacuum Specific Impulse, sec | 465.5 | 465.5 |
| Mixture Ratio | 5.88 | 5.88 |
| Chamber Pressure, psi | 633 | 4.36 MPa |
| Area Ratio | 285 | 285 |
| Pump Inlet Press., Start/Run | Nom. psi | kPa |
| Oxidizer | 28.0 | 193 |
| Fuel | 27.0 | 186 |
| Mass, lbm | 510 | 231 kg |
| Thrust to Weight Ratio | 48.5 | 48.5 |
| Dimensions, in | | |
| Installed Length | 64.1 | 1.63 m |
| Extended Length | 163.5 | 4.15 m |
| Diameter | 84.1 | 2.14 m |
| Gimbal Angle, ° (square pattern) | 4 | 4 |

# HUMAN MARS MISSION: SEP V4.0A - 2011/2014
# CREW TAXI STAGE PROPULSION SYSTEM MASS SUMMARY

Presented is a propulsion system component mass summary for the crew taxi stage. For the most part, component masses are based on existing hardware. At this point in the study, however, the reaction control mass is a best estimate. The 10 percent miscellaneous includes plumbing, mounting hardware, and brackets. Propellant tank and associated structural masses are not included. Those values are provided by structural analysis.

The propulsion system schematic for this stage is identical to that of other Mars mission stages that use liquid oxygen (LOX) and liquid hydrogen ($LH_2$) as propellants. Regulated helium gas provides LOX tank pressurization. The budgeted amount of helium, and hence the number of helium storage tanks, is dependant on the required LOX load plus the additional helium required for engine functions. The fuel tank is pressurized autogenously with hydrogen gas supplied from the engines. To satisfy an engine out requirement, the stage employs two Pratt & Whitney RL10B-2 engines for main propulsion.

The crew taxi stage is proposed launched to low earth orbit (LEO) by the space shuttle. The stage is launched empty with in-flight propellant fill from the shuttle main propulsion system. The mass summary reflects the stage propulsion system mass at deployment from the orbiter. Orbiter/stage interface hardware mass for the most part is chargeable to the orbiter, however, any additional propulsion system interface hardware mass for the stage at this point is not identified.

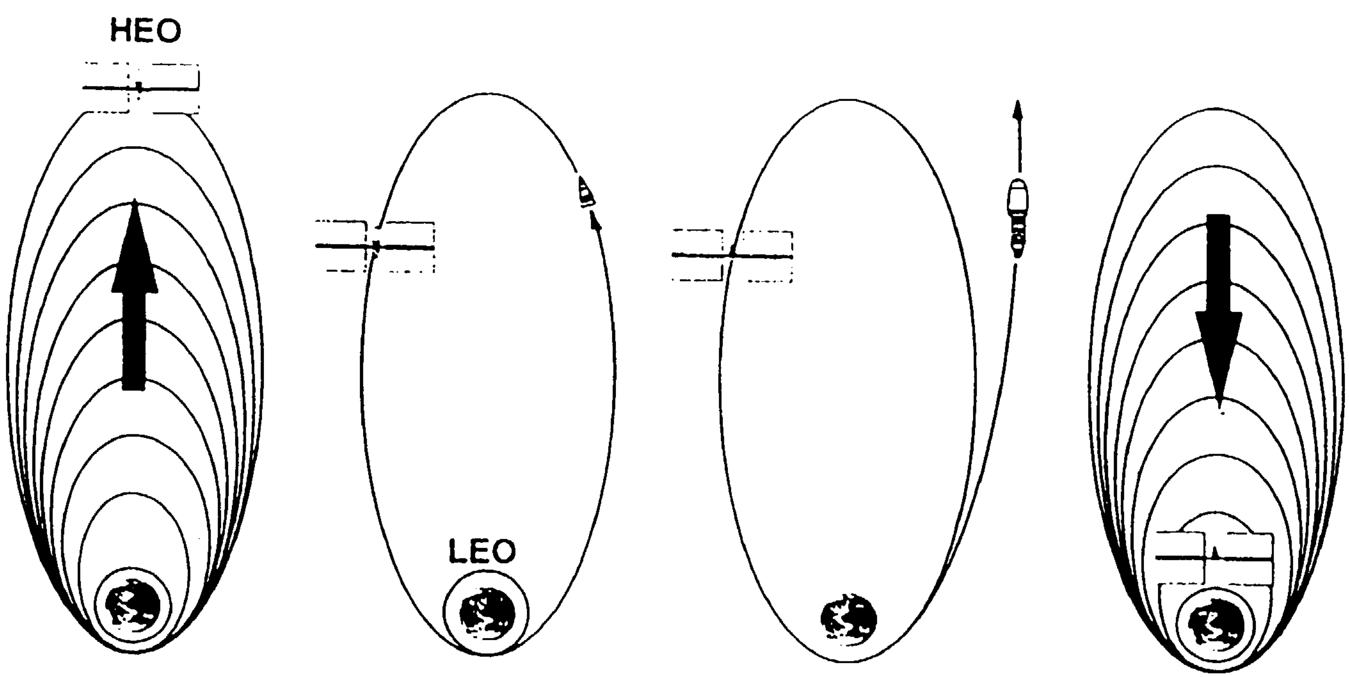
# SEP V4.0A - 2011/2014
# MARS MISSION CREW TAXI STAGE
# PROPULSION SYSTEM MASS SUMMARY

| UNIT WT (LB) | PROPULSION SYSTEM COMPONENTS | LOX/LH2 Crew Taxi NUMBER | LOX/LH2 Crew Taxi WT (LB) |
|---|---|---|---|
| 42 | PRESSURANT TANK | 5 | 210.0 |
| 3 | ISOLATION VALVE, PRESSURIZATION | 7 | 21.0 |
| 0.6 | TEST PORT, FILL & DRAIN VALVE (HE) | 7 | 4.2 |
| 0.6 | FILTER, HIGH PRESSURE | 2 | 1.2 |
| 2.5 | PRESSURE REGULATOR | 2 | 5.0 |
| 0.5 | PRESSURE RELIEF VALVE | 2 | 1.0 |
| 1.2 | QUAD CHECK VALVE | 1 | 1.2 |
| 1.2 | FILL & DRAIN VALVE | 4 | 4.8 |
| 5 | ISOLATION VALVE, PROPELLANT | 8 | 40.0 |
| 510 | ENGINE | 2 | 1020.0 |
| 3 | THRUSTER, REACTION CONTROL | 24 | 72.0 |
| 10% | MISCELLANEOUS | | 138.0 |
| | TOTAL DRY WEIGHT | | 1518.4 |
| | PROPELLANT, USABLE | | 29851.3 |
| | RESIDUAL PROPELLANT | | 609.2 |
| | PRESSURANT GAS, HE | | 55.0 |
| | TOTAL SUBSYSTEM WEIGHT | | 32033.9 |

## SOLAR ELECTRIC PROPULSION (SEP) MISSION CONCEPT

An overview of the SEP mission concept is shown in this figure. The SEP transfers the Mars bound cargo to High Earth Orbit (HEO) from a Low Earth Orbit (LEO) over a period of about twelve months. The HEO is achieved by the SEP's low thrust spiral trajectory. The crew of six is delivered to the SEP's HEO in a X-38 derived vehicle by a high energy, high thrust propulsion stage. After the crew is transferred to the Mars bound cargo, the Trans-Mars Injection is performed by a chemical propulsion system. The SEP returns to LEO for refueling and reuse.

# SOLAR ELECTRIC PROPULSION MISSION CONCEPT




Electric Propulsion (EP) space tug performs low-thrust transfer for Mars-bound cargo to High Earth Orbit (many months transfer)

Crew delivered in "small" chemically-propelled transfer vehicle - X-38 derived (few days rendezvous time)

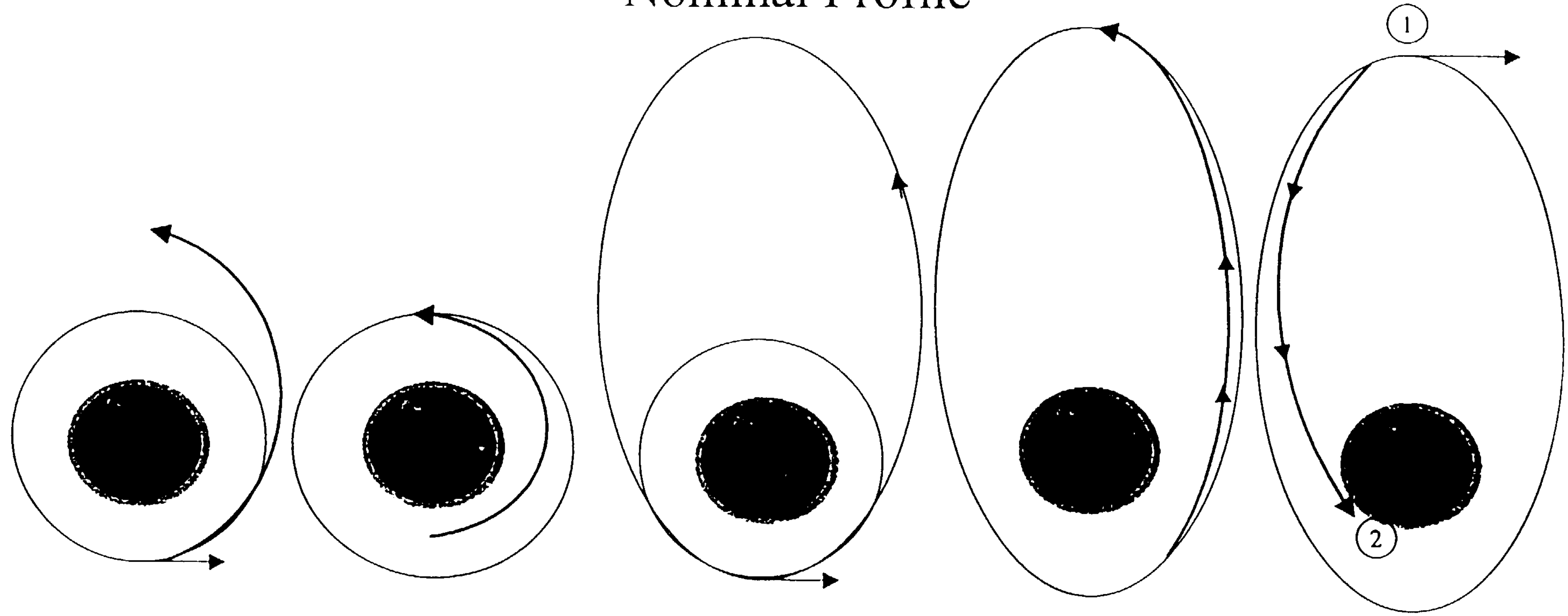
Remainder of trans-Mars injection performed by chemically-propelled system

Space tug returns for refueling and next assignment (faster or more efficient return since no payload present)

# CREW TAXI PROPULSION MANEUVERS
## NOMINAL PROFILE

The nominal profile for the crew taxi propulsion maneuvers are shown where the stage departs from a 200 X 278 km altitude orbit 60 hours and 45.50 minutes after shuttle launch. From the 200 X 278 km orbit, the crew taxi goes into a 200 X 750 km orbit. From the 200 X 750 km orbit, the stage circularizes into a 750 X 750 phasing orbit. TPI is executed 63 hours and 46.80 minutes after shuttle launch. The TPI orbit is 750 X 70081; from this orbit TPF is performed, resulting in the SEP's orbit of 800 X 70379 km. After the crew taxi rendezvous and docks with the SEP's payload, the X-38 is de-boosted into a return to Earth trajectory.

# CREW TAXI PROPULSION MANEUVERS
## Nominal Profile




→ 200 X 278 KM
200 X 750
HR:MIN 60 : 45.50
RAISE APOGEE
BURE TIME (B.T.) : 13.9 SEC

→ 200 X750
750 X 750
61 : 32.50
CIRCULARIZE
B.T. : 15.5 SEC

→ 750 X 750
750 X 70081
63 : 46.80
TPI
B.T. : 3 MIN 21.5 SEC

→ 750 X 70081
800 X 70379
65 : 57.20
TPF
B.T. : 1.4 SEC

1 800 X 70379
→ 1 X 70289
76 : 8.40
DEBOOST
T.B. : 2.2 SEC
2 RETURN TO EAETH
87 : 22.60

# Crew Taxi Propulsion Stage Sizing

This page shows the echoed input and output from the Computer Orbital Performance Analysis (COPA) program, written by J. McCarter/TD54 and J. Mulqueen/TD54. The X-38-derived payload is shown at 7450 kg (7.45 mt), the crew at 0.6 mt, RCS propellent including that required for the Automatic Rendezvous and Capture (AR&C) at 0.15 mt, and a placeholder for additional interface (I/F) mass or margin at 0.1 mt. The delta velocities ($\Delta$Vs) shown here were obtained from V. Dauro/Alpha Technology using his Integrated Mission Program (IMP) model of the CT mission. A more detailed explanation of the CT mission will follow in a later section.

The maximum propellant load that can be packaged in a 15' x 60' cargo bay with the previously shown X-38-derived crew delivery taxi, and using a partially nested LOx tank, is 13.5 mt. Since the current mission does not require this full propellant load, a $\Delta$V performance margin for accommodating an orbital launch window has been computed to be 436 meters/second (m/s) or, equivalently 1.2 mt of $LOx/LH_2$ propellant. The mass budget shown here is 3601 kg, not including the RCS prop and I/F margin, so effectively the stage mass budget is 3848 kg. The mass at shuttle lift-off will be 12.1 mt (total mass minus the off-loaded propellant) plus the airborne support equipment (ASE) required to fit both the CT and the CTPS in the cargo bay and deploy it once on orbit.

## MISSION DESCRIPTION

Case: CT_July99

| EVENT DESCRIPTION | TIME | ACTIVE STAGE | DELTA MASS | DELTA VEL | ISP | BOILOFF RATE |
|---|---|---|---|---|---|---|
| Crew Taxi | 0. | 1 | +7450. | | | 0.01 |
| Add 6 Crew | 0. | 1 | +600. | | | |
| Add AR&C Prop. | 0. | 1 | +147. | | | |
| Interface | 0. | 1 | +100. | | | |
| Back off | 0. | 1 | | 1.0 | 999. | |
| Transfer | 0.06 | 1 | | 131.6 | 466. | |
| Circularize | 0.1 | 1 | | 151.7 | 466. | |
| SEP Rendezvous | 0.19 | 1 | | 2638.7 | 466. | |
| Orbital Window | 0.69 | 1 | | 436. | 466. | |
| Orbit Adj. | 0.69 | 1 | | 7.7 | 466. | |
| Orbit Adj. | 0.69 | 1 | | 0.8 | 999. | |
| Drop Crew | 0.69 | 1 | -600. | | | |
| Drop AR&C RCS | 0.69 | 1 | -112. | | | |
| Deorbit | 1.68 | 1 | | 48.8 | 466. | |
| Separtation | 1.72 | 1 | | 0.2 | 999. | |
| DV Margin | 2. | 1 | | 60.0 | 466. | |

## SCALING EQUATIONS

| STAGE | A | B | C |
|---|---|---|---|
| 1 | 3601. | 0. | 0. |

## STAGE MASS SUMMARY

| STAGE | MASS AT 1st IGN | BURNOUT MASS | PROPELLANT CAPACITY | TOTAL STAGE MASS | MASS FRACTION |
|---|---|---|---|---|---|
| 1 | 25435.7 | 3601.0 | 13537.7 | 17138.7 | 0.789891 |

## THRUST EVENT SUMMARY

| Description | Delta V | Start Mass | Prop Used | End Mass |
|---|---|---|---|---|
| Back off | 1.0 | 25.436 | 0.003 | 25.433 |
| Transfer | 131.6 | 25.433 | 0.722 | 24.711 |
| Circularize | 151.7 | 24.711 | 0.807 | 23.904 |
| SEP Rendezvous | 2638.7 | 23.903 | 10.485 | 13.418 |
| Orbital Window | 436.0 | 13.416 | 1.221 | 12.195 |
| Orbit Adj. | 7.7 | 12.195 | 0.021 | 12.175 |
| Orbit Adj. | 0.8 | 12.175 | 0.001 | 12.174 |
| Deorbit | 48.8 | 11.457 | 0.122 | 11.335 |
| Separtation | 0.2 | 11.335 | 0.000 | 11.335 |
| DV Margin | 60.0 | 11.334 | 0.148 | 11.186 |

## CREW TAXI PROPULSION STAGE REUSABILITY

The Crew Taxi Propulsion Stage (CTPS) was sized for maximum propellant load that can be packaged in a 15' x 60' cargo bay with the X-38 derived crew delivery taxi. Using a partially nested LOX tank the maximum propellant load is 13.5 mt. The nominal mission profile, which expends the CTPS in the Earth's atmosphere after deorbiting the X-38 to return to earth landing does not use the total 13.5 mt of propellant. A Computer Orbital Performance Analysis (COPA) program run was made to determine the required propellant load to return the CTPS to the International Space Station (ISS) orbit, where the CTPS could be refurbished for reuse. The required propellant load to return the CTPS to ISS orbit is 18.1 mt.. The CTPS configured to hold the 18.1 mt would exceed the 15' x 60' cargo bay envelope when packaged with the derived X-38. In order for the CTPS to fly in a reusable mission profile the derived X-38 would have to be delivered to the ISS Orbit on a separate shuttle flight.

## MISSION DESCRIPTION    Case: CT_reusable

| EVENT DESCRIPTION | TIME | ACTIVE STAGE | DELTA MASS | DELTA VEL | ISP | BOILOFF RATE |
|---|---|---|---|---|---|---|
| Crew Taxi | 0. | 1 | +7450. | | | 0.11 |
| Add 6 Crew | 0. | 1 | +600. | | | |
| Add AR&C System | 0. | 1 | +405. | | | |
| Interface | 0. | 1 | 0. | | | |
| Crew Taxi miniDVs | 0.1 | 1 | | 284.42 | 466. | |
| CrewTaxi Rendez. | 0.2 | 1 | | 2645.44 | 466. | |
| Orbit adj. | 0.7 | 1 | | 6.06 | 466. | |
| Deorbit | 1.7 | 1 | | 51.50 | 466. | |
| DV Margin | 2. | 1 | | 60. | 466. | |
| Orbital Window | 2. | 1 | | 0.047 | 466. | |
| Drop Crew | 2. | 1 | -600. | | | |
| Separation | 2.5 | 1 | -7450. | | | |
| SS Rendezvous | 2.5 | 1 | | 25.5 | 466. | |
| Circularization | 3. | 1 | | 2700 | 466. | |

## SCALING EQUATIONS

| STAGE | A | B | C |
|---|---|---|---|
| 1 | -1081.7 | 0.58865 | -0.000017896 |

## STAGE MASS SUMMARY

| STAGE | MASS AT 1st IGN | BURNOUT MASS | PROPELLANT CAPACITY | TOTAL STAGE MASS | MASS FRACTION |
|---|---|---|---|---|---|
| 1 | 30268.5 | 3709.7 | 18104.5 | 21814.2 | 0.829941 |

## THRUST EVENT SUMMARY

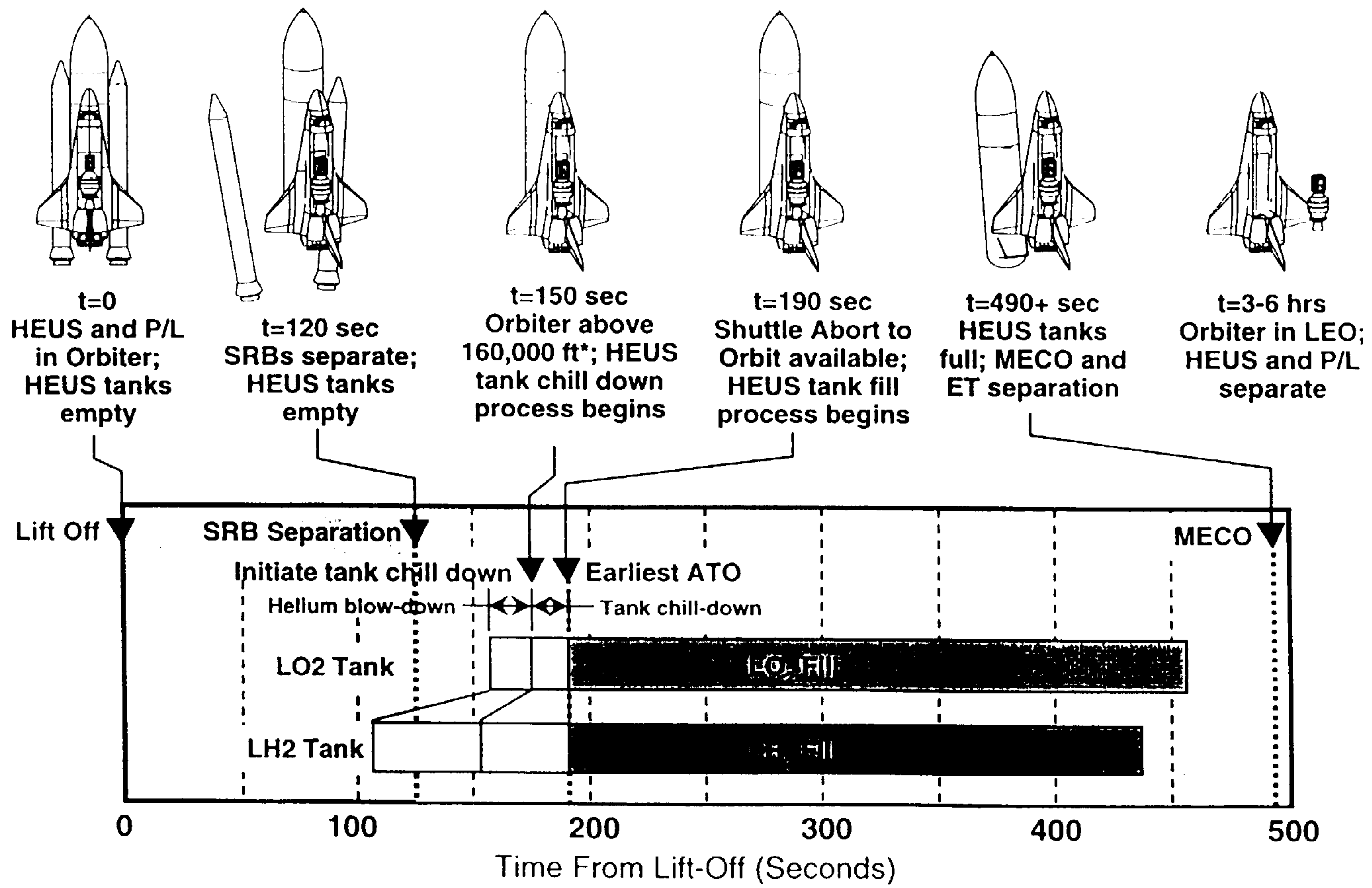
| Description | Delta V | Start Mass | Prop Used | End Mass |
|---|---|---|---|---|
| Crew Taxi miniDVs | 284.4 | 30.269 | 1.826 | 28.442 |
| CrewTaxi Rendez. | 2645.4 | 28.441 | 12.499 | 15.942 |
| Orbit adj. | 6.1 | 15.939 | 0.021 | 15.918 |
| Deorbit | 51.5 | 15.912 | 0.178 | 15.734 |
| DV Margin | 60.0 | 15.732 | 0.205 | 15.527 |
| Orbital Window | 0.0 | 15.527 | 0.000 | 15.526 |
| SS Rendezvous | 25.5 | 7.473 | 0.042 | 7.432 |
| Circularization | 2700.0 | 7.429 | 3.314 | 4.115 |

## HEUS MISSION SEQUENCE MAINTAINS SAFE, RELIABLE SHUTTLE OPERATIONS

The sequence of events during shuttle ascent is shown to maintain safe and reliable operations for the $LO_2$ and the $LH_2$ propellant transfer from ET to the Crew Taxi Stage (High Energy Upper Stage, HEUS) the stage propellant tank fill process begins after the shuttle has reached abort to orbit capability. The propellant transfer process takes about five minutes to fill the Crew Taxi Stage tanks.

This chart was provided by Boeing North American.

# HEUS Mission Sequence Maintains Safe, Reliable Shuttle Operations



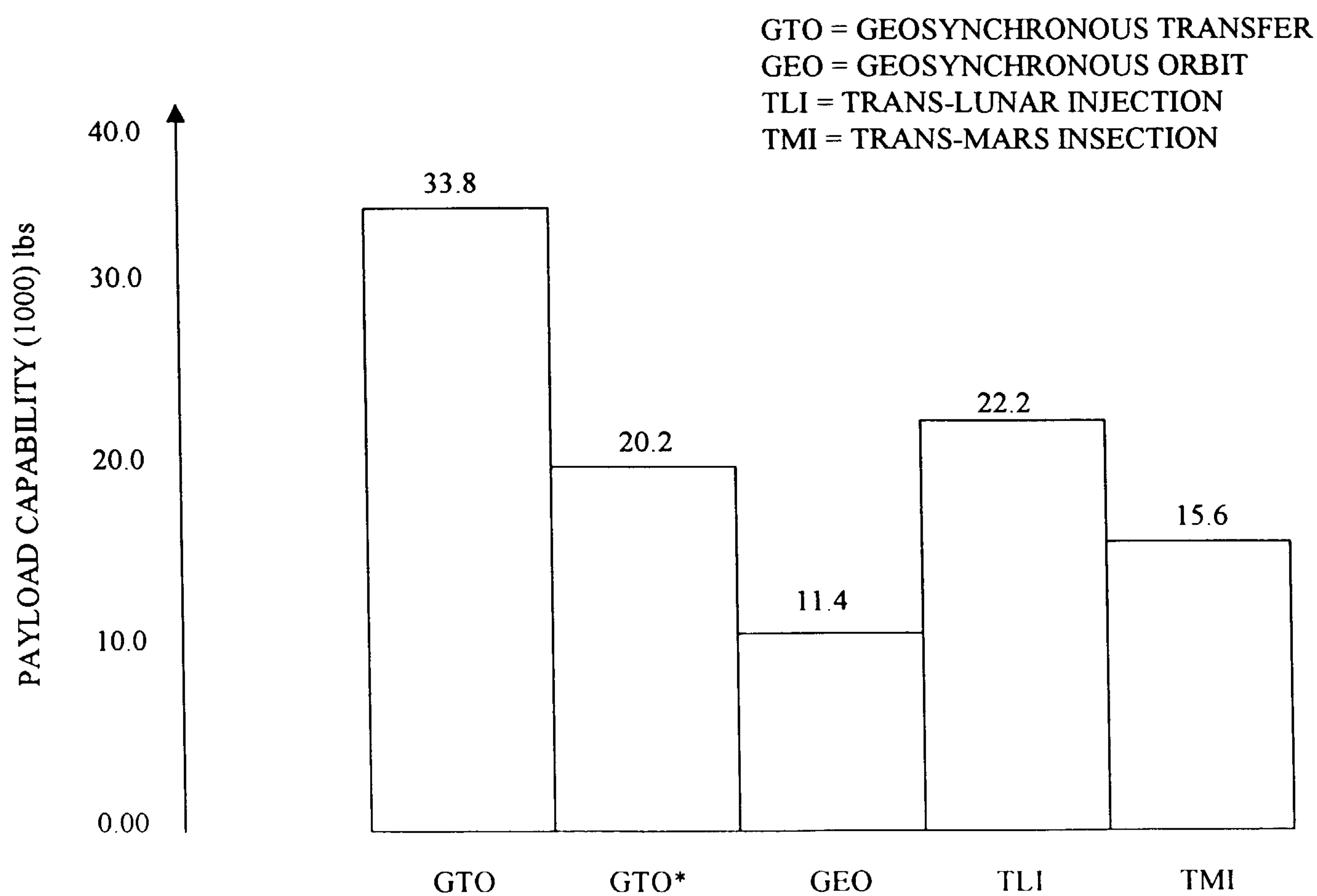

* Safe altitude for $H_2$ and $O_2$ propellants; combustion cannot be sustained at ambient pressure

# CREW TAXI STAGE PERFORMANCE CAPABILITY TO OTHER MISSION

The Crew Taxi Stage has excellent potential to deliver payload to other missions. Payload delivery capability for four different missions is given. These missions include: Geosynchronous Transfer Orbit (GTO), Geosynchronous Earth Orbit (GEO) Trans-Lunar Injection and Trans-Mars Injection. Two data points are given for GTO which includes 33,800 pounds delivery capability where the stage is expended and 20,200 pounds capability where the stage is returned to Low Earth Orbit (LEO) for reuse.

For comparison, the Inertial Upper Stage (IUS) will deliver 5,000 pounds to GEO; this compares to 11,400 pounds for the Crew Taxi Stage.

# CREW TAXI STAGE PERFORMANCE CAPABILITY TO OTHER MISSIONS

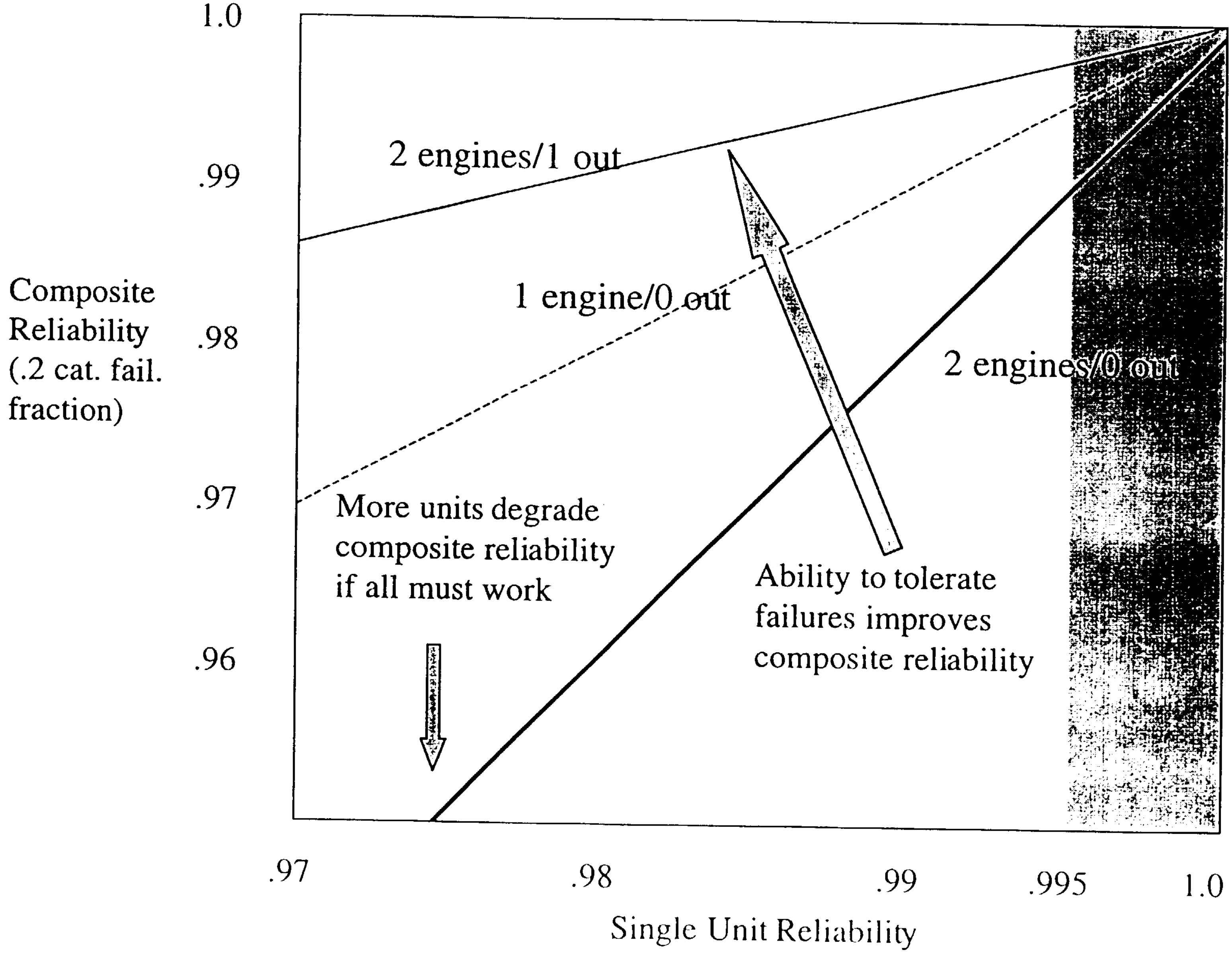



* STAGE RETURNS TO LEO

# ENGINE OUT ANALYSIS (cont.)

This chart presents an engine out reliability comparison of three cases: 1). 2 engines with 1 engine out capability, 2). 1 engine with no out capability, and 3). 2 engines with no out capability. The lines for these three converge rapidly to the right in this graph. Several conclusions may be drawn. First, more units degrade composite reliability, if all must work. Second, the ability to tolerate failures improves the composite reliability, given this model. Third, and as stated earlier, as single unit reliability improves, the overall composite reliability benefit diminishes. Comparing the top two lines on the graph reflects that at some TBD point, the benefit to engine out capability may not be clear. In this graph the area at 0.995 and above is shaded. In this region, the benefit to engine out diminishes to the point that performance or operations issues favoring fewer engines may take precedent.

# ENGINE OUT ANALYSIS (cont.)



# MARS TRANSPORTATION SYSTEM

## CREW TAXI COST ESTIMATE SUMMARY

The cost section of this report includes a cost estimate of the Crew Taxi with it's associated groundrules and assumptions and a cost estimate of the Advanced Shuttle Upper Stage which was developed in previous studies with it's associated groundrules and assumptions. The Crew Taxi cost estimate includes the development and the unit cost of the Crew Taxi stage by subsystem. The significance of the Advance Shuttle Upper stage cost estimate is that the avionics and the Main Propulsion System (MPS) modifications cost required for the Space Shuttle Transportation System to enable the in flight propellant transfer capability could be used as a bench mark for estimating the cost of in flight propellant transfer capability for the Crew Taxi stage.

# Human Mars Mission (HMM)

## Crew Taxi Preliminary Cost Estimating Groundrules And Assumptions

- All costs are in millions of FY 1998 dollars.

- NAFCOM96 (NASA/Air Force Cost Model) was used to develop the cost estimate for the HMM Crew Taxi stage.

- The cost estimate includes the DDT&E cost and the unit cost for the Crew Taxi Stage of HMM.

- Apollo LM was used as the anolog for costing all the Crew Taxi stage subsystems with the exception of the propellant tanks.

- ET LH2 tanks were used as the most analogous data point to cost the Crew Taxi propellant tanks.

- It is assumed that the HMM Crew Taxi stage will be a new design with no inheritance from the previous NASA programs with the exception of the main engines.

- The main engines of the Crew Taxi stage are RL-10A derivitives and assumed to be existing.

- The New Ways Of Doing Business (NWODB) cost savings factors have been applied to the system level costs to reflect the differences in the management of the today's NASA programs from the historical NASA programs.

- The cost estimate includes a 30% contingency to account for uncertainties in program, a 15% program support to account for all additional costs to the program office beyond the prime contractors cost, and a 10% fee for the prime contractor.

# Human Mars Mission

## Crew Taxi Preliminary Cost Estimate

### (All Costs Are in Millions of FY 1998 Dollars)

| WBS Element | Weight | DDT&E | Flight Unit | Total | D&D Cmplx. | D&D Inher. | Unit Cmplx. |
|---|---|---|---|---|---|---|---|
| **GRAND TOTAL** | **6756** | **$4,081.9** | **$183.0** | **$4,264.9** | | | |
| **SYSTEM 1: CREW TAXI STAGE** | **6756** | **$4,081.9** | **$183.0** | **$4,264.9** | | | |
| **HARDWARE TOTAL** | **6756** | **$1,207.4** | **$89.0** | **$1,296.4** | | | |
| STRUCTURES (APOLLO LM) | 1730 | $171.5 | $18.0 | $189.5 | 1 | 1 | 1 |
| TANKAGE (ET LH2) | 2104 | $12.1 | $2.4 | $14.5 | 1 | 1 | 1 |
| THERMAL (APOLLO LM) | 456 | $7.5 | $0.2 | $7.8 | 1 | 1 | 1 |
| AVIONICS (APOLLO LM) | 165 | $328.3 | $30.1 | $358.4 | 1 | 1 | 1 |
| ELECTRICAL POWER (LM) | 891 | $307.3 | $8.9 | $316.2 | 1 | 1 | 1 |
| RCS (LM) | 72 | $61.9 | $5.5 | $67.5 | 1 | 1 | 1 |
| PROPULSION | 1338 | $318.7 | $23.8 | $342.5 | | | |
| ENGINES (RL-10A DERIVATIVE) (2 ENGINES) | 1122 | $0.0 | $7.5 | $7.5 | | | |
| PROPULSION LESS ENGINES (APOLLO LM) | 216 | $318.7 | $16.4 | $335.0 | 1 | 1 | 1 |
| **SYSTEM INTEGRATION SUBTOTAL** | | **$1,274.7** | **$22.3** | **$1,297.0** | | | |
| IACO | | $17.2 | $5.4 | $22.5 | 0.52114 | | 0.64022 |
| STO | | $28.6 | N/A | $28.6 | 0.47536 | | |
| GSE | | | | | | | |
| Tooling | | $305.0 | N/A | $305.0 | 0.83422 | | |
| M/E GSE | | $457.5 | N/A | $457.5 | 0.83422 | | |
| SE&I | | $362.7 | $11.4 | $374.2 | 0.97629 | | 0.79217 |
| PM | | $103.8 | $5.5 | $109.3 | 0.62017 | | 0.59773 |
| LOOS | | $0.0 | N/A | $0.0 | 0.84146 | | |
| **Contingency** | | **$744.6** | **$33.4** | **$778.0** | | | |
| **Program Support** | | **$484.0** | **$21.7** | **$505.7** | | | |
| **Fee** | | **$371.1** | **$16.6** | **$387.7** | | | |

# Human Mars Mission
## Space Transportation: Reaction and Control System Technology

The focus of the technology effort was an in-space Reaction and Control System (RCS). The in-space technology would be applicable to all Human Mars Mission (HMM) Architectures.

The accompanying chart gives the key factors associated with a desired RCS to accommodate a variety of space transfer vehicles. The desired end product of the RCS technology would be; (1) the development of an efficient RCS for space transfer vehicles, (2) RCS designed to use main engine propulsion system propellant, and (3) an RCS that is cost effective.

Listed also are the benefits, key technology areas, key decision criteria, and current status of the desired RCS.

# Human Mars Mission
## Space Transportation: Reaction & Control system (RCS) Technology

- End Product
  - Development of efficient RCS for space transfer vehicles
  - RCS designed to use main propulsion system propellant
  - Cost effective RCS

- Benefits
  - Enhance effective use of propellant
  - Reduce initial mass in low-Earth orbit
  - Increase packaging efficiency

- Key Technology Areas
  - High Reliability and Versatility
  - Cryogenic RCS (pump feed/pressure feed system)
  - Thruster performance and indurance
  - Propulsion system lay-out to assure packaging efficiency

- Key Decision Criteria
  - Affordability, and Reliability
  - Capable to accommodate a variety of Space Transfer Vehicles (TMI stage, Mars Descent and Ascent stages, TEI stage, etc)
  - Environmental issues (toxicity, contamination)

- Current Status
  - Shuttle system up-grades
  - Design future ground test where RCS would use main propulsion system propellant

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