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OTV Propulsion Issues

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16. Abstract

A 2-day conference was held at the NASA Lewis Research Center to focus on the issues for future orbital transfer vehicles. The statistical technology needs of aero-assist maneuvering, propulsion, and usage of cryogenic fluids were presented. Industry panels discussed the servicing of reusable space based vehicles and propulsion-vehicle integration. This publication is a compilation of presentations and panel discussions.

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FOREWORD

Enhanced capabilities for Orbital Transfer Vehicles (OTV) will be needed in the mid to late 1990's to meet expanding payload requirements for transporting materials and possibily men to high Earth orbit. It is anticipated that the new OTV will embody significant departures in current design and operational philosophy for upper stages. A 2-day conference at the NASA Lewis Research Center focused on the issues for future OTV.

The status and technology needs of aeroassist maneuvering, propulsion, and cryogenic fluid usage were presented. Industry panels discussed the servicing of reusable space-based vehicles and propulsion-vehicle integration.

This publication is a compilation of the materials from the presentations and panel discussions.

Larry P. Cooper
Conference Chairman

CONTENTS

SESSION IV - PROPELLANT MANAGEMENT
Chairman: J. Aydelott, NASA Lewis Research Center

N85-16990

NASA OAST PERSPECTIVE

Frank Stephenson
NASA Headquarters

An advanced OTV is one of a number of advanced STS vehicles that the NASA OAST Space Systems Division Transportation Systems Office has identified as candidates for future vehicle development. Vehicle requirements as well as technology needs and need dates have been established and technology programs initiated to support those potential developments in a timely manner.

It is assumed that the advanced OTV will be space based and fully reusable for low cost operations, will use aeroassist for return to low-Earth-orbit, and will evolve to a man-rated system. The propulsion system will need to maintain high performance over a wide thrust range for mission flexibility, ranging from the transfer of large, acceleration limited structures from LEO to GEO, to demanding high reliability round-trip manned missions. Technology advances are needed in propulsion, aerobraking, low-gravity cryogenic fluid management, and in environmentally compatible, low-loss cryogenic tankage. In addition, diagnostic instrumentation for monitoring the health of on-board components and systems, and automated check-out capability will enhance low-cost space based OTV operations.

The technology programs currently in place within OAST will provide the technology base in time to support a mid-1990's OTV IOC date, provided proposed FY 86 augmentations in advanced propulsion and in aerobraking technology, including a flight experiment, are approved, and if a focused technology program in light-weight, low-loss cryogenic tankage is initiated in the near future.

SPACE TRANSPORTATION SYSTEM SCENARIO

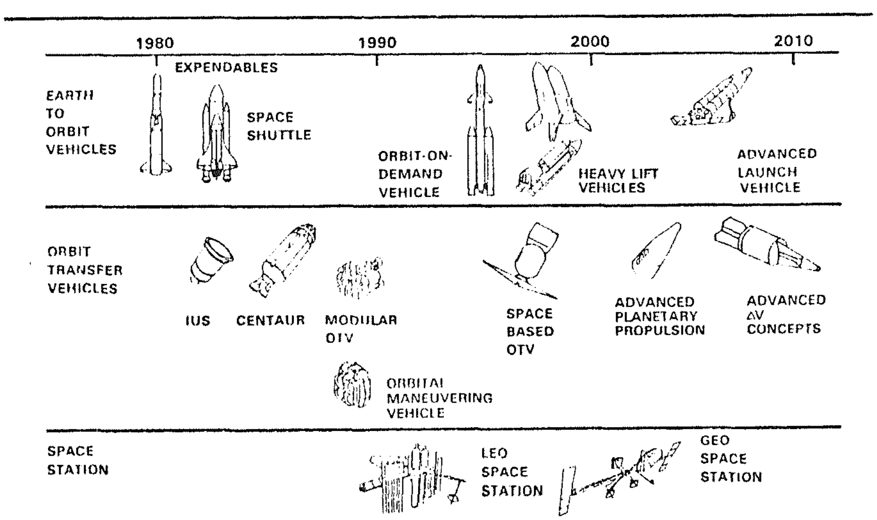


Figure 1

INTEGRATED SPACE TRANSPORTATION

1990's SCENARIO

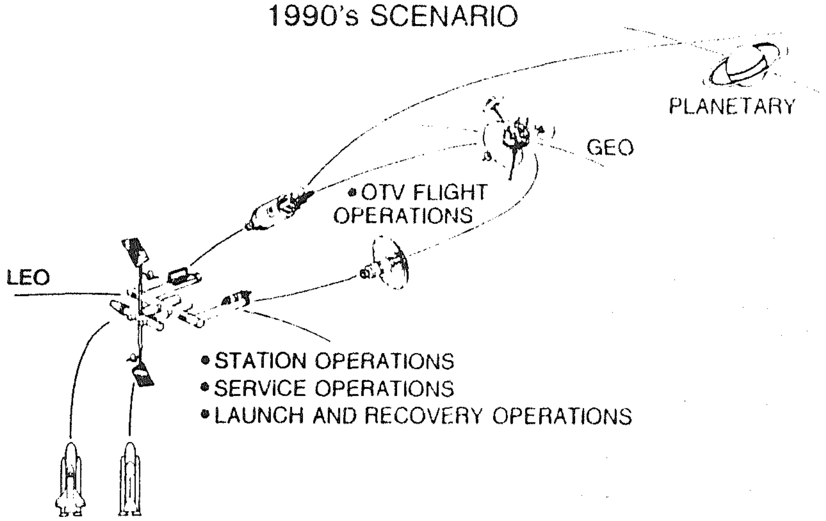


Figure 2

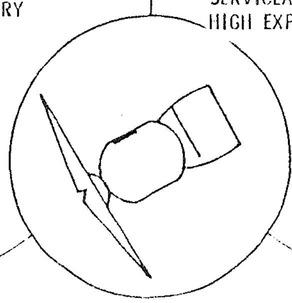
SPACE BASED ORBIT TRANSFER VEHICLE

VEHICLE CHARACTERISTICS

- PAYLOAD 20,000 LBS TO GEO
- LOW COST PAYLOAD DELIVERY
- REUSABLE/ SPACE-BASED USING AEROASSIST
- THROTTLEABLE PROPULSION FOR LARGE STRUCTURE TRANSFER
- TRANSPORTATION SYSTEM AND SPACE STATION COMPATIBLE
- EVOLVE TO MANNED RATING

TECHNOLOGY REQUIREMENTS

- PROPULSION (REUSABLE, SPACE SERVICEABLE, HIGH CHAMBER PRESSURE, HIGH EXPANSION RATIO NOZZLE)
- AEROBRAKE (CONCEPT, STRUCTURE, TPS, NAVIGATION, AERO/ AEROTHERMODYNAMICS, ENVIRONMENT DEFINITION)
- SPACE BASING (HEALTH STATUSING, MAINTENANCE)
- CYROGENIC PROPELLANT MANAGEMENT AND TRANSFER
- LIGHTWEIGHT, LOW-LOSS CRYOGENIC TANKAGE



RATIONALE

ORBITAL TRANSFER CAPABILITY WHICH UTILIZES SPACE STATION FOR LOW COST IN-SPACE OPERATIONS

IOC

MID 1990's

NASA

OAST NASA HQ RS84 305 (1) 11 7 83

Figure 3

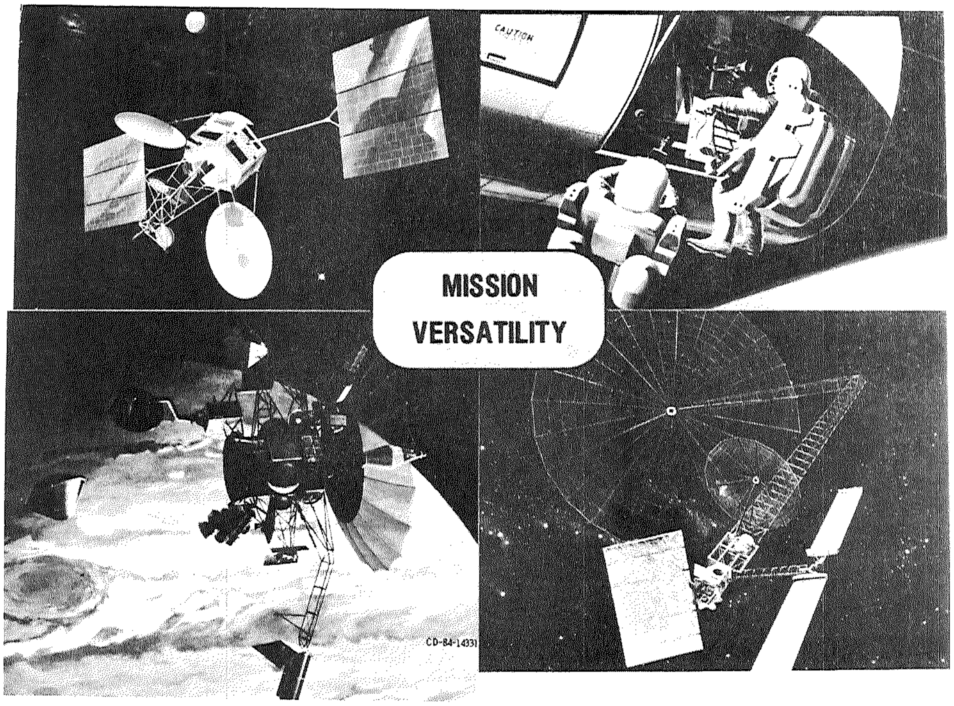


Figure 4

ENGINE PERFORMANCE

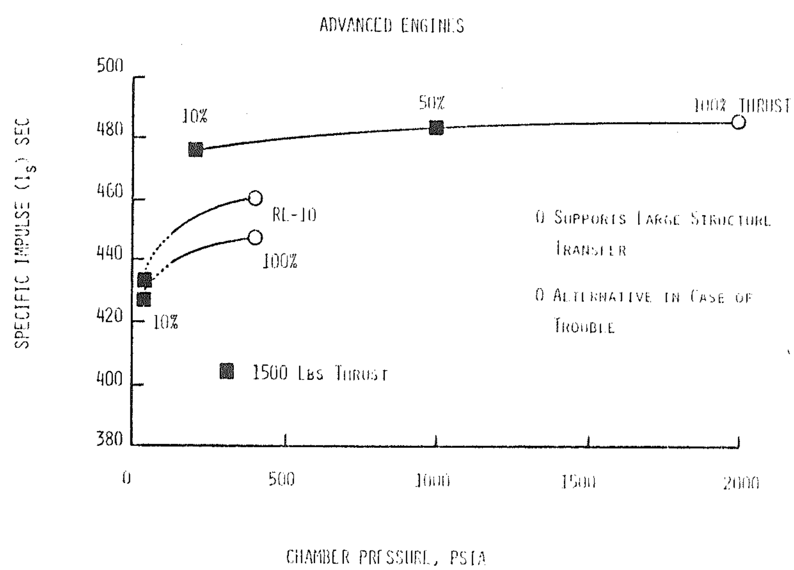


Figure 5

EFFECT ON SPECIFIC IMPULSE ON OTV
PROPELLANT REQUIREMENTS

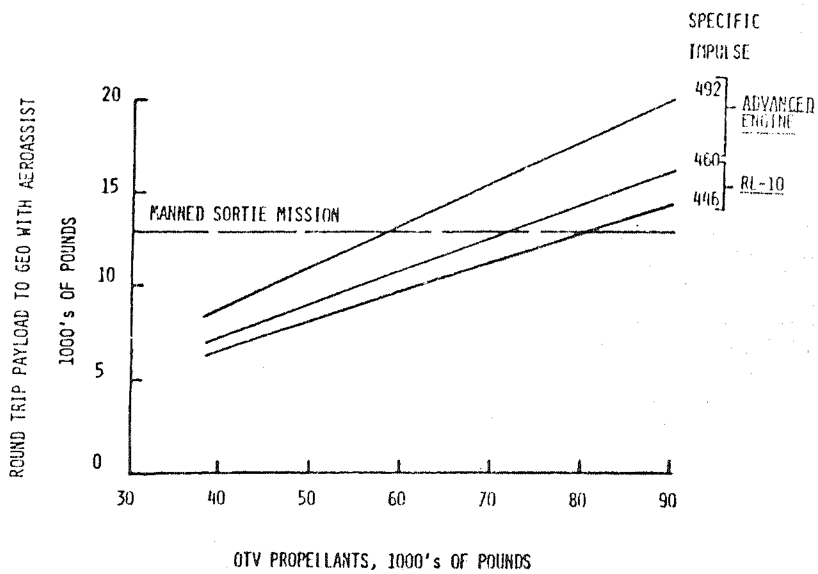


Figure 6

COST SAVINGS FROM ADVANCED ENGINES

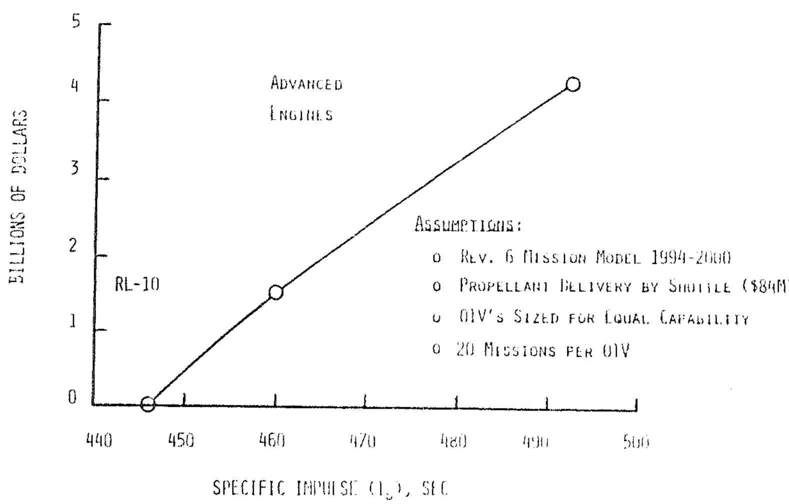


Figure 7

AERO ASSIST IMPROVES OTV

RETURN PERFORMANCE

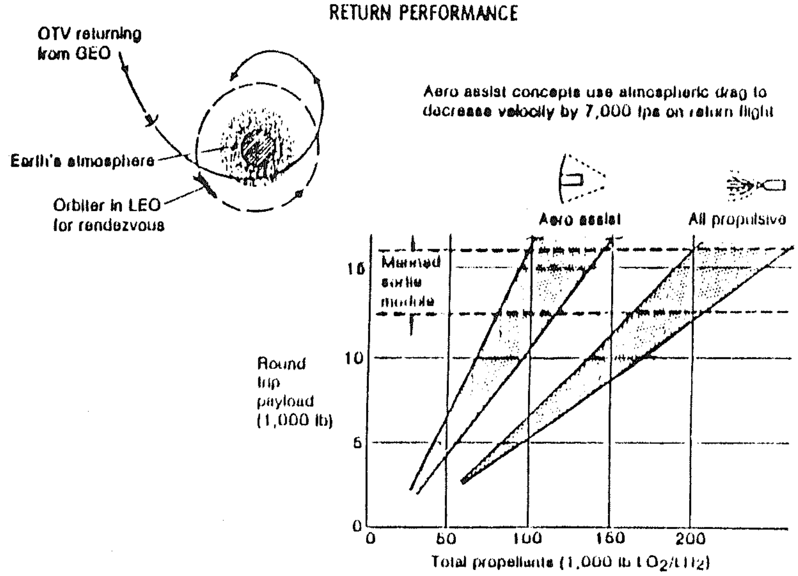


Figure 8

SHUTTLE UPPER STAGE PERFORMANCE

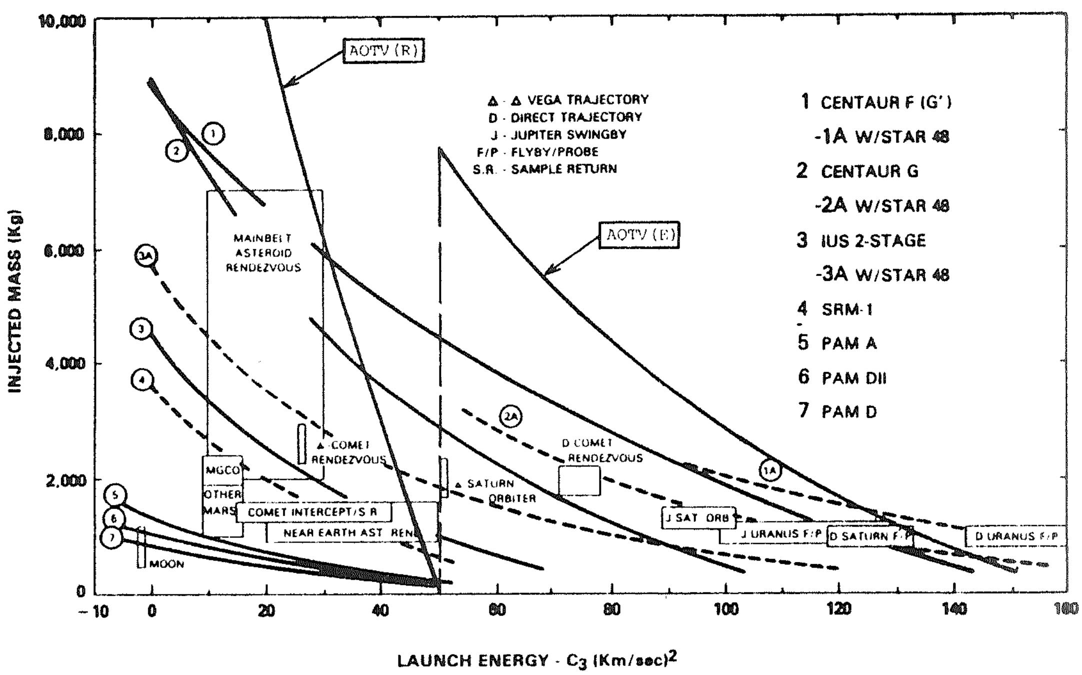


Figure 9

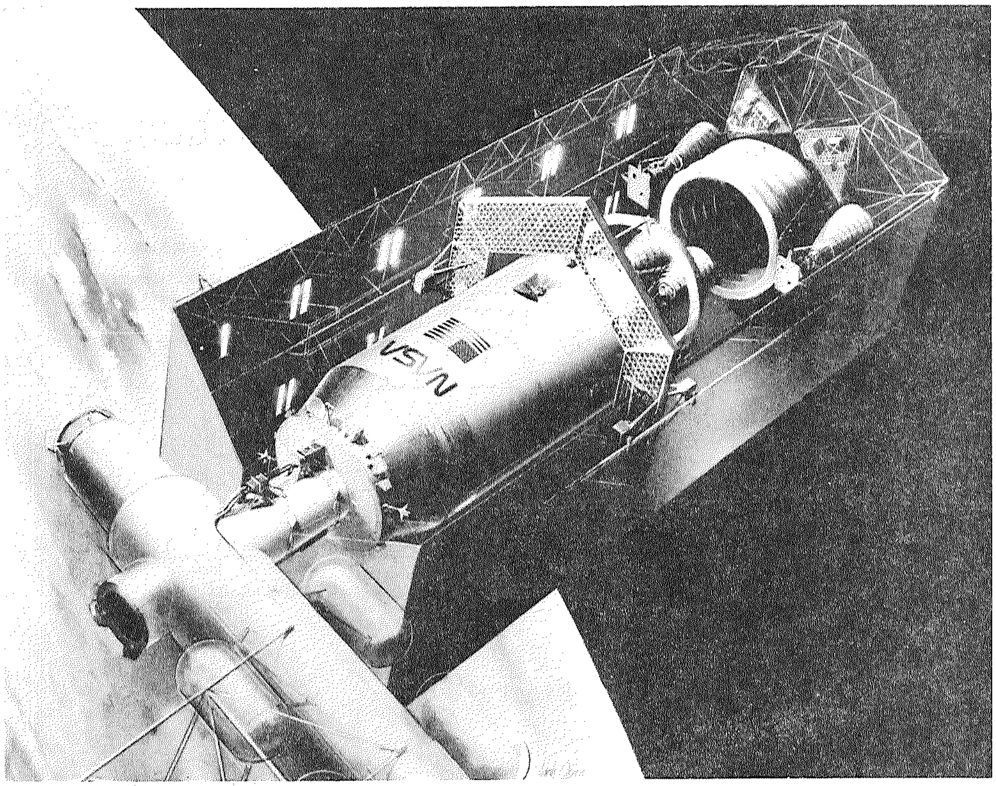


Figure 10

DIAGNOSTICS FOR MAINTAINABILITY APPROACH

ACHIEVED BY USING A BETWEEN FLIGHT AND/OR IN FLIGHT CONDITION MONITORING SYSTEM CONSISTING OF STATE OF THE ART AND/OR NOVEL AUTOMATED DETECTION TECHNOLOGIES AND TAILORED DATA PROCESSING AND COMPUTERS

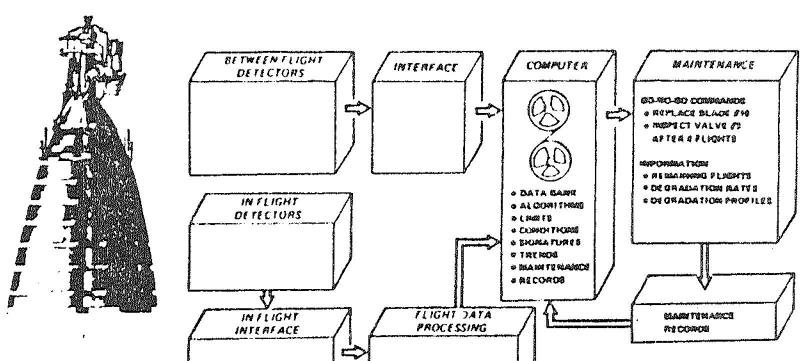


Figure 11

SPACE BASING

REQUIRES • • •

0 IN-SPACE REFUELING

MUST KNOW - PROPELLANT QUALITY

PROPELLANT QUANTITY

STATUS OF ACQUISITION DEVICES

0 SPACE COMPATIBLE TANKAGE

ENVIRONMENT/DEBRIS TOLERANT

AEROBRAKE COMPATIBLE

LOW PROPELLANT LOSS

Figure 12

OTV TECHNOLOGIES

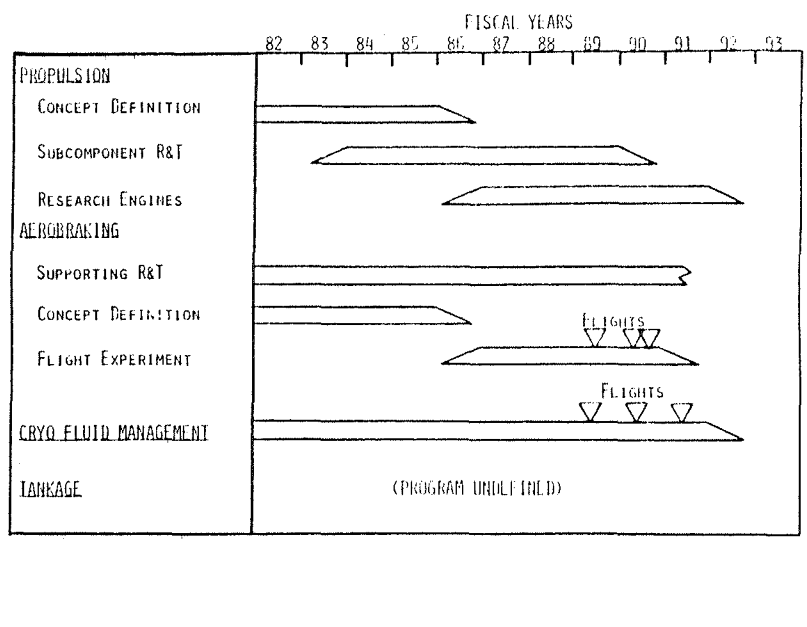


Figure 13

NASA OSF PERSPECTIVE

L. Edwards
NASA Headquarters

No text available at time of printing.

WHAT'S AN OTV

- A HIGH-PERFORMANCE UPPER STAGE FOR GENERAL USE IN THE 1990s.
  - WAS SHUTTLE-LAUNCHED
  - NOW SPACE-BASED (MAINLY? EXCLUSIVELY ?)
  - MAINLY LEO TO GEO
  - LOW COST (REUSABLE)
  - CONFIGURATION TBD

Figure 1

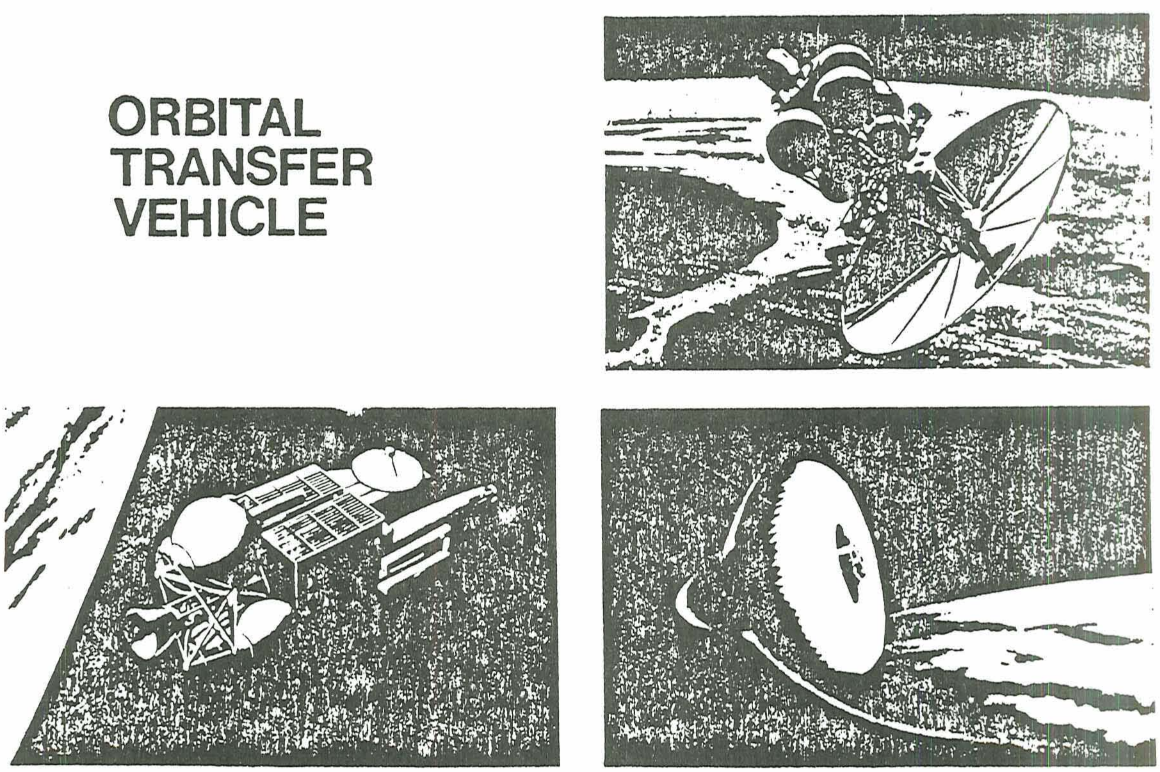


Figure 2

ORBITAL TRANSFER VEHICLE (OTV)

IN CARGO BAY | BEHIND ET

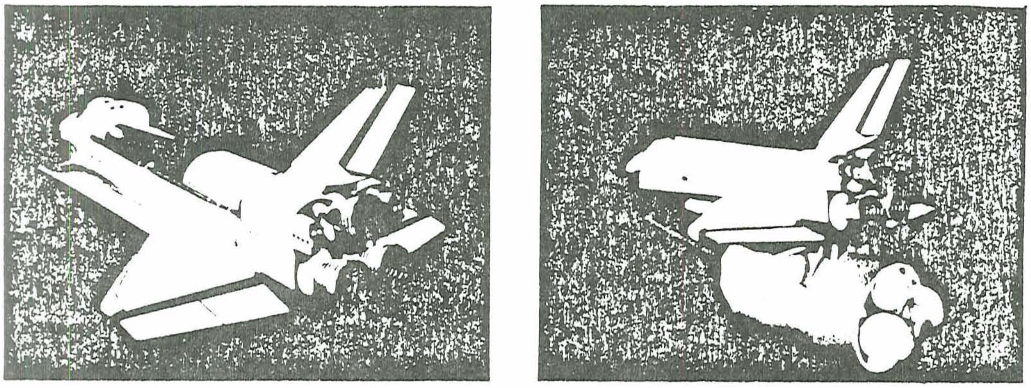
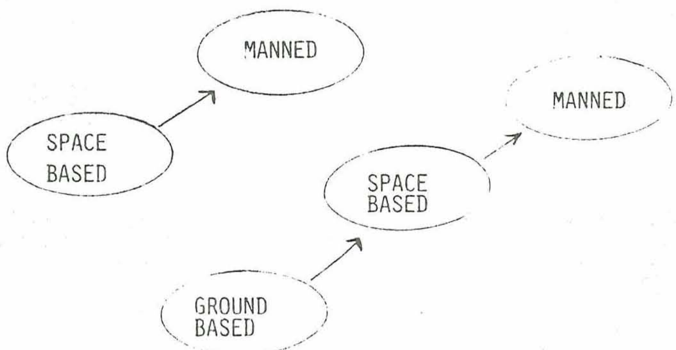


Figure 3

NASA HQ MT82-1222(3)
6-16-82

UPCOMING OTV STUDIES

- 2 CONTRACTORS, $1M EACH, 15 MONTHS
- BEST CONFIGURATION IN TWO SCENARIOS:



- IMPACT OF OTV ON SPACE STATION
  - HANGAR/MAINTENANCE
  - PROPELLANTS
  - CREW REQUIREMENTS

Figure 4

UTILIZATION OF SPACE—BASED OTV

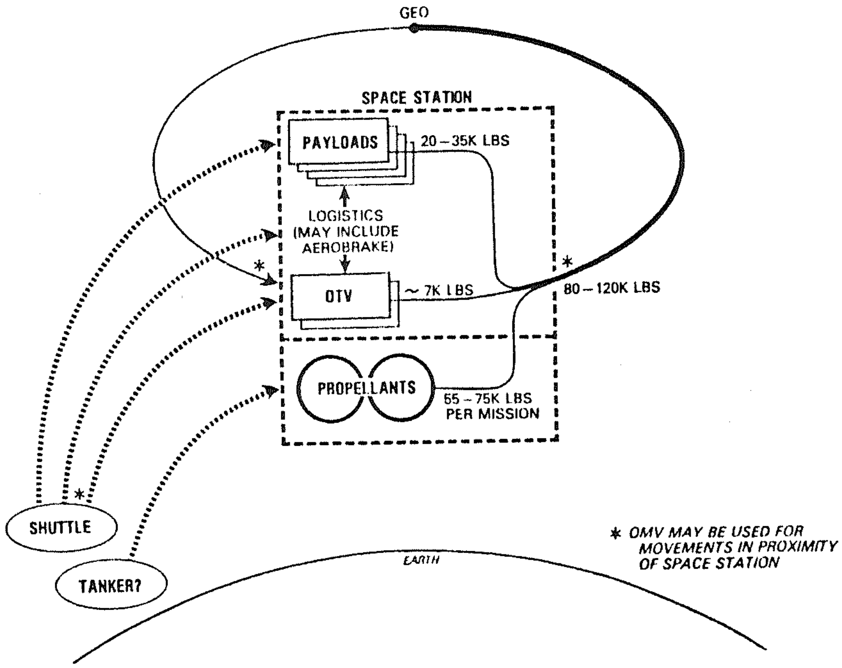


Figure 5

RELATED OTV STUDIES

AEROASSIST	OAST	MSFC ET AL	BOEING
CRYO STORAGE & HANDLING	OAST	LERC	MARTIN
SBOTV DEVELOPMENT MISSION	SSTF	MSFC	CONVAIR
SCAVENGING	OSF	JSC MSFC	ROCKWELL MARTIN
AFT CARGO CARRIER	OSF	MSFC	MARTIN
GN&C	OSF	JSC	DRAPER
MANNED OTV CAPSULE	OSF	JSC	TBD
GROUND/SPACE SUPPORT	OSF	KSC	TBD
RL10 ENGINE	OSF	LERC	PWA
ADVANCED ENGINE	OAST	LERC	3 CONTRACTS

Figure 6

CRYO vs STORABLE FOR OTV

- UPCOMING CONTRACTS WILL INCLUDE TRADEOFFS
- CRYO WILL BE NEEDED EVENTUALLY
- CRYO TAKES ADVANTAGE OF SCAVENGING FROM ET
- ONLY CRYO PERMITS SINGLE REUSABLE STAGE FOR SPACE-BASED MISSIONS
- INTERESTING POSSIBILITY FOR MANNED GEO MISSIONS:
  - CRYO/EXPENDABLE GOING TO GEO
  - STORABLE/REUSABLE FOR LOITER AND RETURN

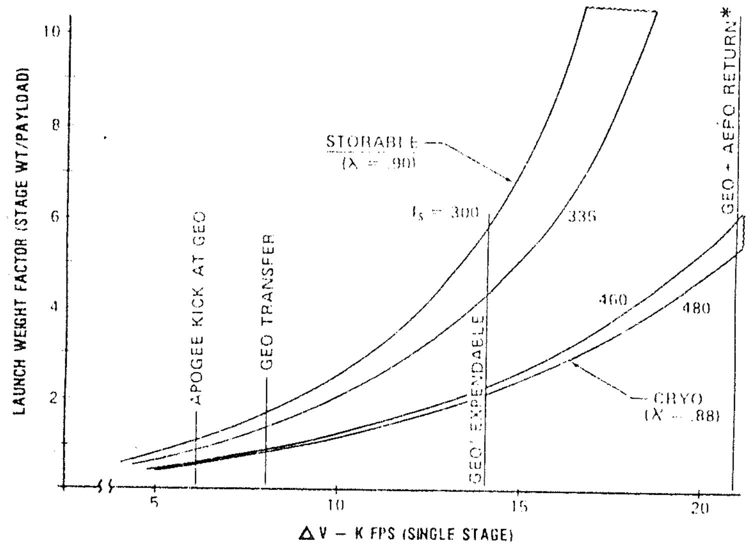
Figure 7

KEY REQUIREMENTS FOR OTV ENGINE

- SUITABLE FOR SPACE BASING & REUSE
  - LONG LIFE, MANY STARTS
  - EASY CHECKOUT
  - EASY SERVICING/MAINTENANCE/REPLACEMENT UNLESS THESE CAN BE SHOWN TO BE UNNECESSARY
- COMPATIBLE WITH AEROBRAKE
- I_S AT LEAST 460 SECONDS
- THRUST 15-20K POUNDS (MR 6:1)
- ALTERNATE THRUST $\sim$1500 POUNDS (NO KITS)
  - VARIABLE THRUST MAY BE USEFUL BUT NOT MANDATORY
- STOWED LENGTH NOT OVER 55 INCHES
- OCCASIONAL MANNED FLIGHTS
  - MAY REQUIRE DUAL ENGINES

Figure 8

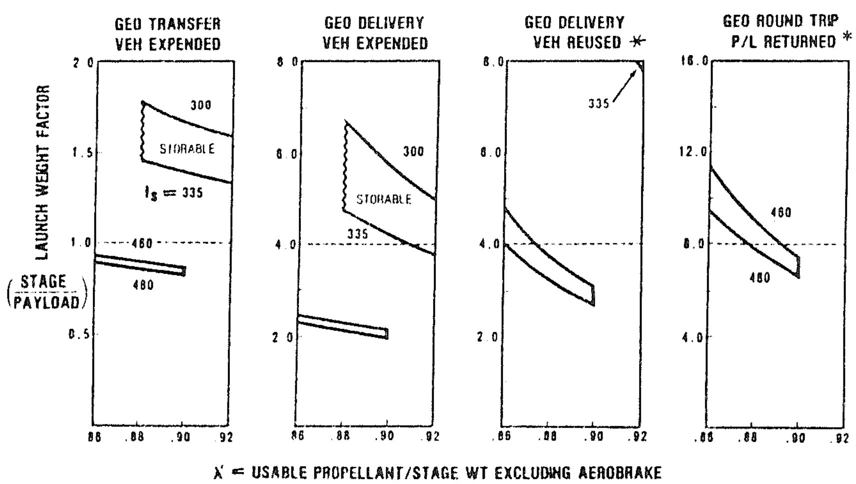
NOMINAL PERFORMANCE COMPARISON
STORABLE AND CRYO



* AEROBRAKE WT. OMITTED IN THIS PLOT

Figure 9

NORMALIZED PERFORMANCE
STORABLES & CRYO FOR VARIOUS MISSIONS, I_s, λ' SINGLE STAGE



* INCLUDES AEROBRAKE AT 15% OF RE ENTERING WT; 1000 FPS SAVED FOR FINAL MANEUVER + MARGIN

Figure 10

SPACE STATION SCENARIOS
ARBITRARY 20K PAYLOAD TO GEO

PRELIMINARY
L.E. 3-2-84

	SCENARIO	1st STAGE PROP.	1st STAGE BOW	2nd STAGE PROP.	2nd STAGE BOW	MUST REPLACE
		$\lambda' = .90$		$\lambda' = .85$		
	S1: ONE STAGE EXPENDABLE	92/72	10/8	-	-	PROP. + 1 VEHICLE 120-80 K LBS
STORABLES $I_s = 312/342$	S2: ONE STAGE REUSABLE*	-/219	-/24	-	-	PROPELLANTS ∞ - 219 K
	S3: TWO STAGES EXPENDABLE	60/48	6.7/5.3	19/17	3/4/3.0	PROP. + 2 VEHICLES 89-73 K
	S4: TWO STAGES; ONE STAGE REUSABLE*	65/51	7/6	19/17	3.4/3.0	PROP. + 1 VEHICLE 87-71 K
		$\lambda' = .88$				
CRYO $I_s = 460$	C1: ONE STAGE EXPENDABLE	40	5.5	-	-	PROP. + 1 VEHICLE 46K + BOILOFF
	C2: ONE STAGE, REUSABLE*	65	10.5	-	-	PROPELLANTS 65K + BOILOFF

* INCLUDES AEROBRAKE @ 15% OF REENTRY WEIGHT.

Figure 11

RESUPPLY REQUIREMENT FOR SPACE STATION SCENARIOS
ARBITRARY 20K PAYLOAD TO GEO

PRELIMINARY
L.E. 3/2/84

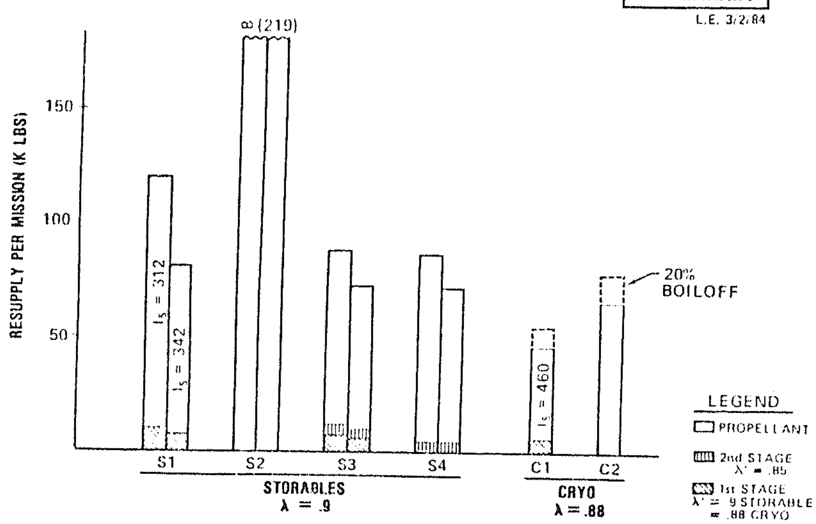


Figure 12

SHUTTLE/CENTAUR PROJECT PERSPECTIVE

Edwin T. Muckley
National Aeronautics and Space Administration
Lewis Research Center

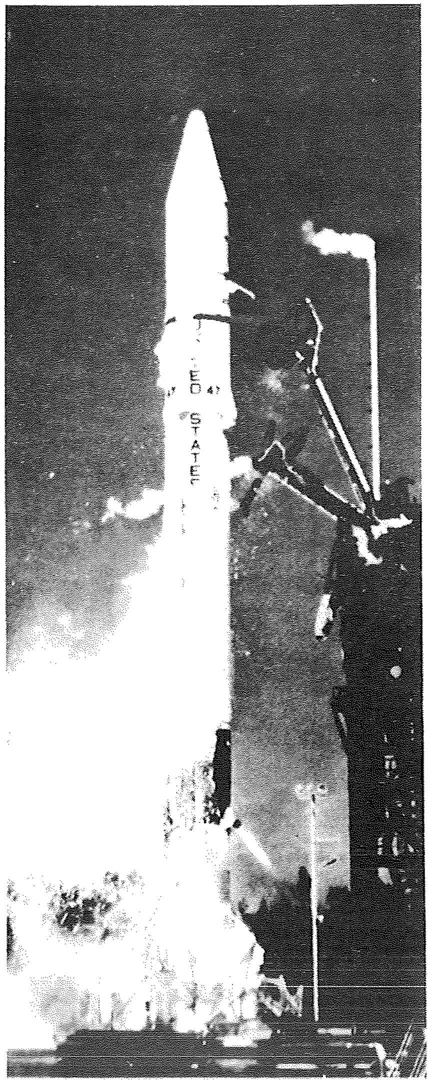
The Shuttle/Centaur vehicle is being developed as an expendable, cryogenic high energy upper stage for use with the National Space Transportation System (NSTS). The stage is expected to meet the demands of a wide range of users including, NASA, the DOD, private industry and the European Space Agency (ESA). The Shuttle/Centaur will be a modification of the highly successful Centaur stage, used extensively with the Atlas and Titan boosters since 1966 to launch planetary, geosynchronous and earth orbital missions for these aforementioned users. This paper describes the design changes required for use with the NSTS. These are primarily related to: 1) tank resizing to take advantage of the orbiter payload bay dimensions; 2) provisions for physically adopting Centaur to the orbiter; and, 3) accommodating safety requirements of the manned NSTS. The paper will also describe the expected performance capabilities of two versions of the Shuttle/Centaur. The initial version, designated G-prime, is the larger of the two, with a length of about 9.1m (30 ft.). This vehicle will be used to launch the Galileo and International Solar Polar Missions (ISPM) to Jupiter in May 1986. The Galileo to be launched for NASA's Jet Propulsion Laboratory, will orbit the planet, observe its satellites, and a probe portion will separate and descend into the Jovian atmosphere. The European Space Agency ISPM spacecraft will use Jupiter's mass to deflect its trajectory out of the ecliptic plane and gather data in the sun polar region.

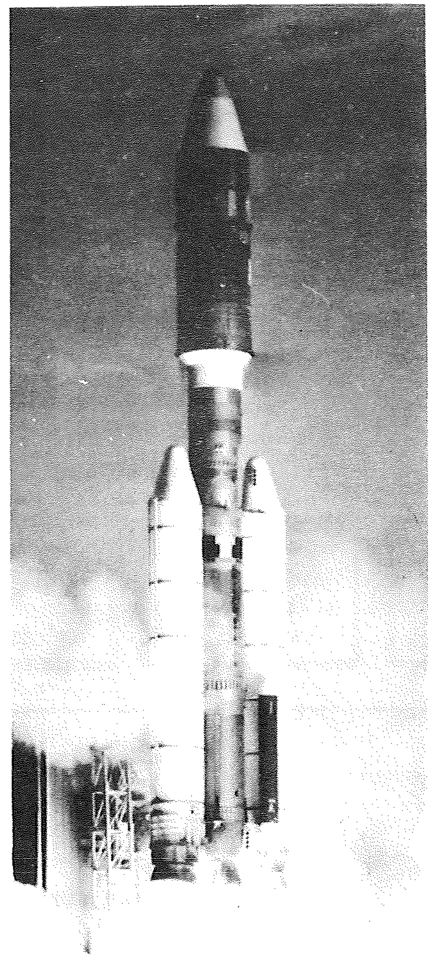
The second version of the Shuttle/Centaur designated the G vehicle, is about 3.0m (ten ft.) shorter than the G-prime. This shorter stage also takes advantage of the orbiter 4.6m (15 ft.) diameter, but maximizes the spacecraft length capability in the payload bay to about 12.2m (40 ft.). It is currently scheduled to launch payloads for the DOD, the NASA Venus Radar Mapper and TDRSS Missions in 1988, and is expected to provide launch services well into the 1990's.

CENTAUR IS A MATURE, FLIGHT PROVEN, HIGH-ENERGY UPPER STAGE

Atlas/Centaur

Payloads	**61**
Test Flight	8
Surveyor	7
ATS	2
OAO	3
Mariner Mars	4
Intelsat IV	8
Intelsat IVA	6
Pioneer F	1
Pioneer G	1
MVM	1
Comstar	4
HEAO A	1
HEAO B	1
HEAO C	1
Fltsatcom	5
Pioneer Venus	2
Intelsat V	6





Titan/Centaur

Payloads	**7**
Test Flight	
Helios A	
Helios B	
Viking A	
Viking B	
Voyager 1	
Voyager 2	

Figure 1

LEWIS RESEARCH CENTER/CENTAUR ORGANIZATION

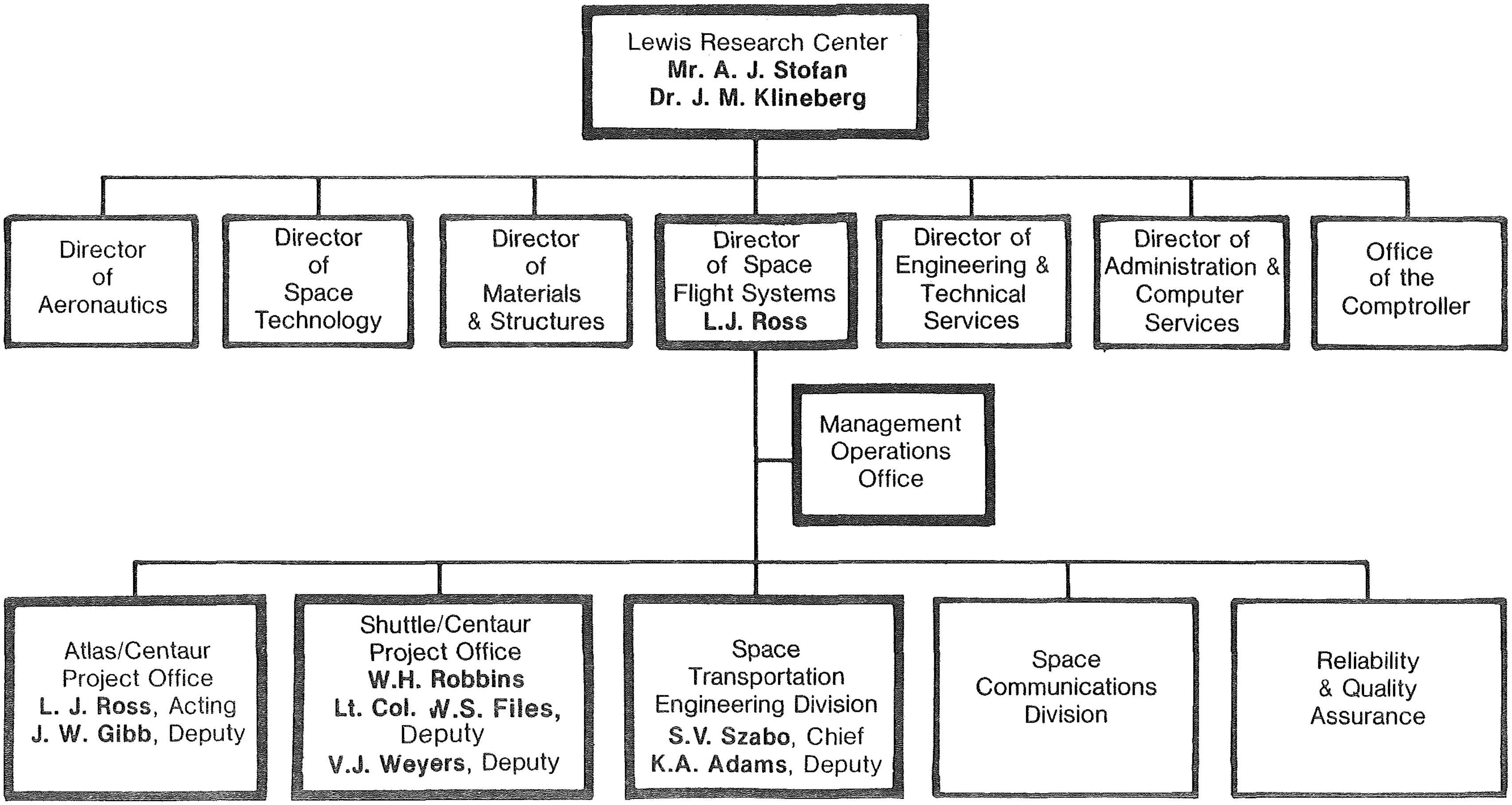


Figure 2

SHUTTLE/CENTAUR IMPLEMENTATION POLICY

- Shuttle/Centaur is a NASA/USAF cooperative program
- Project management has been assigned to a joint NASA/USAF project office at the NASA Lewis Research Center
- Funding is provided by both agencies

Figure 3

SHUTTLE/CENTAUR PROGRAM MANAGEMENT RELATIONSHIPS

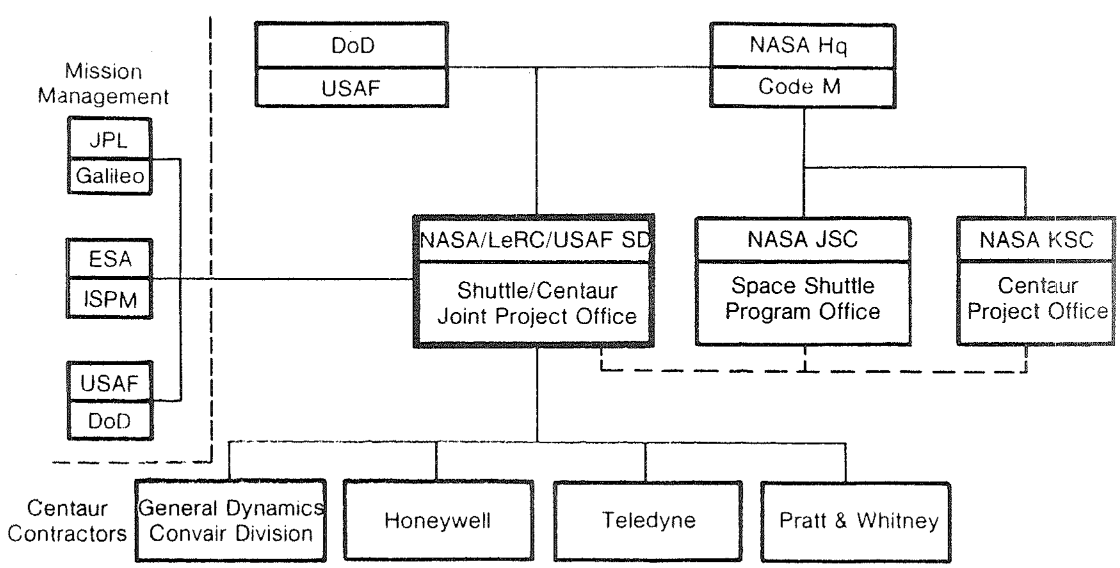


Figure 4

SHUTTLE/CENTAUR REQUIREMENTS

General

- Design & develop a high-energy upper stage for use with Space Transportation System
- Two versions will be developed

USAF

- Performance
  - 10,000 lb to geosynchronous orbit
  - 11,500 lb to 12-hr orbit
- Accommodate a 40-ft payload in orbiter/bay
- Support two USAF missions

NASA

- Performance
  - Meet interplanetary velocity requirements
- Accommodate a 30-ft payload
- Support Galileo & ISPM missions in 1986

Figure 5

CENTAUR INTEGRATED SUPPORT SYSTEM MINIMIZES CHANGES TO SHUTTLE

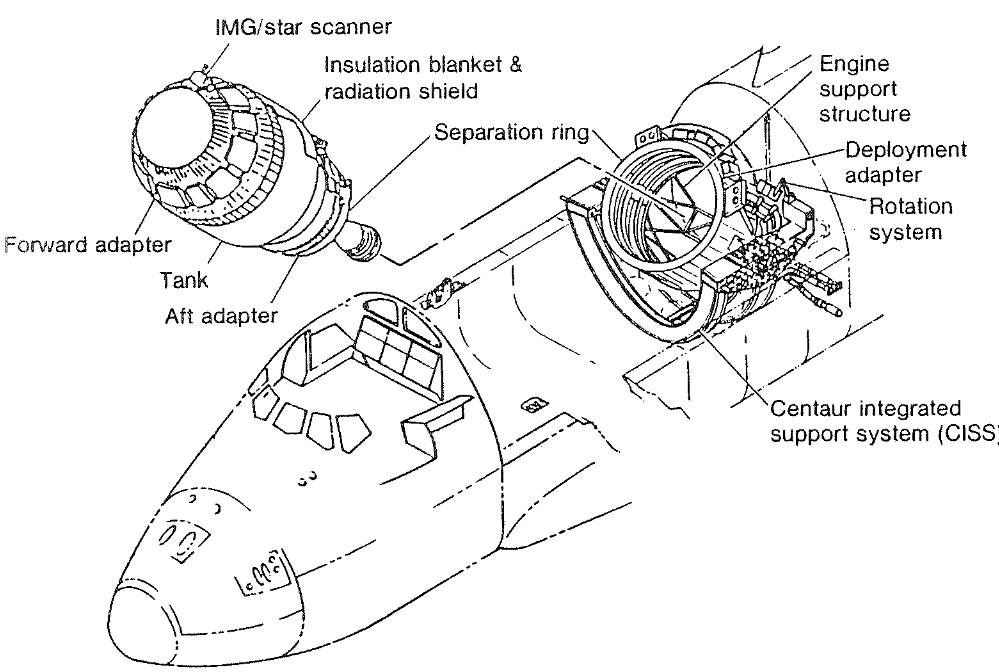


Figure 6

CENTAUR CONFIGURATIONS

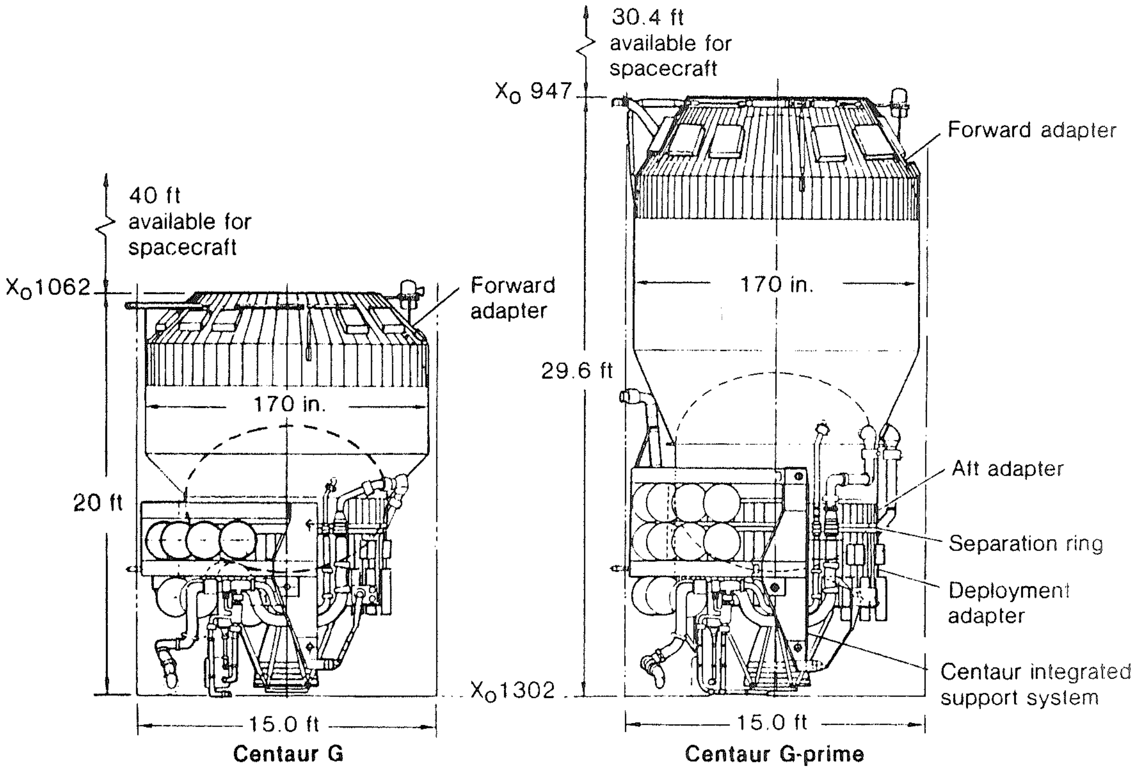


Figure 7

CENTAUR INTEGRATED SUPPORT SYSTEM

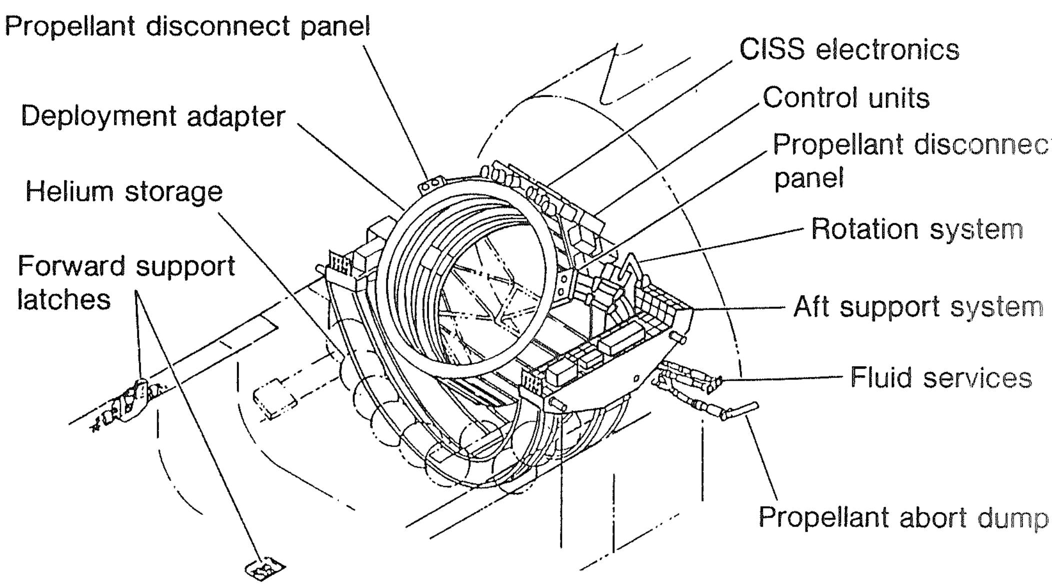
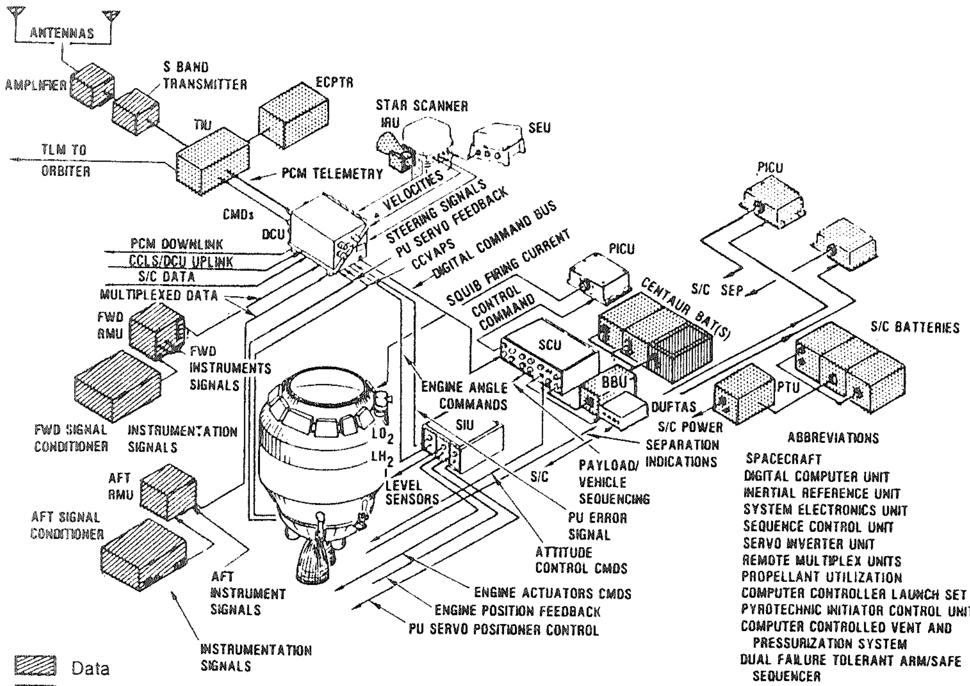


Figure 8

CENTAUR AVIONICS

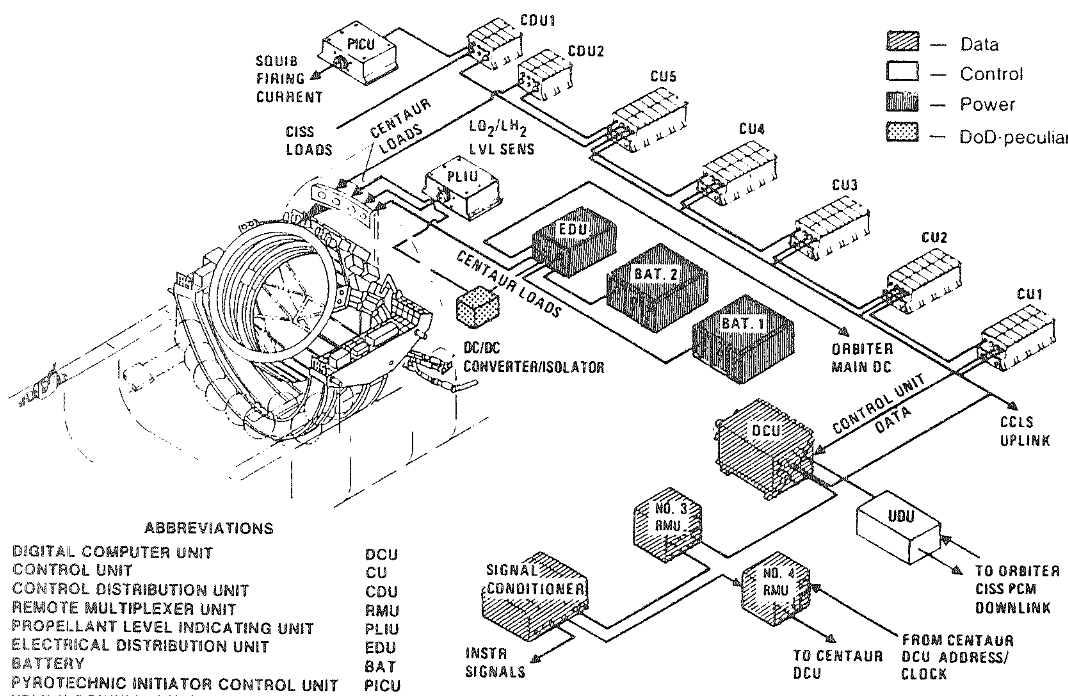


ABBREVIATIONS

SPACECRAFT	S/C
DIGITAL COMPUTER UNIT	DCU
INERTIAL REFERENCE UNIT	IRU
SYSTEM ELECTRONICS UNIT	SEU
SEQUENCE CONTROL UNIT	SCU
SERVO INVERTER UNIT	SIU
REMOTE MULTIPLEX UNITS	RMU
PROPELLANT UTILIZATION	PU
COMPUTER CONTROLLER LAUNCH SET	CCLS
PYROTECHNIC INITIATOR CONTROL UNIT	PICU
COMPUTER CONTROLLED VENT AND PRESSURIZATION SYSTEM	CCVAPS
DUAL FAILURE TOLERANT ARM/SAFE SEQUENCER	DUFTAS
BATTERY	BAT
S/C POWER TRANSFER UNIT	PTU
TELEMETRY INTERFACE UNIT	TIU
ENCRYPTOR	ECPTR
BATTERY BUSSING UNIT	BBU
POWER TRANSFER UNIT	PTU

Figure 9

CISS AVIONICS SYSTEMS

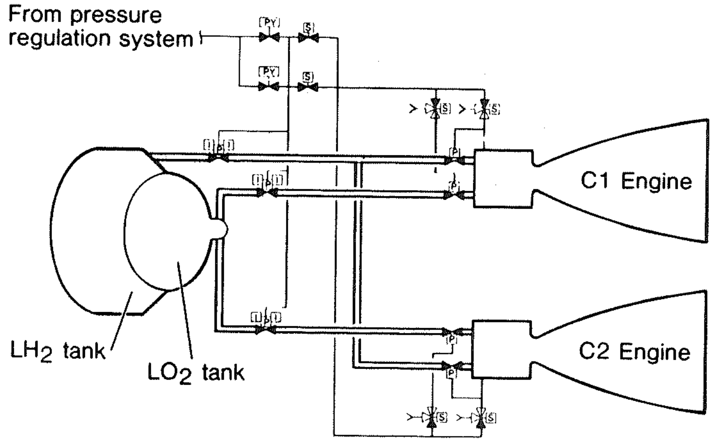


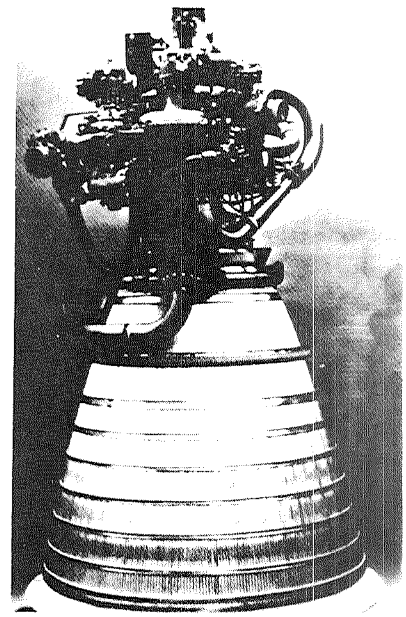
ABBREVIATIONS

DIGITAL COMPUTER UNIT	DCU
CONTROL UNIT	CU
CONTROL DISTRIBUTION UNIT	CDU
REMOTE MULTIPLEXER UNIT	RMU
PROPELLANT LEVEL INDICATING UNIT	PLIU
ELECTRICAL DISTRIBUTION UNIT	EDU
BATTERY	BAT
PYROTECHNIC INITIATOR CONTROL UNIT	PICU
UPLINK DOWNLINK UNIT	UDU

Figure 10

CENTAUR MAIN PROPULSION SYSTEM





	Centaur-G′	Centaur-G
P&W engine	RL10A-3-3A	RL10A-3-3B
Mixture ratio	5:1	6:1
Thrust	16,500 lbf	15,000 lbf
Isp	446.4 sec	440.4 sec

Figure 11

CENTAUR VEHICLE MODIFICATIONS FOR SHUTTLE COMPATIBILITY

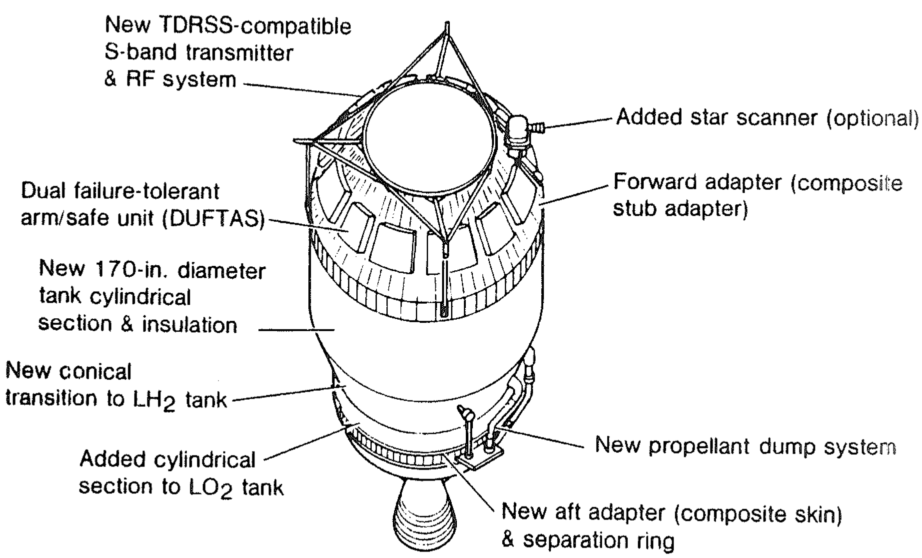


Figure 12

ORBITER MODIFICATIONS
FLUID CONNECT & OUTLET LOCATIONS

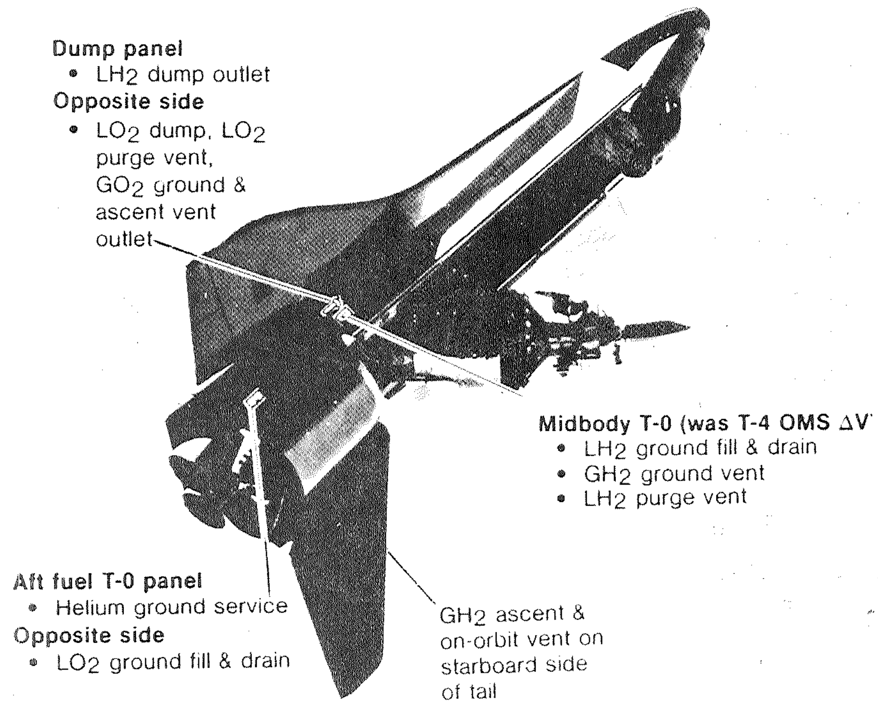


Figure 13

SHUTTLE/CENTAUR LAUNCH OPERATIONS FLOW — ELS

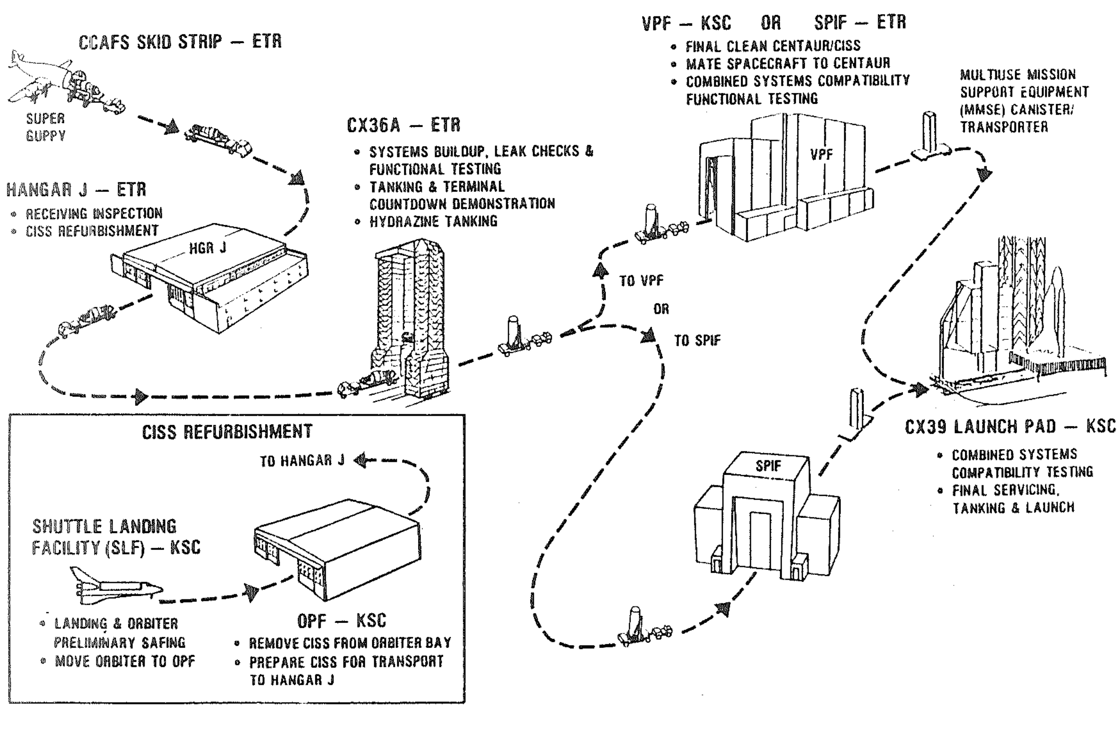


Figure 14

FLIGHT OPERATIONS

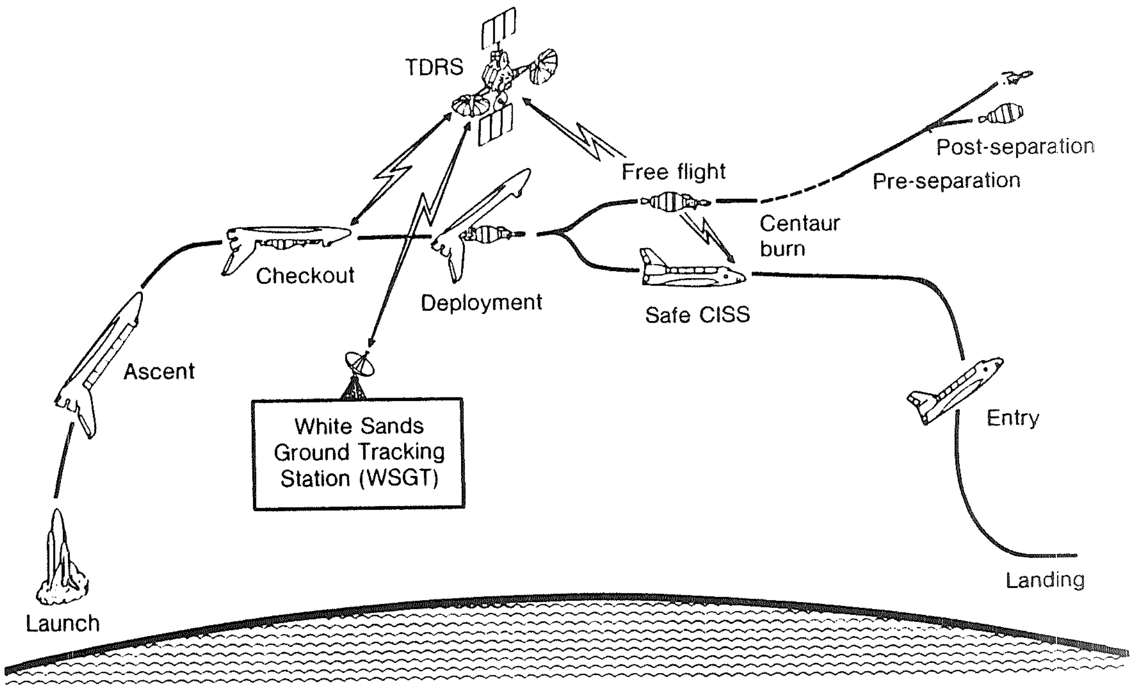


Figure 15

SHUTTLE/CENTAUR INTEGRATED SCHEDULE

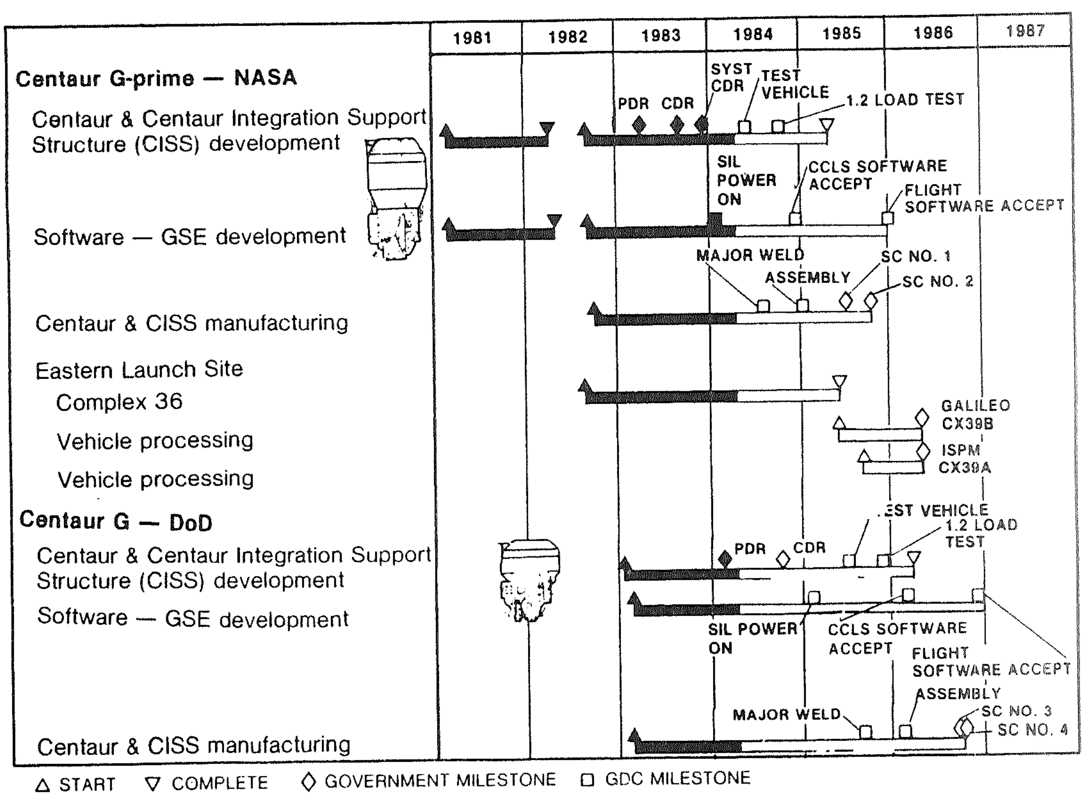


Figure 16

INTERNATIONAL SOLAR POLAR MISSION (ISPM)

- First ever exploration far from ecliptic plane & sun polar region (out of ecliptic)
- Gravitational field of Jupiter used as "sling shot"
- Cooperative program with ESA (European Space Agency)
- Single launch using Shuttle & Centaur
- Weight of spacecraft: 350 kg
- Launch: May 1986

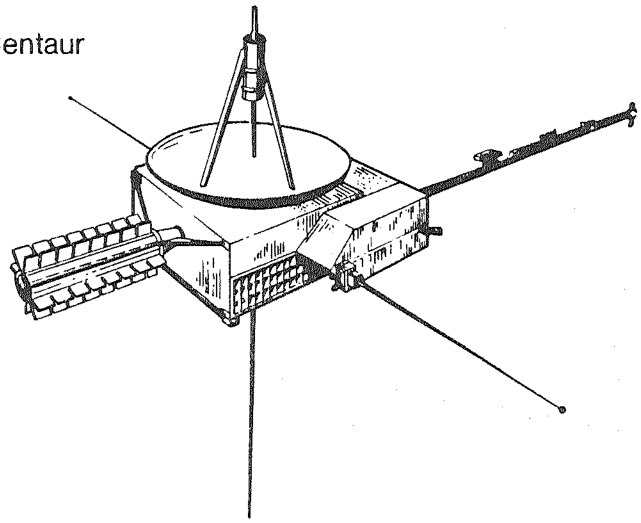


Figure 17

ISPM MISSION

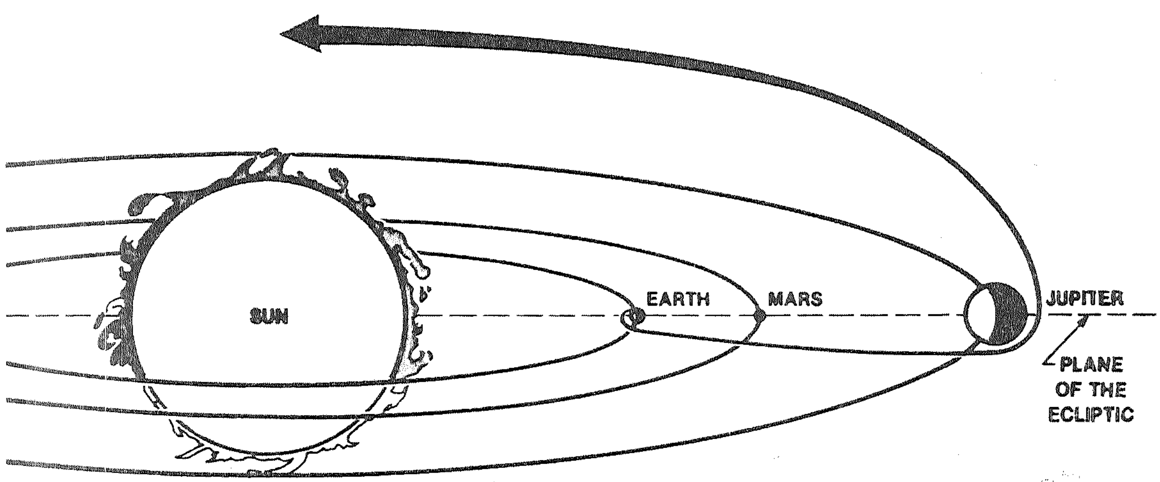


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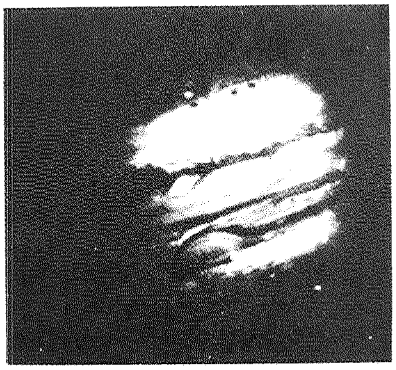
Figure 18

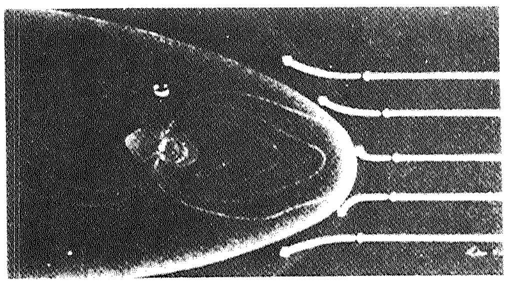
PROJECT GALILEO WILL INVESTIGATE THE . . .



Chemical composition & physical states
of the Jovian satellites

Chemical composition & physical state
of Jupiter's atmosphere

Structure & physical dynamics of the Jovian
magnetosphere



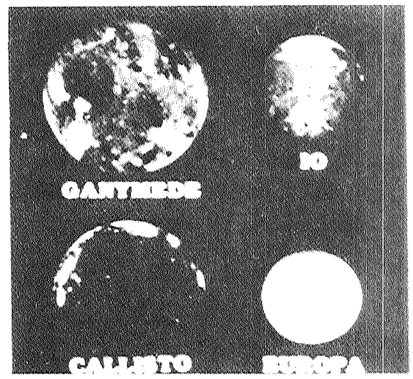


Figure 19

GALILEO

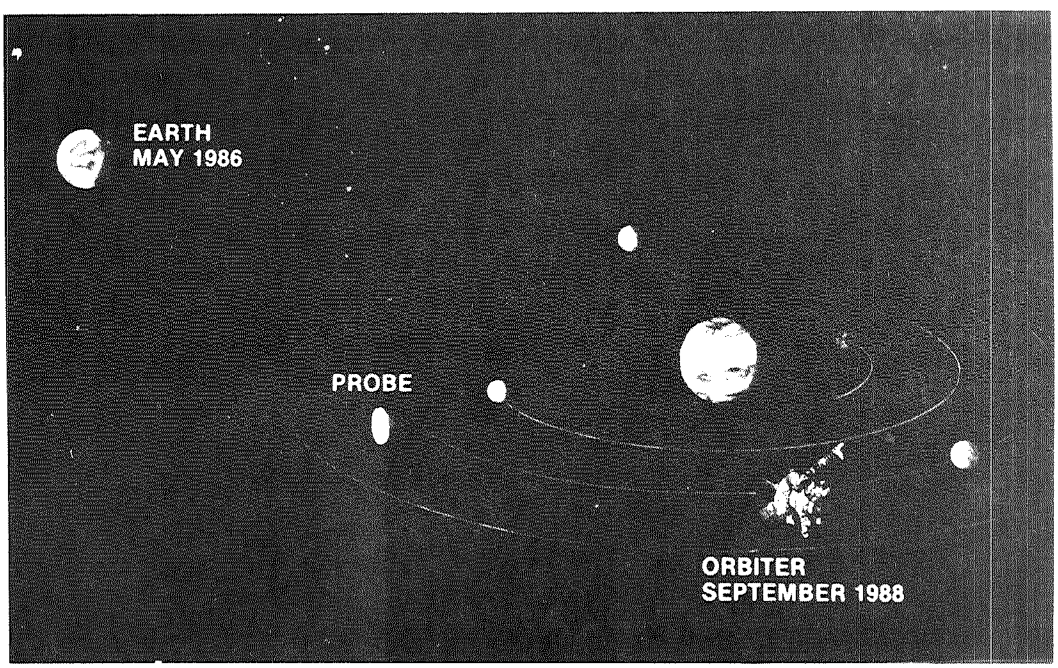


Figure 20

N85-16992 D3

SPACE STATION TASK FORCE PERSPECTIVE

C. Hicks
NASA Headquarters

No text available at time of printing.

PRESENTATION OUTLINE

- PRELIMINARY PROGRAM DESCRIPTION
  - DEFINITIONS
  - FUNCTIONS
  - CAPABILITIES
  - MANAGEMENT APPROACH
  - SCHEDULES
- SPACE STATION SERVICING CAPABILITY
- SPACE STATION – ORBITAL TRANSFER VEHICLE (OTV) PROGRAM INTERFACES

Figure 1

SPACE STATION PLANNING GUIDELINES

MANAGEMENT RELATED

- Three year extensive definition (5-10% of program cost)
- NASA-wide participation
- Development funding in FY 1987
- IOC: early 1990's
- Cost of initial capability: $8.0B
- Extensive user involvement
  - Science and applications
  - Technology
  - DoD
  - Commercial
- International participation

ENGINEERING RELATED

- Continuously habitable
- Shuttle dependent
- Manned and unmanned elements
- Evolutionary
- Maintainable/restorable
- Operationally autonomous
- Customer friendly
- Technology transparent

Figure 2

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SPACE STATION PROGRAM ARCHITECTURE: WHAT IS A SPACE STATION

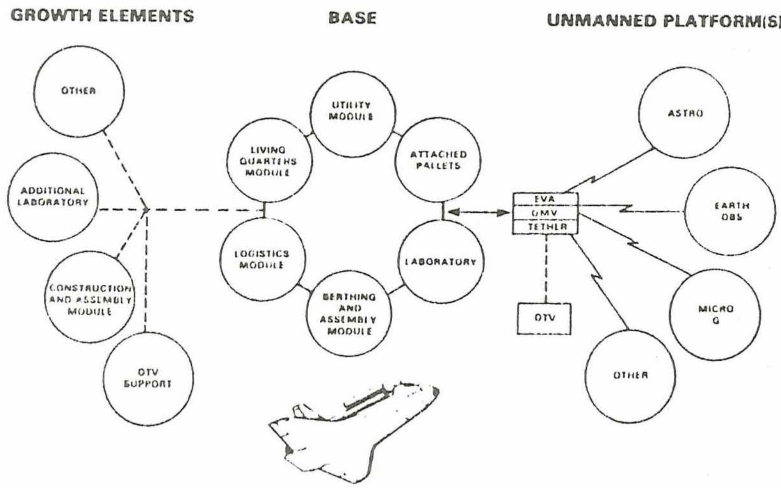


Figure 3

FUNCTIONS OF A SPACE STATION

- On-orbit laboratory
  - — Science and applications
  - — Technology
- Permanent observatory(s)
- Transportation node
- Servicing facility
  - — Free flyers
  - — Platforms
- Communications and data processing node
- Manufacturing facility
- Assembly facility
- Storage depot

A space station is a multi-purpose facility

Figure 4

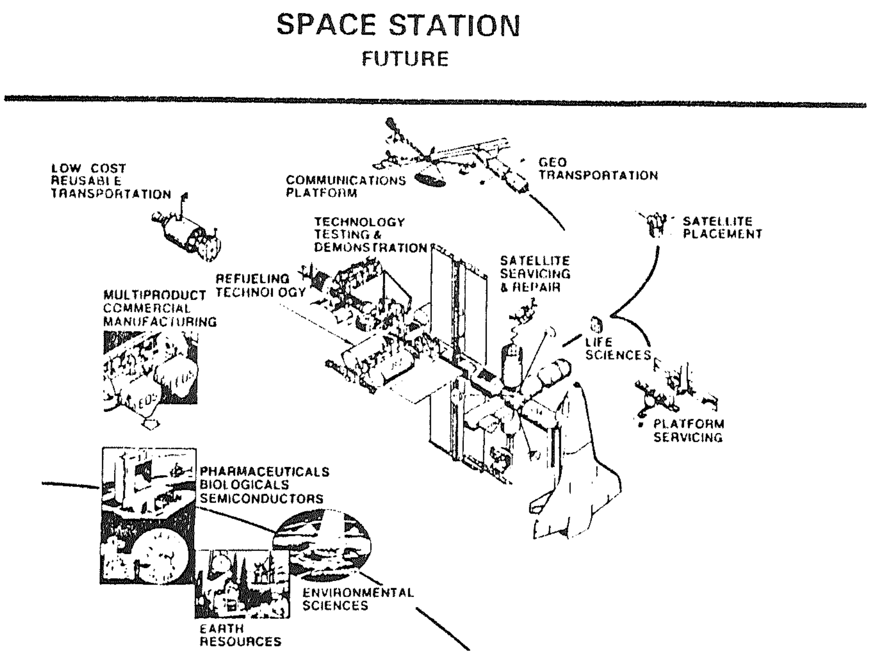


Figure 5

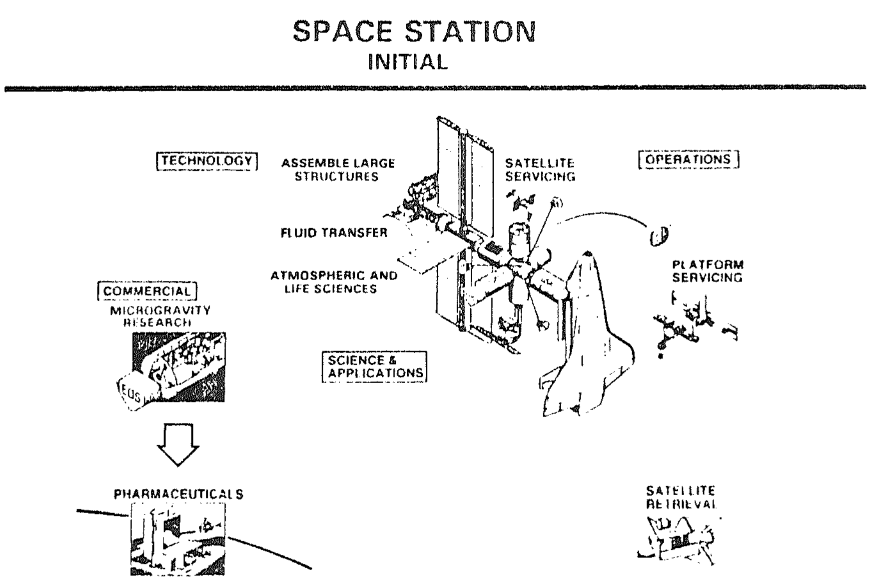


Figure 6

THE RELATIONSHIP BETWEEN THE SPACE STATION PROGRAM AND OTHER PROGRAMS

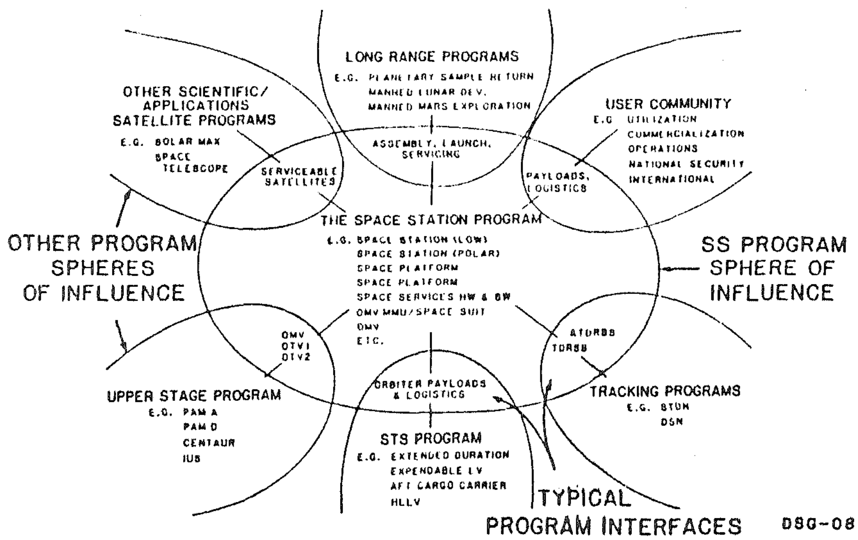


Figure 7

SPACE STATION DEFINITION PRELIMINARY MISSION DATA BASE (1991-2000)

- Initial Data Base
- Derived from Shuttle and ELV Base
- Will Change as Station Capabilities Become Better Understood and Mission Priorities Shift
- Not the List of Mission/Payloads the Station Will Fly in 1991

SCIENCE AND APPLICATIONS
- Astrophysics
- Earth Science and Applications
- Solar System Exploration
- Life Sciences
- Materials Science
- Communications

COMMERCIAL
- Materials Processing In Space
- Earth and Oceans Observations
- Communications

TECHNOLOGY DEVELOPMENT
- Materials and Structures
- Energy Conversion
- Computer Science and Electronics
- Propulsion
- Controls and Human Factors
- Space Station Systems/Operations
- Fluid and Thermal Physics

Figure 8

SCOPE OF INITIAL SPACE STATION

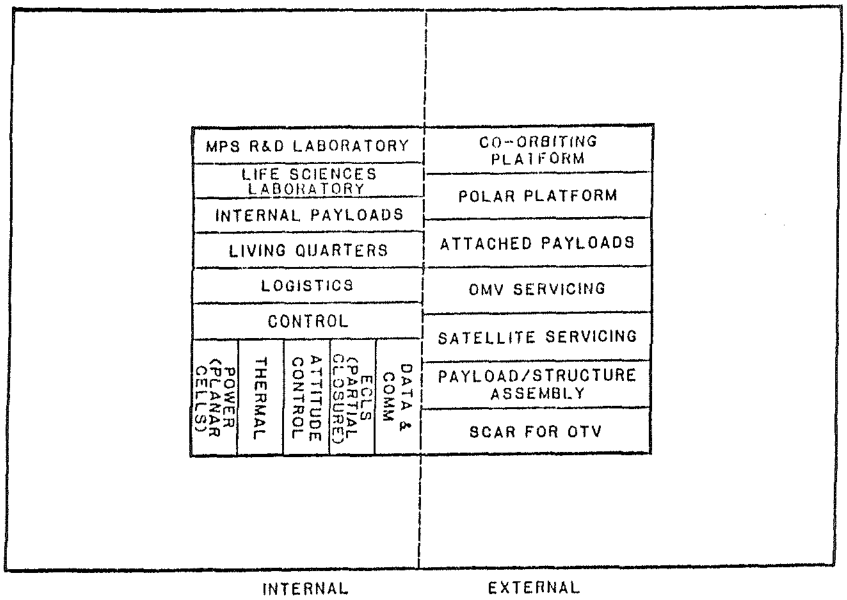


Figure 9

ADDED SCOPE FOR INTERNATIONAL AND COMMERCIAL PARTICIPATION

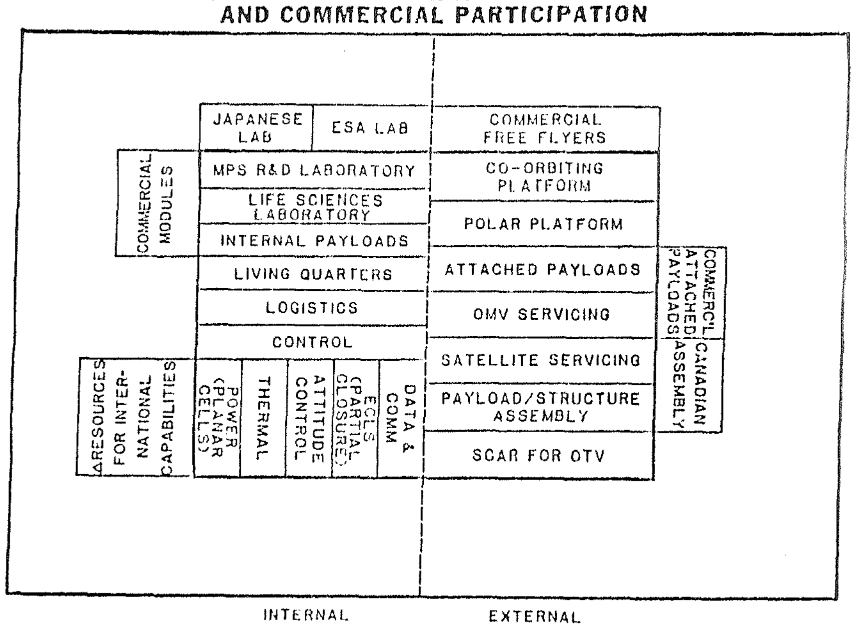


Figure 10

SCOPE OF GROWTH CONFIGURATION

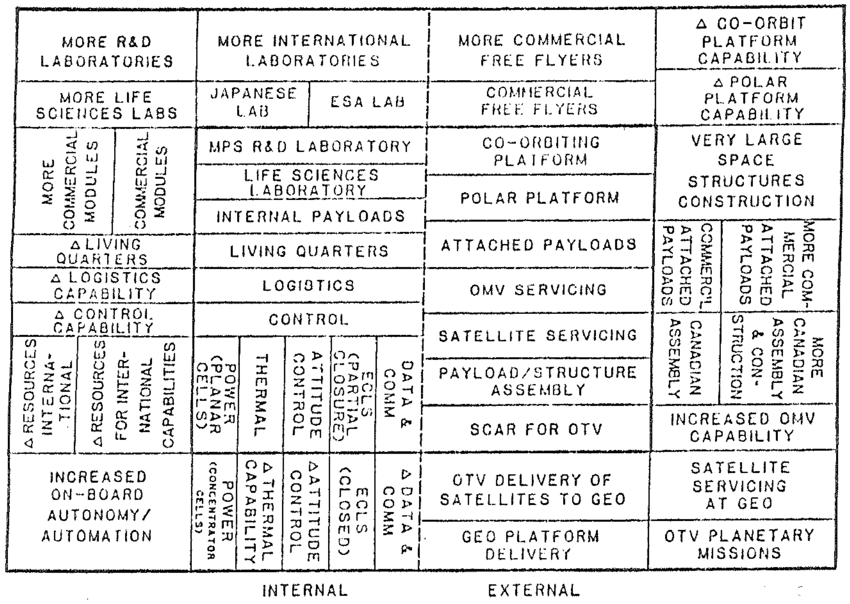


Figure 11

THE SPACE STATION PROGRAM WILL EVOLVE THROUGH A "BLOCK" SERIES

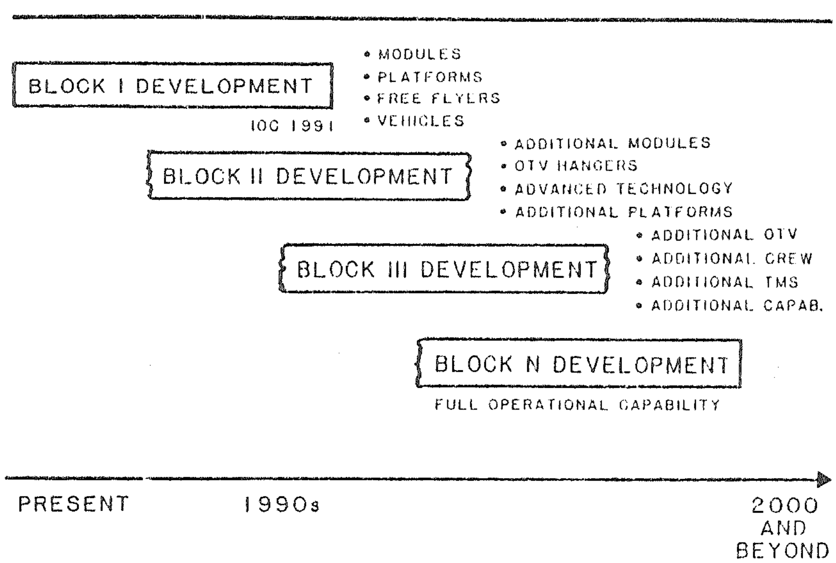


Figure 12

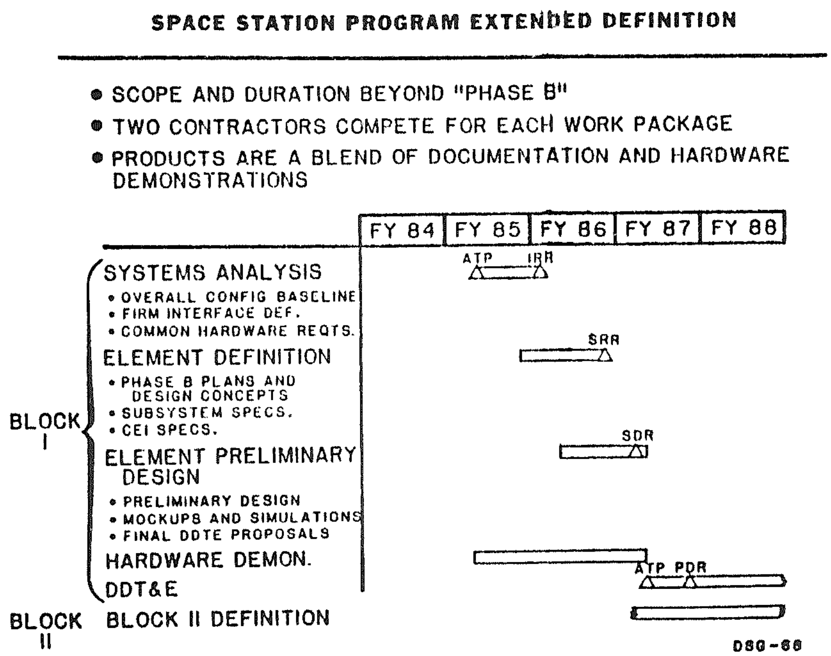


Figure 13

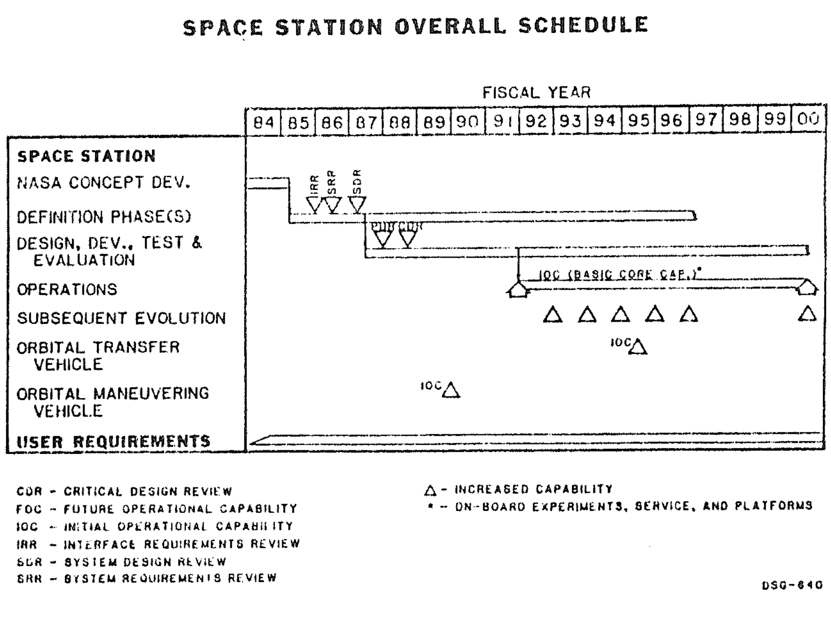


Figure 14

SPACE STATION PLANNING SCHEDULE

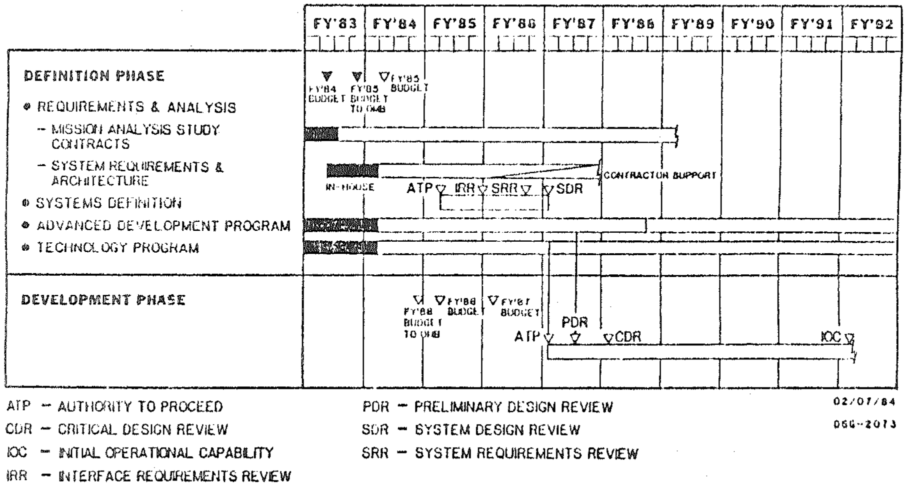


Figure 15

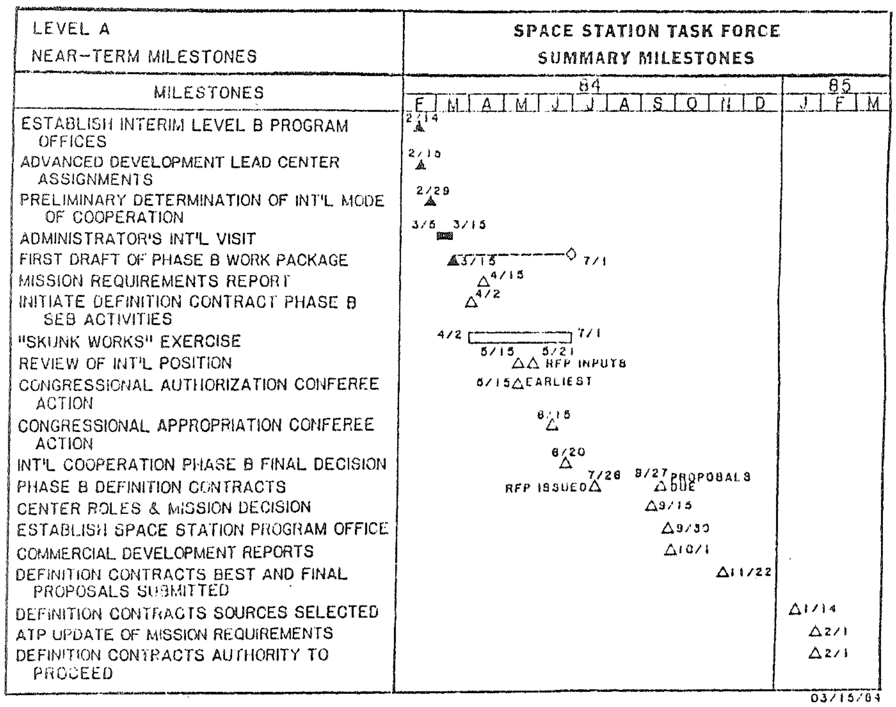


Figure 16

SPACE STATION PROGRAM DEFINITION ACTIVITY

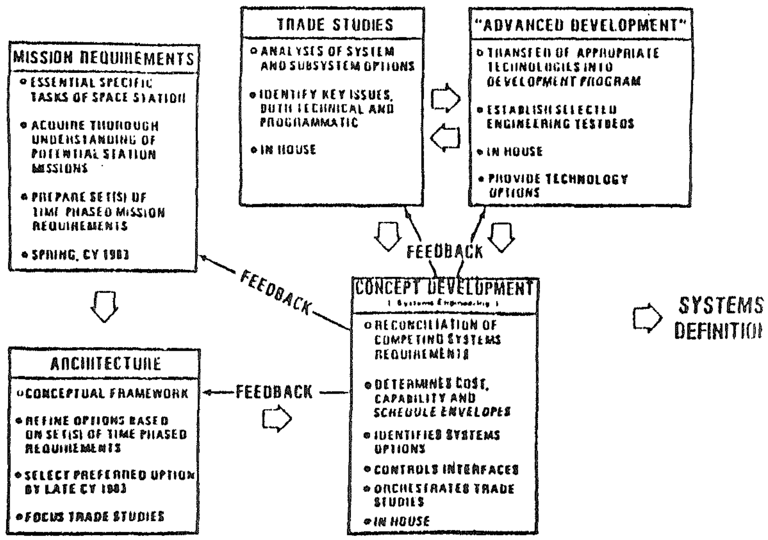


Figure 17

SPACE STATION SERVICING CAPABILITY

THE SPACE STATION BASE WILL HAVE THE CAPABILITY TO SERVICE OR PROVIDE SERVICING SUPPORT FOR:

- PAYLOADS ATTACHED TO THE STATION
- SATELLITES BROUGHT TO THE STATION BY THE TMS OR SERVICED REMOTELY BY THE TMS
- TMS BASED AT THE STATION
- CO-ORBITING PLATFORM AND ITS PAYLOADS
- LARGE SPACE STRUCTURE TDM'S
- PAYLOADS TO BE PLACED IN ORBIT BY THE TMS AND TO BE LAUNCHED TO HIGHER ENERGY ORBITS
- SPACE-BASED REUSEABLE OTV
- SATELLITES IN GEO SERVICED REMOTELY BY THE TMS

SERVICING FUNCTIONS AT THE SPACE STATION WILL INCLUDE:

- REPLENISHMENT OF CONSUMABLES
  PROPELLANTS
  PRESSURANTS
  COOLANTS
- RECHARGING/REPLACEMENT OF BATTERIES
- CONSTRUCTION OF LARGE SPACE STRUCTURES
- ASSEMBLY (POSSIBLE FUELING) AND MATING OF PAYLOADS
- CHECKOUT
  - SATELLITES
  - TMS
  - OTV
  - PAYLOADS
- REPAIR AND UPGRADING, PRIMARILY BY ORU EXCHANGE

Figure 18

SERVICING FACILITIES AT THE SPACE STATION

COMMON FACILITIES

- SUPPORT STRUCTURE
- REMOTE MANIPULATOR SYSTEM (RMS) - RELOCATABLE
- MANIPULATOR FOOT RESTRAINT (MFR)
- MANNED MANEUVERING UNITS (MMU) - TWO
- MODULAR EQUIPMENT STORAGE ASSEMBLY (MESA)
- GENERAL STORAGE AREA - ENCLOSED
  MMU'S, MFR, MESA
- WORK AREA (CONSTRUCTION OF LARGE SPACE STRUCTURES)
- EXTERNAL WORK SITE MONITORING AND CONTROL STATION (IN A PRESSURIZED MODULE)

MULTIPURPOSE PRESSURIZED WORK VOLUME-NEED TO BE DETERMINED

ORBITAL TRANSFER VEHICLE (OTV) FACILITIES

- BERTHS - TWO
- PROPELLANT AND PRESSURANT TANKS
- ELECTRICAL POWER STATION
- CHECKOUT EQUIPMENT
- HANGARS UNPRESSURIZED - TWO
- PAYLOAD ASSEMBLY/CHECKOUT AREA - ENCLOSED
- STORAGE AREA - ENCLOSED
  SPARE ASSEMBLIES, ORU'S, MANNED GEO MISSION MODULE

Figure 19

THE SERVICING FACILITY AND OPERATIONS

- PLACE SEVERE REQUIREMENTS ON THE SPACE STATION
  - SAFETY
  - CONTAMINATION
  - CONTROL STATION VIEWING OF SERVICING OPERATIONS
  - APPROACH/DEPARTURE CORRIDORS
  - THERMAL CONTROL OF FLUIDS STORED ON THE STATION
  - EVA CORRIDORS
  - ACCESS TO PRESSURIZED WORK VOLUME (IF DEEMED NECESSARY)
  - CONSUMABLES AND CARGO TRANSFER
  - ATTITUDE CONTROL AND PROPULSION
  - RMS REACH CAPABILITY
  - POSSIBLE CRYOGENIC PROPELLANT BOIL-OFF USAGE (ECLS, PROPULSION, POWER)
  - GROWTH CAPABILITY
- AFFECT OTHER ELEMENTS OF THE SPACE STATION
  - SCIENTIFIC INSTRUMENTS FIELDS OF VIEW
  - G LEVEL OF THE LABORATORIES
  - CONTAMINATION OF ENVIRONMENT

THE SERVICING FACILITY AND OPERATIONS ARE A MAJOR DRIVER FOR BOTH THE INITIAL AND GROWTH STATIONS

Figure 20

CRITICAL TECHNOLOGY DEVELOPMENT FOR OMV/OTV/SATELLITE SERVICING

- FLUID MANAGEMENT
  - CRYOGENICS
  - STORABLE FLUIDS
- LONG-TERM ORBITAL STORAGE OF CRYOGENICS
- CONTAMINATION CONTROL/REMOVAL
- IMPROVED EXTRAVEHICULAR MANEUVERING UNIT (EMU)
- ROBOTIC SERVICING CAPABILITY
- RENDEZVOUS, APPROACH, AND BERTHING
  - OMV
  - OTV
  - SATELLITES
  - PLATFORM

Figure 21

TOP LEVEL SERVICING FACILITY ISSUES

- OTV PROPELLANT DEPOT LOCATION
  - ATTACHED
  - TETHERED
  - FREE FLYING
- DEGREE OF SERVICING AUTOMATION
  - INITIAL STATION
  - GROWTH STATION
- NEED FOR A PRESSURIZED WORK VOLUME

Figure 22

DESIRABLE FEATURES FOR A SPACE STATION BASED OTV

- SPACE MAINTAINABLE
- MODULAR
- HIGH REUSEABILITY
- SIMPLE PAYLOAD INTEGRATION AND SERVICING CAPABILITY
- SYNERGISTIC WITH SPACE STATION SYSTEMS/ ELEMENTS
- COMMONALITY WITH SPACE STATION SYSTEMS/ ELEMENTS
- STANDARDIZED INTERFACES – OMV, SATELLITES, SPACE STATION
- GROWTH CAPABILITY
- HIGH EFFICIENCY (LOW WEIGHT, HIGH ISP)
- NON-CONTAMINATING
- WIDE THRUST LEVEL CAPABILITY

Figure 23

PROPOSED OTV TECHNOLOGY DEVELOPMENT FLIGHT EXPERIMENTS

SHUTTLE SORTIE FLIGHTS (1987 - 1990)

- PROPELLANT TRANSFER, STORAGE, AND REFRIGERATION/ RELIQUEFACTION
- DOCKING AND BERTHING
- EMU/EVA OPERATIONS
- PAYLOAD MATING/INTERFACE
- OTV SHELTER STRUCTURE
- SERVICING FACILITIES/EQUIPMENT

TECHNOLOGY DEVELOPMENT MISSIONS ON SPACE STATION (1991 -)

- PROPELLANT TRANSFER, STORAGE, AND REFRIGERATION/ RELIQUEFACTION
- DOCKING AND BERTHING
- MAINTENANCE
- PAYLOAD INTEGRATION

SPACE-BASED OTV OPERATIONS (1995)

Figure 24

ORBITAL TRANSFER VEHICLE (OTV) - SPACE STATION PROGRAM INTERFACES

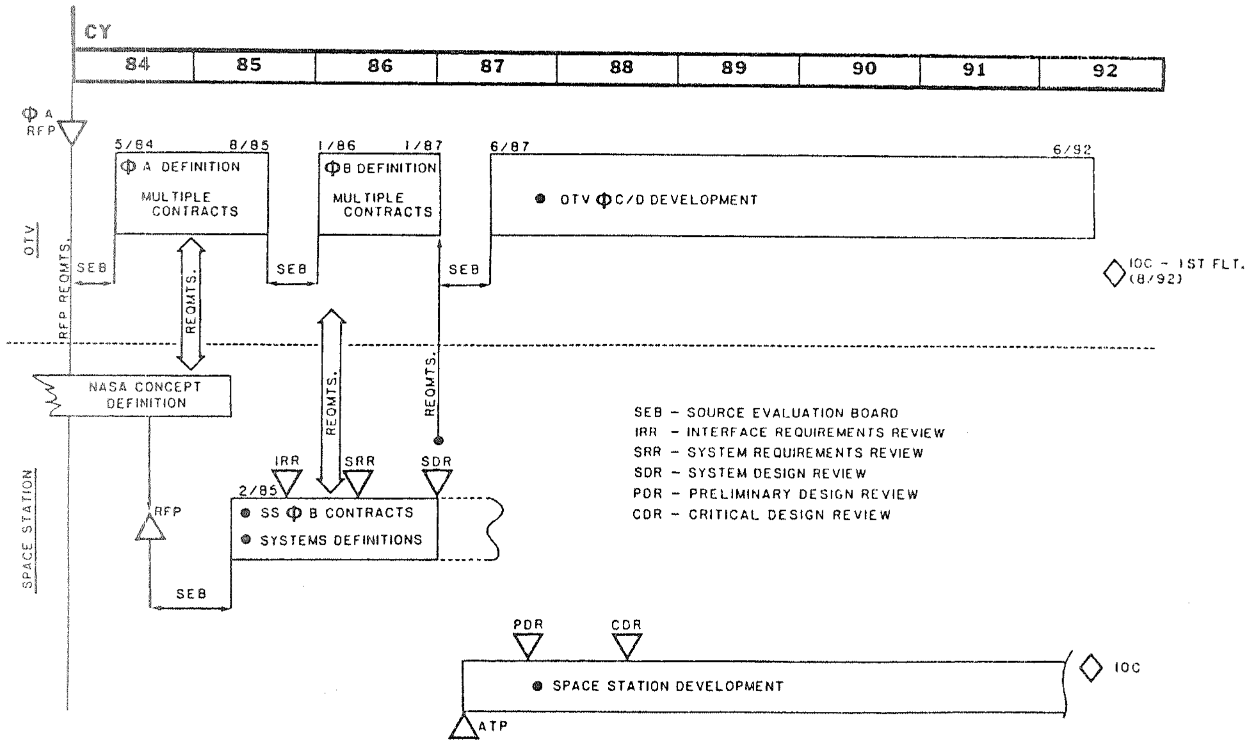


Figure 25

PERFORMANCE ASSESSMENT OF AERO-ASSISTED ORBITAL TRANSFER VEHICLES

Richard W. Powell, Theodore A. Talay, Alan W. Wilhite
John J. Rehder, Nancy H. White, J. Chris Naftel, Howard W. Stone,
James P. Arrington, and Ronald S. McCandless
NASA Langley Research Center

The NASA Langley Research Center is performing analyses of aero-assisted orbital transfer vehicles. The studies to date have been to determine the aerodynamic characteristics over the flight profile and three- and six-degree-of-freedom performance analyses.

The important results, to date, are: 1) The Aerodynamic Preliminary Analysis System, an interactive computer program, can be used to predict the aerodynamics (performance, stability, and control) for these vehicles; 2) the performance capability, e.g. maximum inclination change, maximum heating rate, and maximum sensed acceleration, can be determined using continuum aerodynamics only; 3) guidance schemes can be developed that allow for errors in atmospheric density prediction, mispredicted trim angle of attack, and off-nominal atmospheric interface conditions, even for vehicles with a low lift-to-drag ratio; and 4) multiple pass trajectories can be used to reduce the maximum heating rate.

FLIGHT PROFILES

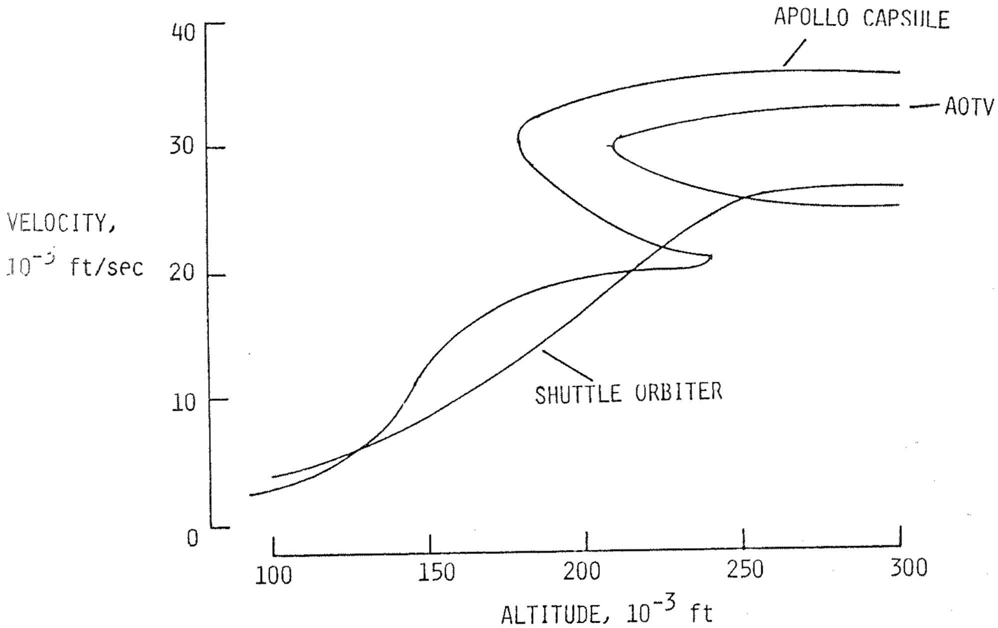


Figure 1

AOTV CONFIGURATION FOR LOW LIFT/DRAG

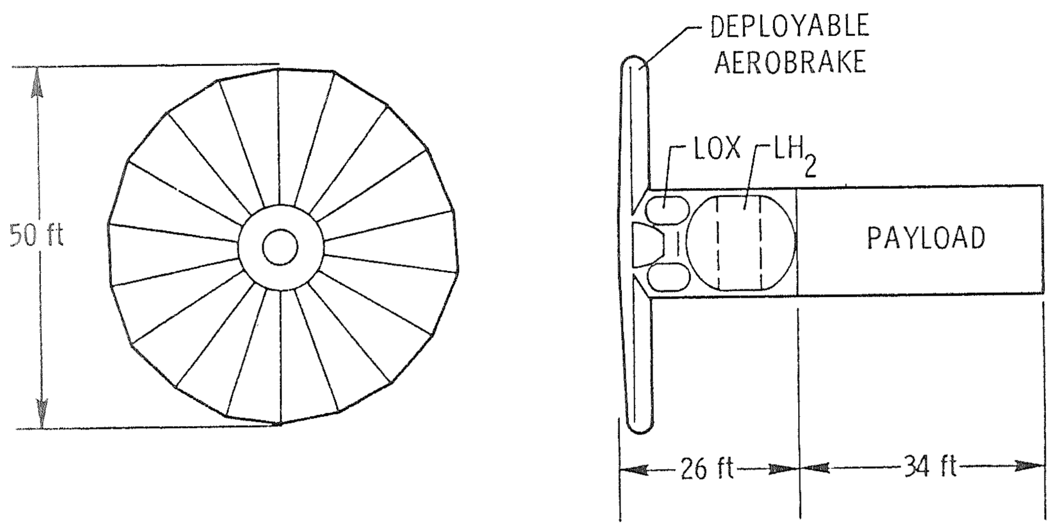


Figure 2

LOW L/D CONCEPT PERFORMANCE AERODYNAMICS

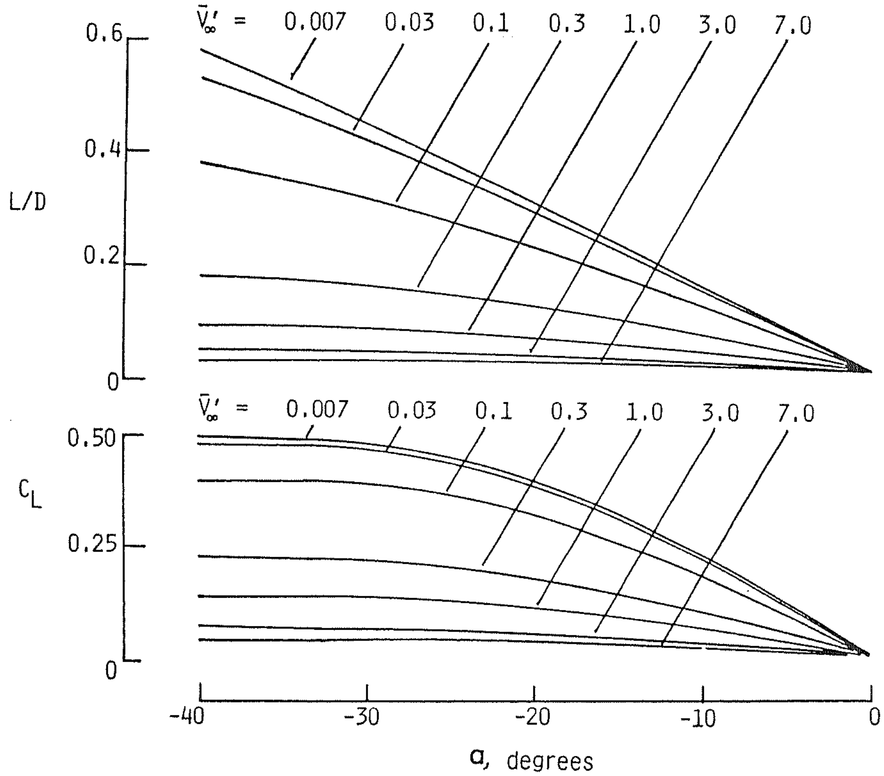


Figure 3

AOTV CONFIGURATION FOR MEDIUM LIFT/DRAG

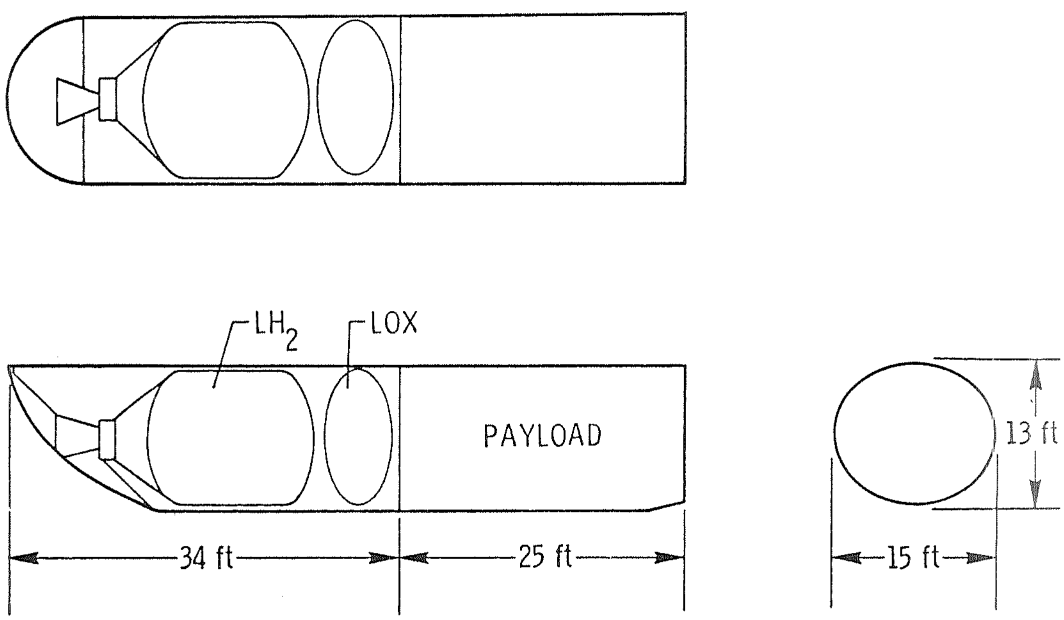


Figure 4

MID L/D CONCEPT PERFORMANCE AERODYNAMICS

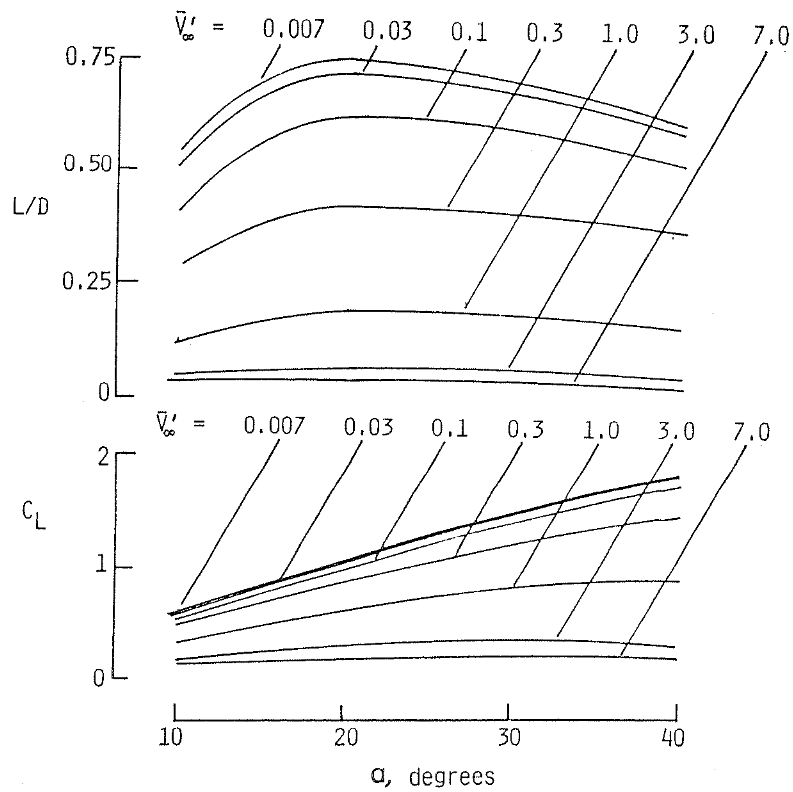


Figure 5

AOTV CONFIGURATION FOR HIGH LIFT/DRAG

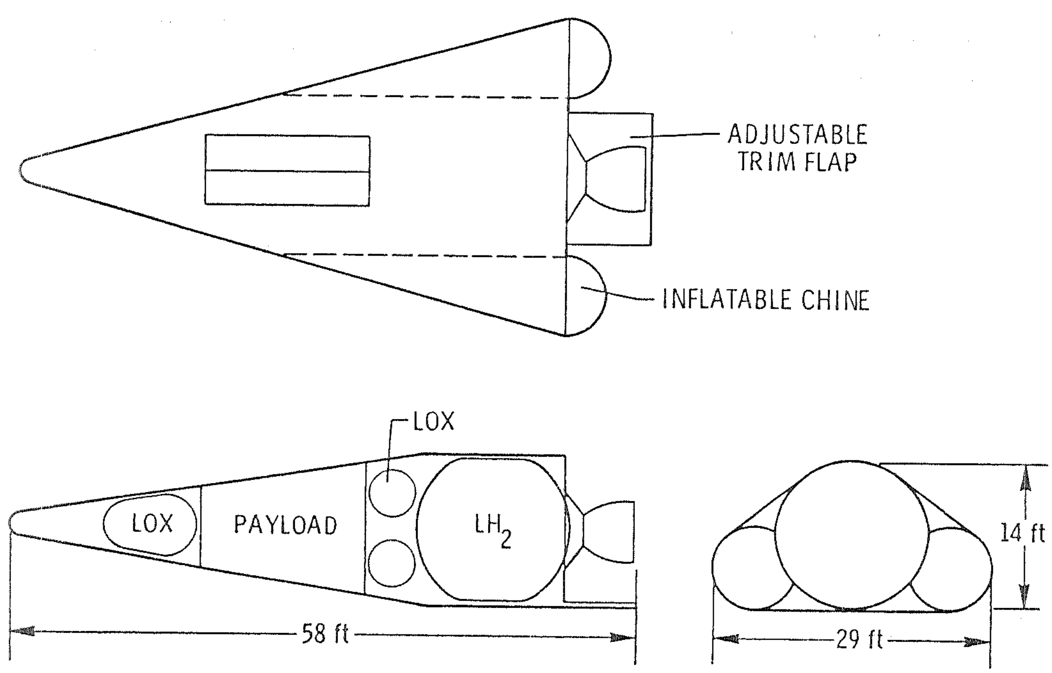


Figure 6

HIGH L/D PERFORMANCE AERODYNAMICS

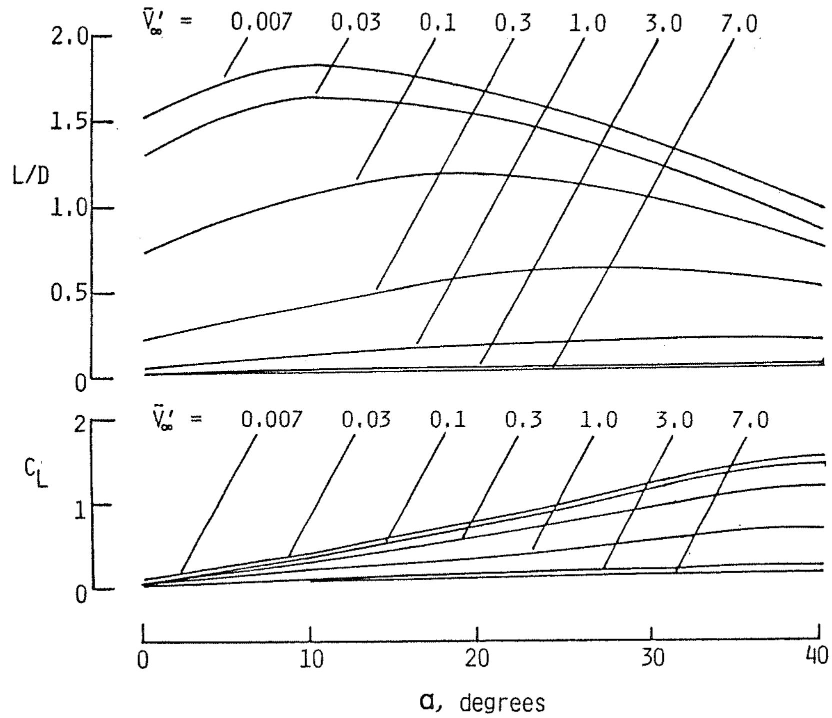


Figure 7

AOTV L/D PERFORMANCE COMPARISON

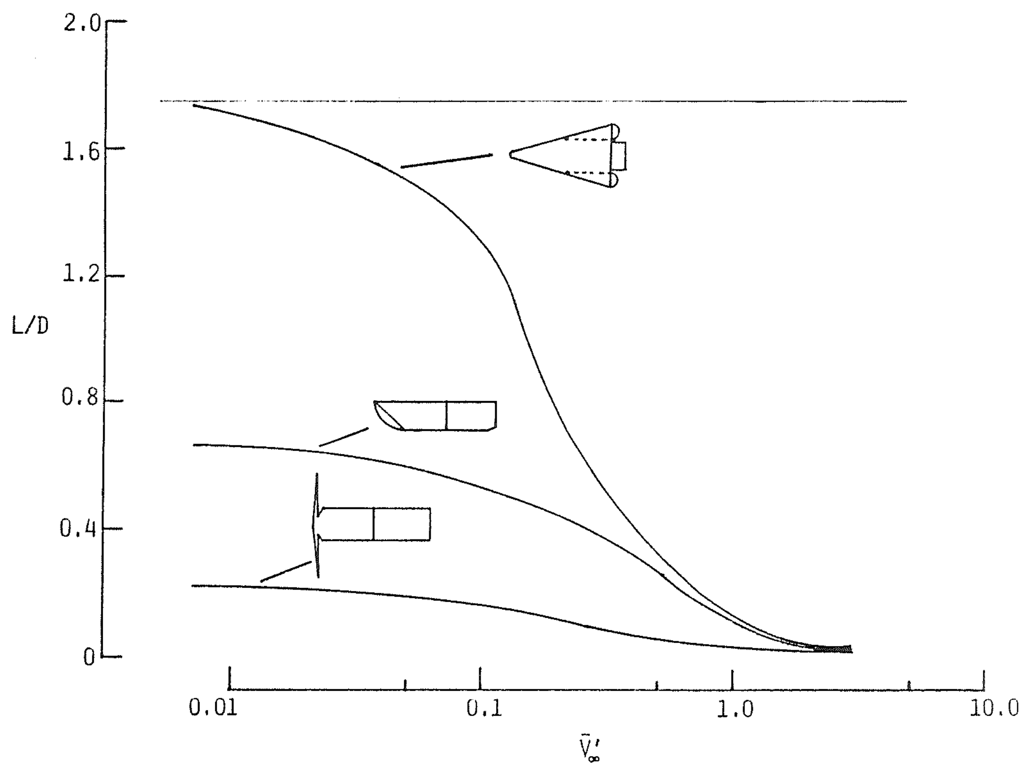


Figure 8

GEOSYNCHRONOUS ORBIT MISSION USING AN AEROASSISTED TRANSFER VEHICLE

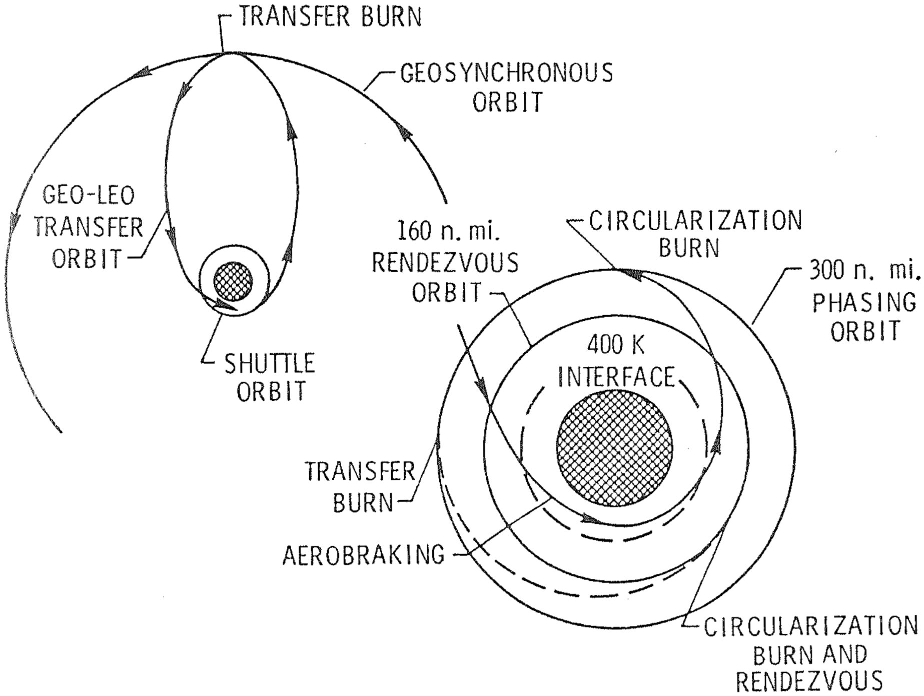


Figure 9

ANALYSIS TECHNIQUE

- o 3-D PROGRAM TO OPTIMIZE SIMULATED TRAJECTORIES (POST)
- o GEO-LEO TRANSFER——TIMING, DURATION, ANGLE
- o ATMOSPHERIC PASS——400,000 FT INTERFACE (1962 U.S. STANDARD)
  - o ALL VEHICLES HAVE A LIFT CAPABILITY
  - o MAINTAIN CONSTANT ANGLE OF ATTACK DURING PASS
  - o ROLL VEHICLE ABOUT VELOCITY VECTOR TO VARY LIFT DIRECTION
  - o TARGET TO 300 NMI PHASING ORBIT, 28.5° INCLINATION, SAME LONGITUDE OF ASCENDING NODE AS SHUTTLE
  - o 3-BURN PROPULSIVE SEQUENCE LEADS AOTV TO RENDEZVOUS WITH SHUTTLE ORBITER

Figure 10

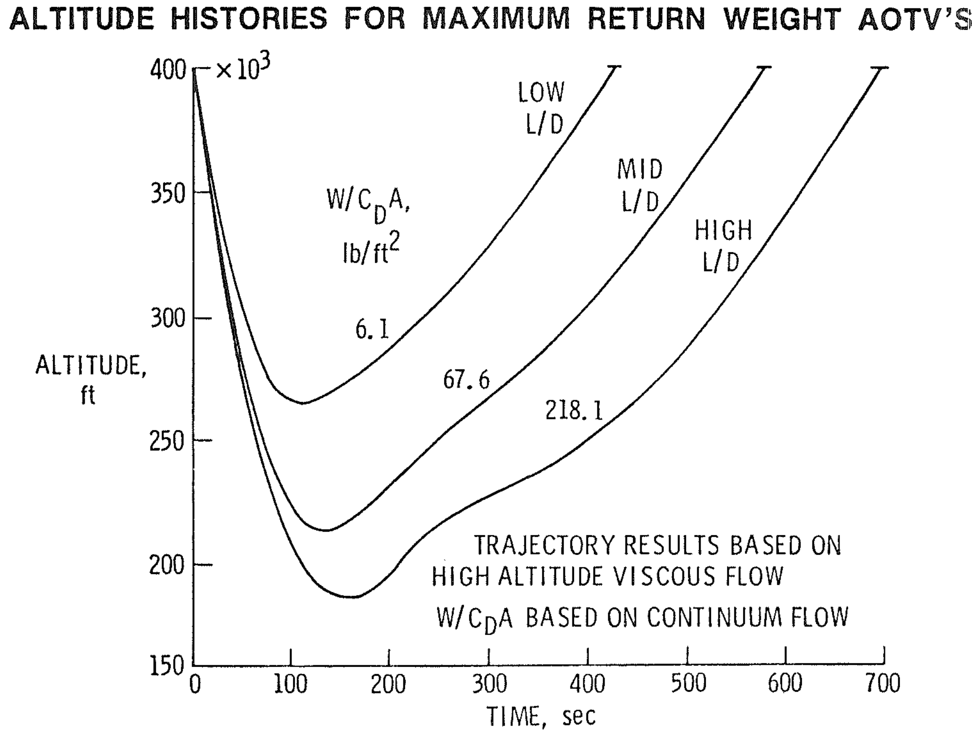


Figure 11

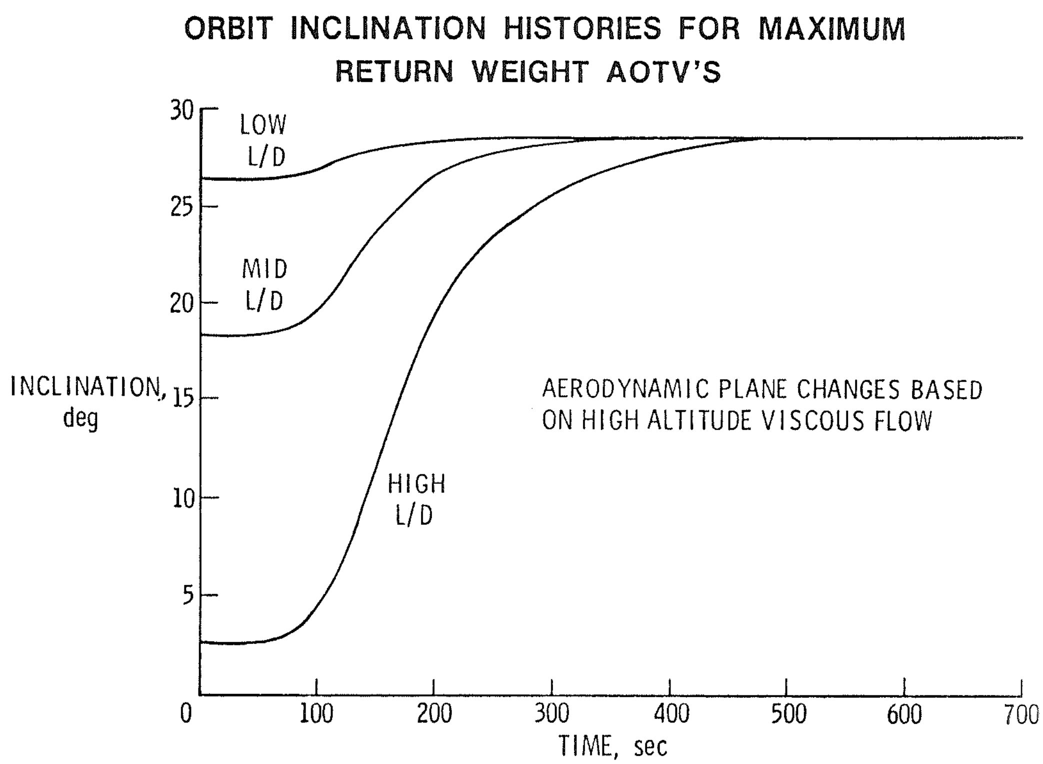


Figure 12

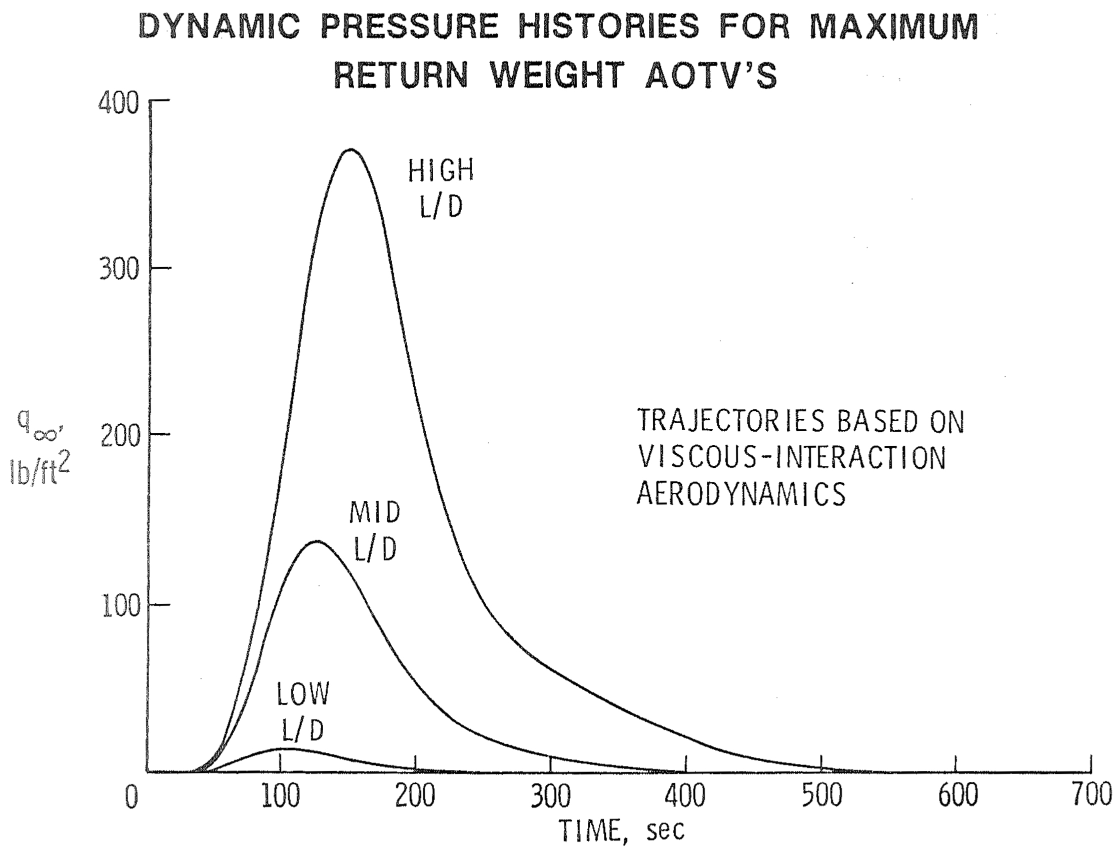


Figure 13

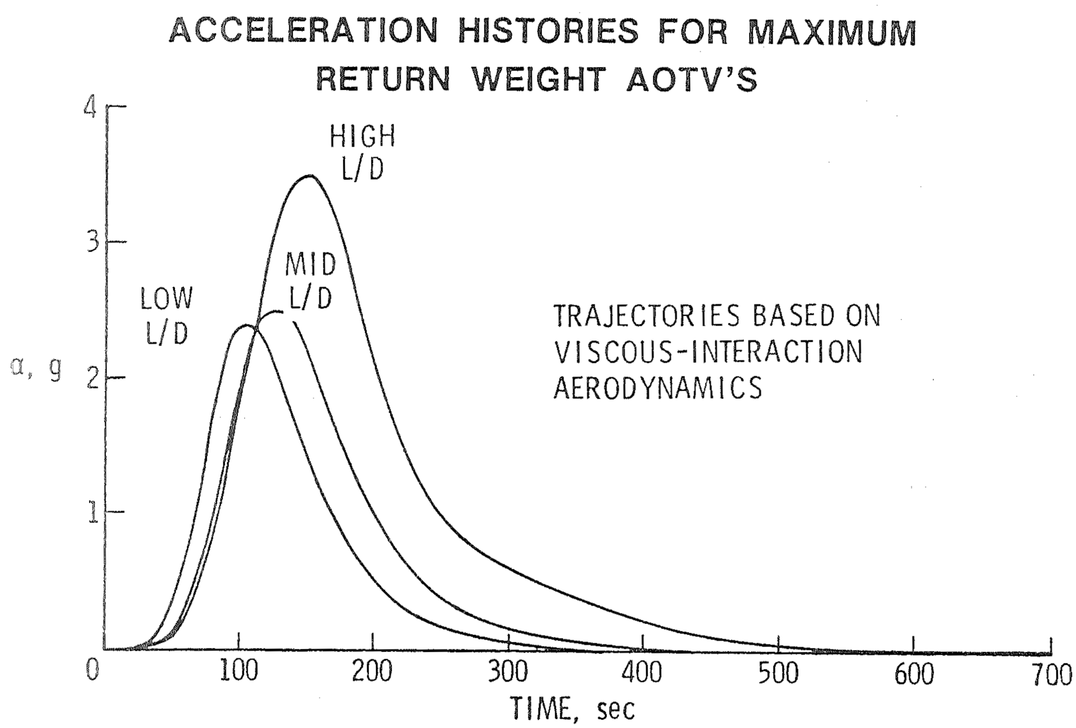


Figure 14

REFERENCE HEATING RATE HISTORIES FOR MAXIMUM RETURN WEIGHT AOTV'S

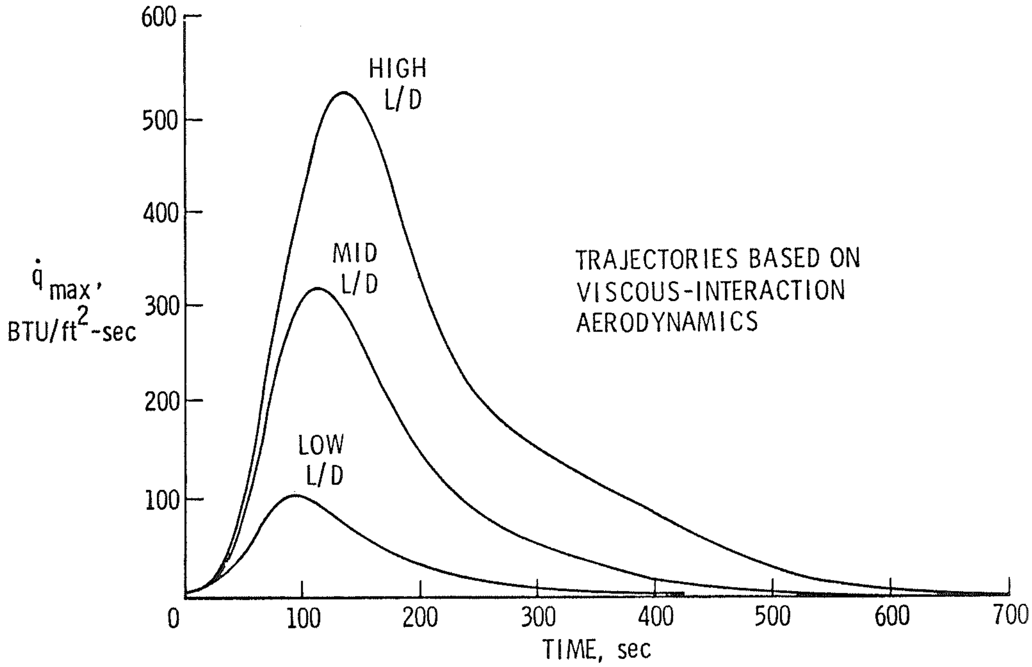


Figure 15

PERFORMANCE COMPARISON OF AN ALL-PROPULSIVE OTV
WITH AOTV's FOR GEO ROUND-TRIP MISSIONS

OTV TYPE	ALL PROPULSIVE	LOW L/D	MID L/D	HIGH L/D
INITIAL WEIGHT, LB	66,000	66,000	66,000	66,000
WEIGHT RETURNED TO SHUTTLE, LB	9,666	15,833	16,396	16,987
MAXIMUM STAGNATATION POINT HEATING RATE TO A 1 FOOT RADUS SPHERE, BTU/FT^2 -SEC		102	317	372
MAXIMUM SENSED ACCELERATION, G'S		2.40	2.51	3.50

Figure 16

LIFT/DRAG HISTORIES OF MAXIMUM RETURN WEIGHT AOTV'S

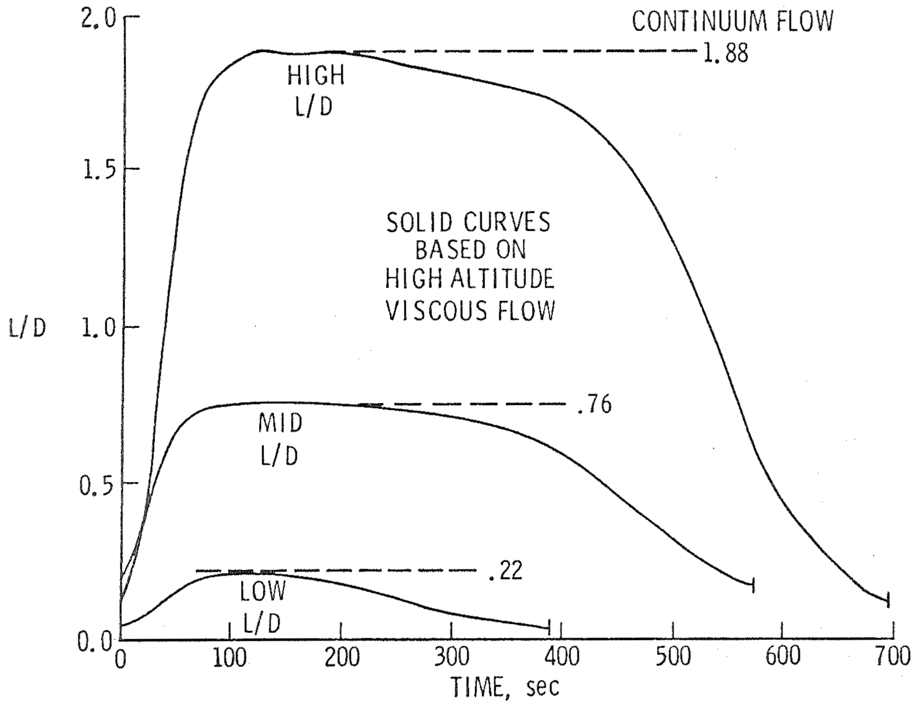


Figure 17

AOTV WEIGHT RETURNED TO SHUTTLE

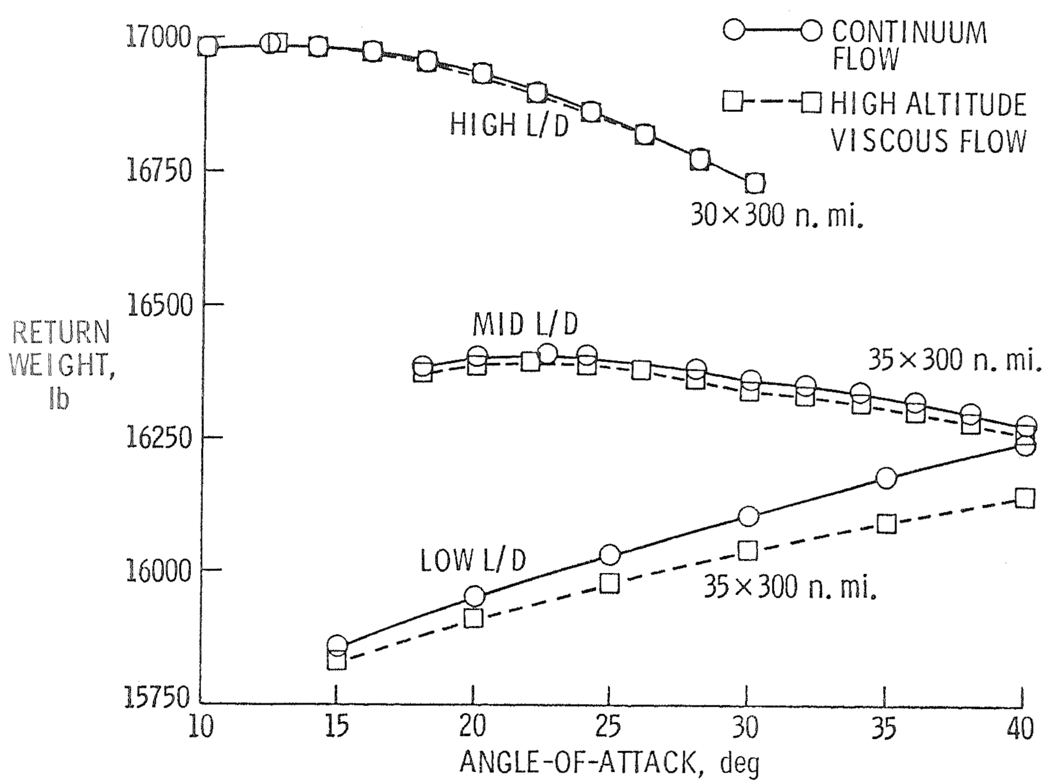


Figure 18

SIX-DEGREE-OF-FREEDOM SIMULATION ANALYSIS

ANALYSIS UNDERTAKEN TO:

1) SIZE THE REACTION CONTROL SYSTEM (RCS)

2) EVALUATE GUIDANCE ALGORITHMS

3) CONFIRM THREE-DEGREE-OF-FREEDOM ANALYSIS

CONTROL SYSTEM DESIGNED UTILIZED RCS ONLY

THREE GUIDANCE ALGORITHMS EVALUATED

1) PREDICTIVE TECHNIQUE

2) DRAG REFERENCE ALGOTITHM DERIVED BY OLIVER HILL OF NASA-JSC

3) REFERENCE ORBITAL ENERGY - FLIGHT PATH ANGLE REFERENCE ALGORITHM

Figure 19

LOW L/D CONFIGURATION RESPONSE CHARACTERISTICS

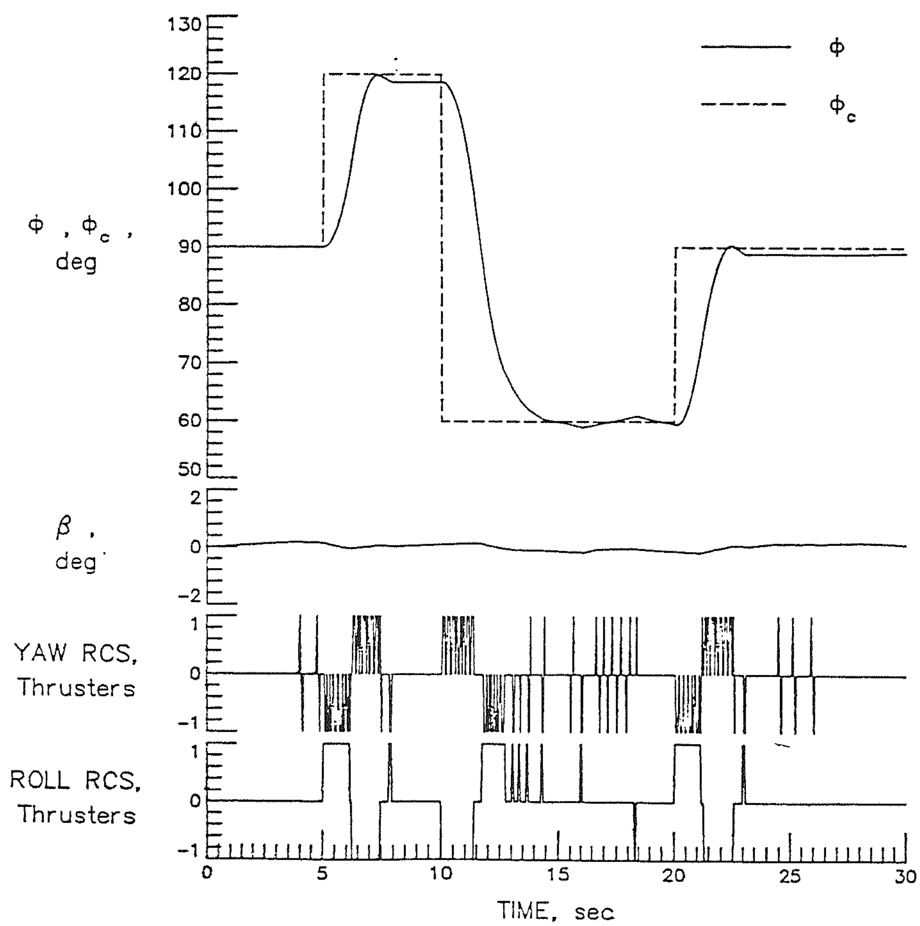


Figure 20

GUIDANCE ALGORITHM

- O AOTV IS COMMANDED TO FLY THE OPTIMUM ORBITAL ENERGY VS INERTIAL FLIGHT PATH ANGLE PROFILE THAT HAD BEEN DETERMINED FROM A 3 DEGREE-OF-FREEDOM ANALYSIS
- O AOTV IS COMMANDED TO FLY CONSTANT ANGLE OF ATTACK
- O INCLINATION IS CONTROLLED BY ROLL REVERSALS

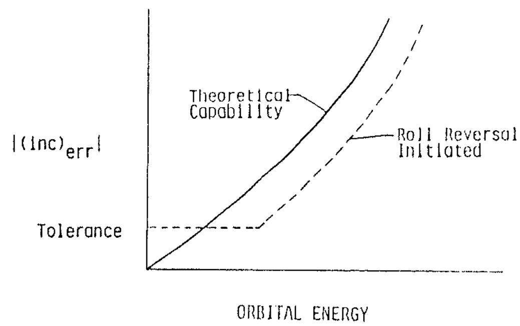


Figure 21

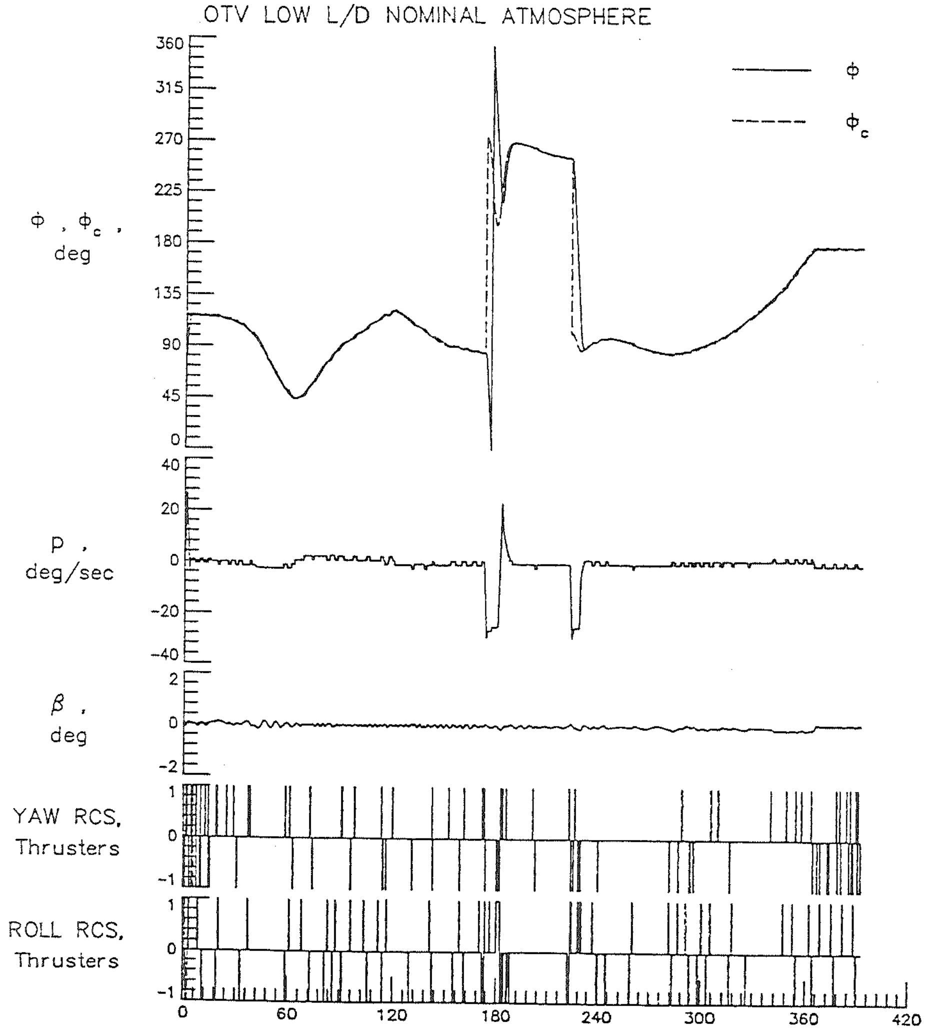


Figure 22

SHUTTLE-DERIVED ALTITUDE-DENSITY PROFILES

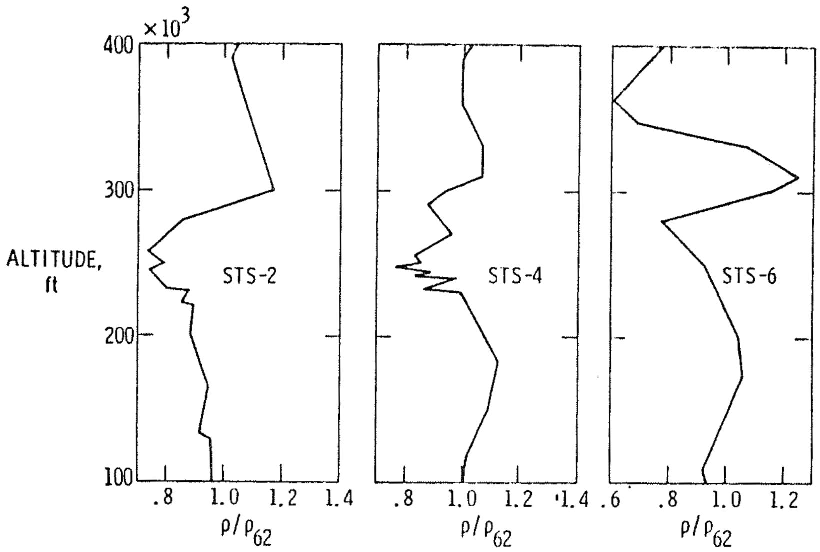


Figure 23

OTV LOW L/D NOMINAL ATMOSPHERE

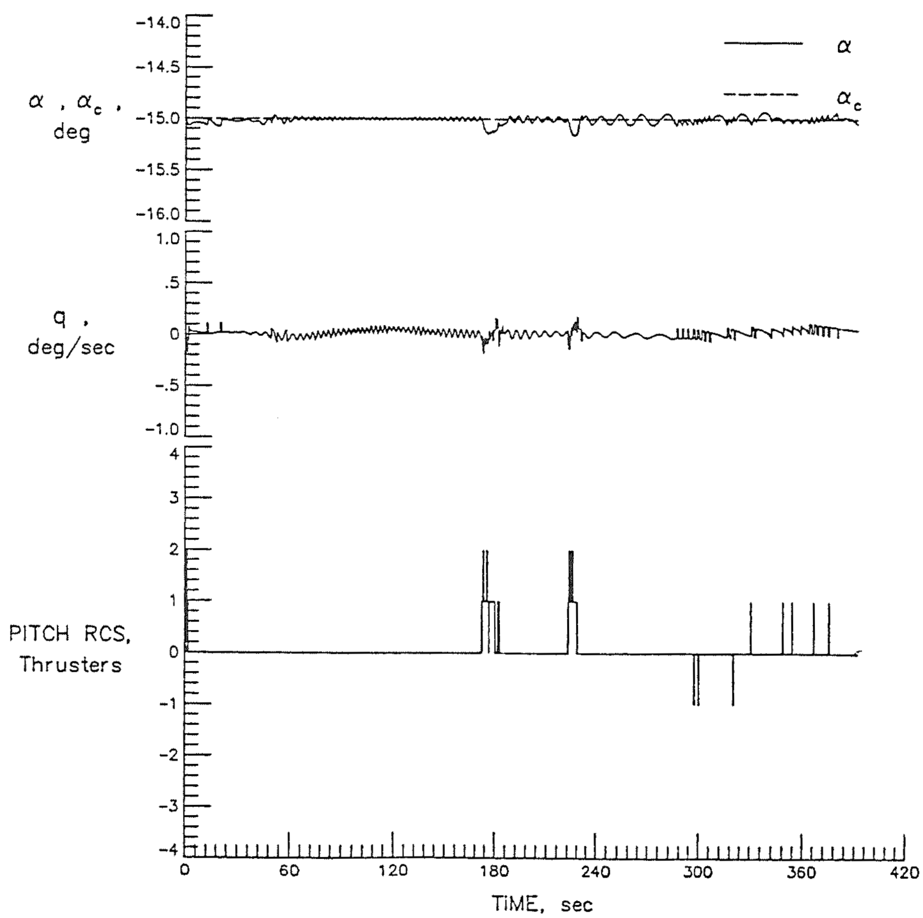


Figure 24

COMPARISON OF ATMOSPHERIC PASS FOR LOW L/D CONFIGURATION

	ATMOSPHERES					
	1962 STANDARD	+25%	-25%	STS-2	STS-4	STS-6
EXIT APOGEE, NMI	306	287	323	313	311	314
EXIT PERIGEE, NMI	34	31	35	35	35	35
EXIT INCLINATION, DEG	28.5	28.5	28.5	28.5	28.5	28.5
TOTAL ATMOSPHERIC PASS TIME, SEC	392	380	407	408	399	411
PITCH RCS ON-TIME, SEC	19.44	22.48	24.44	18.72	19.96	28.00
ROLL RCS ON-TIME, SEC	18.16	22.28	20.72	18.56	18.68	20.92
YAW RCS ON-TIME, SEC	20.08	23.60	24.48	21.76	21.2	29.84

Figure 25

COMPARISON OF ATMOSPHERIC PASS FOR LOW L/D CONFIGURATION

	TRIMMED ANGLE-OF-ATTACK VARIATIONS						
	$\alpha_T = \alpha_D$	$\alpha_T = \alpha_D - 1$	$\alpha_T = \alpha_D + 1$	$\alpha_T = \alpha_D - 3$	$\alpha_T = \alpha_D + 3$	$\alpha_T = \alpha_D - 5$	$\alpha_T = \alpha_D + 5$
EXIT APOGEE, NMI	306	309	305	295	301	297	300
EXIT PERIGEE, NMI	34	34	34	32	33	32	33
EXIT INCLINATION, DEG	28.5	28.5	28.5	28.5	28.5	28.5	28.1
TOTAL ATMOSPHERIC PASS TIME, SEC	392	393	392	393	390	393	389
PITCH RCS ON-TIME, SEC	19.44	23.52	17.16	29.52	3.20	36.40	2.36
ROLL RCS ON-TIME, SEC	18.16	20.48	17.84	19.84	12.28	24.08	11.08
YAW RCS ON-TIME, SEC	20.08	26.32	18.56	29.84	14.0	36.48	17.68

Figure 26

COMPARISON OF ATMOSPHERIC PASS FOR LOW L/D CONFIGURATION

	ENTRY FLIGHT PATH ANGLE VARIATIONS						
	$\gamma_I = \gamma_D$	$\gamma_I = \gamma_D - .05$	$\gamma_I = \gamma_D - .05$	$\gamma_I = \gamma_D - .1$	$\gamma_I = \gamma_D + .1$	$\gamma_I = \gamma_D - .2$	$\gamma_I = \gamma_D + .2$
VACUUM PERIGEE, NMI	45.4	44.9	45.9	44.4	46.4	43.4	47.3
EXIT APOGEE, NMI	306	298	312	263	326	156	394
EXIT PERIGEE, NMI	34	32	35	27	36	-8	40
EXIT INCLINATION, DEG	28.5	28.5	28.5	28.4	28.5	28.5	28.5
TOTAL ATMOSPHERIC PASS TIME, SEC	392	389	394	382	398	369	406
PITCH RCS ON-TIME, SEC	19.44	21.56	23.48	21.68	23.12	15.76	13.72
ROLL RCS ON-TIME, SEC	18.16	19.32	18.72	17.68	19.24	27.36	13.92
YAW RCS ON-TIME, SEC	20.08	22.72	22.32	23.44	18.4	22.4	13.04

Figure 27

COMPARISON OF AOTV TRAJECTORIES

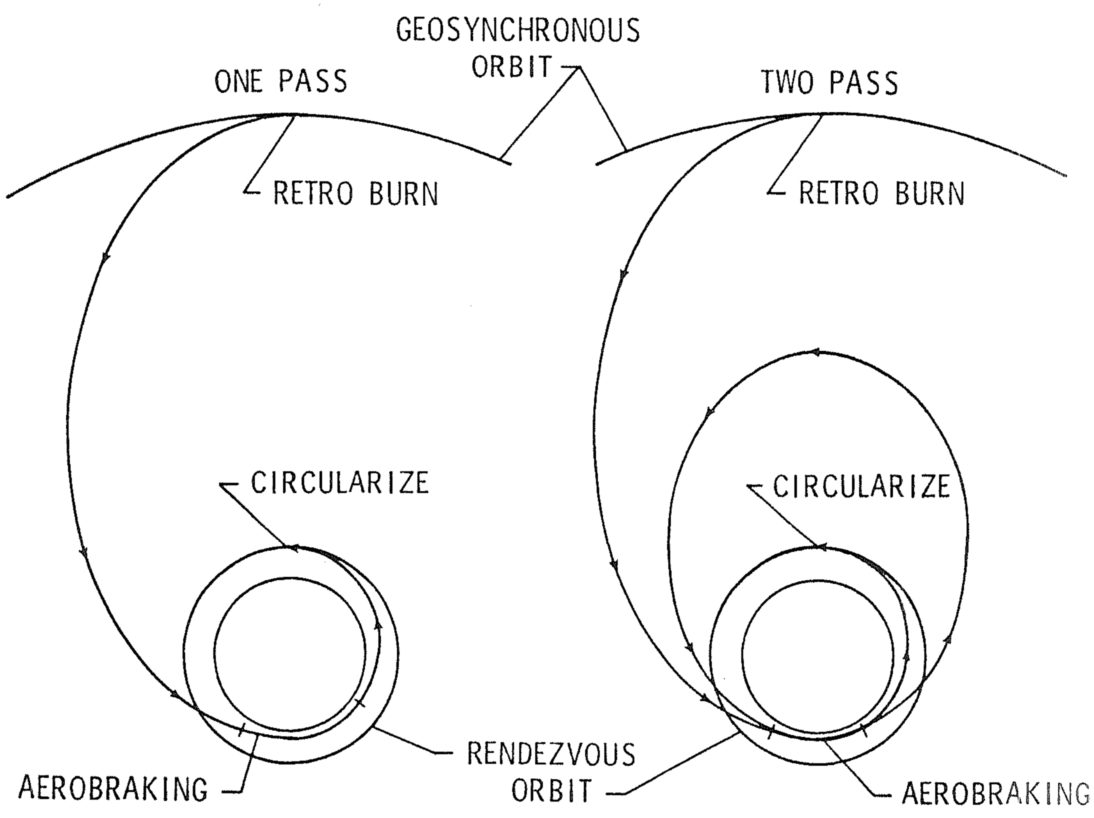


Figure 28

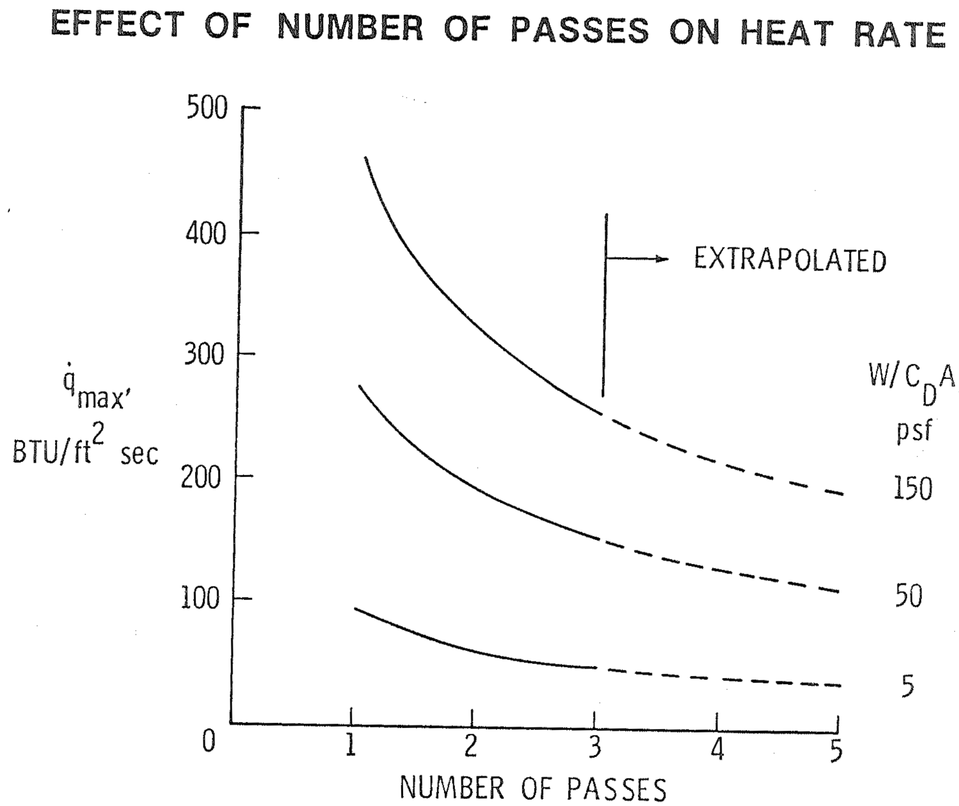


Figure 29

CONCLUSIONS

- APAS IS APPLICABLE FOR AOTV's AND CAN BE USED TO PREDICT AERODYNAMICS FROM THE FREE MOLECULAR FLOW REGION TO THE CONTINUUM REGION
- THREE DoF ANALYSIS SHOWED THAT CONTINUUM AERODYNAMICS IS ADEQUATE FOR PERFORMANCE EVALUATION
- SIX DoF ANALYSIS SHOWED CAPABILITY TO TOLERATE OFF-NOMINAL ATMOSPHERIC DENSITY PROFILES, MISS-PREDICTIVE TRIM ANGLE-OF-ATTACK, AND OFF-NOMINAL ATMOSPHERIC INTERFACE CONDITIONS
- MULTI-PASS TRAJECTORIES OFFER POTENTIAL TO REDUCE MAXIMUM HEATING RATES

Figure 30

LOW LIFT-TO-DRAG AERO-ASSISTED ORBIT TRANSFER VEHICLES

Dana G. Andrews and Richard T. Savage
Boeing Aerospace Company

The results of a systems analysis study conducted on low L/D aero-assisted orbit transfer vehicles (AOTV's) is presented. The objectives of this activity were to (1) systematically assess the technology requirements for this class of vehicle and formulate technology development plans and funding levels to bring the required technologies to readiness levels, and (2) develop a credible decision data base encompassing the entire range of low L/D concepts for use in future NASA AOTV studies.

The study approach was to select suitable AOTV concepts, address major feasibility issues, and generate workable configurations for use in trajectory/ aerothermal analyses. Subsystem trades examined the impact of different technology levels on vehicle performance and noted the levels required to meet basic operating requirements. Finally, technologies were ranked in order of importance towards meeting low L/D AOTV design goals, and program and technology funding costs were estimated.

Study results showed that each of the candidate low L/D concepts, the aerobrake, the lifting brake, and the aeromaneuvering concept could be made to work with technologies achievable by the early 1990's. All of the concepts required flexible structure with flexible thermal protection system (TPS) to be successfully integrated into the shuttle orbiter for launch, all required improvements in guidance and control (G&C) to fly the dispersed atmospheres at high altitude, and all concepts had potential to evolve from ground-based to space-based operations.

The critical advancements in technologies required to implement the low L/D AOTV concepts were in TPS, especially flexible TPS, in aerothermal prediction methods, and in G&C. Other areas where technology advancements appeared to be cost effective (i.e., savings in use outweighted development costs) were propulsion, atmospheric physics (prediction methods), rarified gas aerodynamics, and composite structures.

Study Objectives

- DEFINITION OF A TECHNOLOGY PLAN FOR LOW L/D AOTV'S
  - ENABLING AND HIGH PAYOFF TECHNOLOGIES IDENTIFIED
  - REALISTIC CONSTRAINTS ON TECHNOLOGY FUNDING LEVEL ASSUMED
  - TIME PHASED PLAN DEVELOPED FOR REASONABLE IOC DATE
- DEVELOPMENT OF A DECISION DATA BASE FOR FUTURE NASA AOTV STUDIES
  - INVESTIGATE CONCEPTS THROUGHOUT THE ENTIRE LOW L/D RANGE
  - ADDRESS THE CRITICAL VEHICLE TECHNOLOGIES
  - INCLUDE OPS ANALYSES
  - DEVELOP EVOLUTIONARY GROWTH SCENARIOS
  - ESTIMATE COSTS (NON-RECURRING, RECURRING AND OPS)

Figure 1

Technical Approach

- SYSTEMS TRADES
  - SELECT CANDIDATE CONCEPTS AND RESPOND TO FEASIBILITY ISSUES
  - USE WORKABLE CONFIGURATIONS IN TRAJECTORY/AEROTHERMAL ANALYSES
  - USE MANNED MISSION TO DESIGN ALTERNATE OPERATIONAL MODES
- SUBSYSTEM TRADES
  - BUILD FROM PHASE A-OTV DATA BASE
  - INCORPORATE ADVANCED TECHNOLOGIES
  - ASSESS TECHNOLOGY PAYOFFS
- TECHNOLOGY PLANNING
  - IDENTIFY CURRENT, NORMAL GROWTH, AND ACCELERATED GROWTH TECHNOLOGIES
  - RANK TECHNOLOGIES WITH RESPECT TO PROGRAM REQUIREMENTS
  - PLAN TECHNOLOGY DEVELOPMENT
- COST ANALYSES
  - USE WORK BREAKDOWN STRUCTURE TO ESTIMATE SUBSYSTEM COSTS
  - ESTIMATE PROGRAM COSTS
  - ESTIMATE TECHNOLOGY FUNDING REQUIREMENTS

Figure 2

Design Mission Requirements

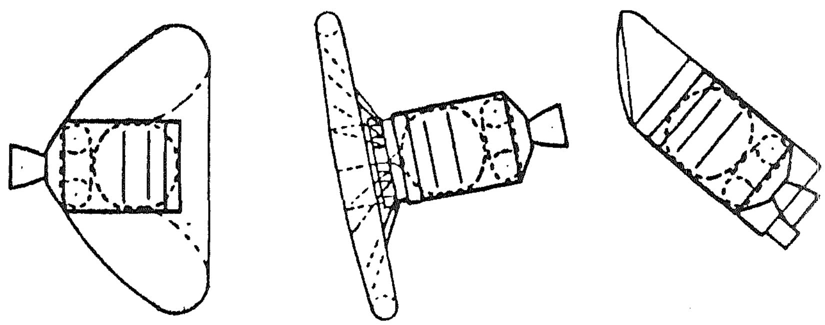
- BASELINE DESIGN MISSIONS (65K STS)
  - GEO DELIVERY
  - 5 x GEO DELIVERY
  - 6 HR. POLAR DELIVERY
- EVOLUTIONARY GROWTH MISSIONS
  - UNMANNED SERVICING (NOT A DESIGN DRIVER)
  - MANNED GEO MISSION (KEY DESIGN MISSION IN ALL MODELS)
    - 14,000 LB. ROUND TRIP
    - REQUIRES ALTERNATE OPERATING MODE

> - BASIC TECHNOLOGY TRADES WERE DONE USING VEHICLES SIZED FOR BASELINE MISSIONS
> - MANNED GEO MISSION WAS USED TO SIZE EVOLUTIONARY GROWTH CONFIGURATIONS AND DETERMINE WORTH OF ALTERNATE OPERATING MODES

Figure 3

Low L/D AOTV Characteristics

(BASELINE CONCEPTS)

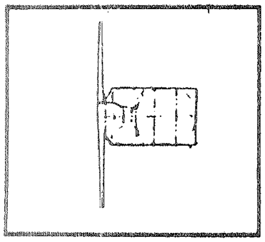


	AEROBRAKE (L/D=0)	LIFTING BRAKE (L/D=0.25)	AEROMANEUVERING (L/D=0.75)
BALLISTIC COEFFICIENT, $\frac{W}{C_D A}$	5-10 PSF	5-10 PSF	25-45 PSF
CONTROL TECHNIQUE	VARIABLE $C_D A$ USING INTERNAL PRESSURE	MOVEMENT OF CG IN Y-Z PLANE USING ELECTROMECHNICAL ACTUATORS	VARIABLE BANK ANGLE USING REDUNDANT RCS THRUSTERS
METHOD OF AERODYNAMIC TRIM	NONE (STABLE)	SAME	AERODYNAMIC TRIM SURFACES
KEY ISSUES	GUIDANCE & CONTROL IN 3σ ATMOSPHERE	GUIDANCE & CONTROL IN 3σ ATMOSPHERE	TRANSPIRATION COOLING OF NOSE CAP
	DYNAMIC STABILITY OF INFLATED STRUCTURE	FLOW INPINGEMENT ON BODY/PAYLOAD	THERMAL CONTROL

Figure 4

Lifting Brake Configuration

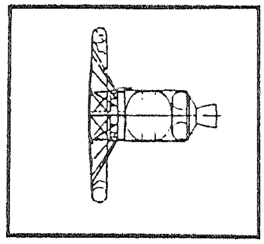
STUDY INITIATION



MAJOR CHANGE:

NON-POROUS & INFLATABLE BRAKE

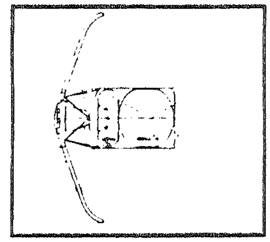
PROPOSED DESIGN



MAJOR CHANGE:

EDGE RADIUS

FINAL BASELINE



AREAS OF CONCERN

- DEPLOYMENT MECHANISM
- STRETCHABLE FABRIC
- POROUS HEAT SHIELD
- BANK ANGLE CONTROL
- GUIDANCE & CONTROL IN 3σ ATMOSPHERE
- FLOW IMPINGEMENT ON BODY/PAYLOAD
- STS INTEGRATION

AREAS OF CONCERN

- NO RETURN PAYLOAD
- FLOW IMPINGEMENT ON BODY/PAYLOAD
- G&C IN 3σ ATMOSPHERE
- STS INTEGRATION

AREAS OF CONCERN

- CONTROL WITH RETURN PAYLOAD
- STS INTEGRATION

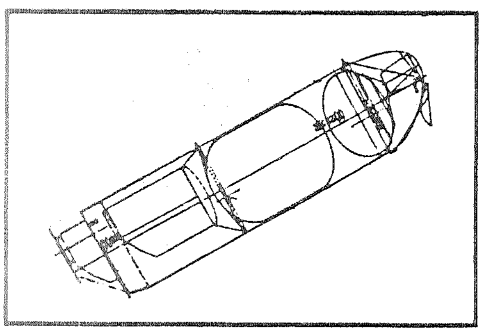
RATIONALE FOR CHANGE:

- NON-POROUS FABRIC: BOUNDARY LAYER TEMPERATURE CONTROL
- EDGE RADIUS: ELIMINATES FLOW IMPINGEMENT

Figure 5

Aeromaneuver Configuration

STUDY INITIATION

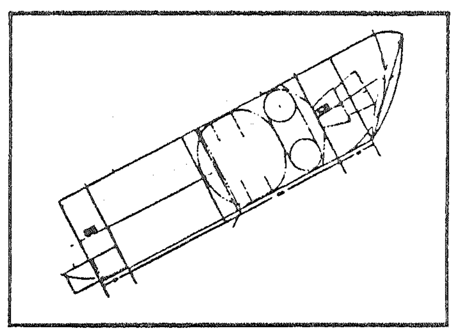


MAJOR CHANGE:

INFLATABLE NOSE,

TRANSPIR – ATION COOLED

BASELINE



AREA OF CONCERN

- LARGE DOORS/SEALS ADDS COMPLEXITY, WEIGHT

AREAS OF CONCERN

- TRANSPIRATION COOLED NOSE

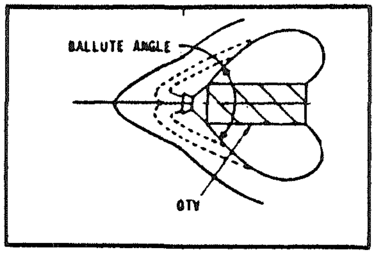
RATIONALE FOR CHANGE

- LIGHTER
- HIGHER RELIABILITY
- CHEAPER

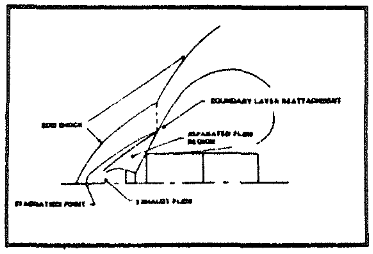
Figure 6

Aerobrake Configurations

STUDY INITIATION



BASELINE



AREAS OF CONCERN

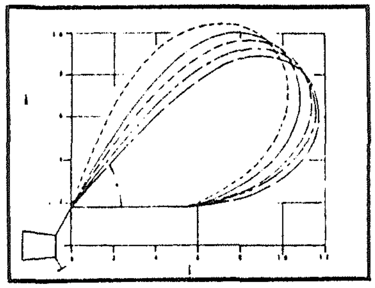
- JET COUNTERFLOW INTERACTION
- FLEXIBLE TPS
- INFLATED STRUCTURE STABILITY

CHANGE: NO JET COUNTERFLOW

AREAS OF CONCERN

- DYNAMIC STABILITY OF INFLATED STRUCTURE
- GUIDANCE & CONTROL IN 3σ ATMOSPHERE
- INTERACTION BETWEEN BALLUTE, JET COUNTER-FLOW, & FREE STREAM FLOW
- FLEXIBLE TPS

ALTERNATE



AREAS OF CONCERN

- FLEXIBLE TPS
- INFLATED STRUCTURE STABILITY

- TESTING OF FLEXIBLE BALLUTE AND TPS PRODUCED NO BALLUTE "SHOW STOPPERS"
- ALTERNATIVE PROPOSED TO ELIMINATE JET COUNTERFLOW UNCERTAINTY

Figure 7

System/Concept Findings

- NONE OF LOW L/D CONCEPTS ELIMINATED BY TECHNOLOGY ISSUES
  - MUCH OF THE TECHNOLOGY REQUIRED IS COMMON
- ALL CONCEPTS SIGNIFICANTLY BETTER UNDERSTOOD/IMPROVED
  - PERFORMANCE AND OPERATIONAL FLEXIBILITY ADVANTAGE TO AEROBRAKE
  - LIFTING BRAKE AND AEROMANEUVERING APPLICATIONS LIMITED BY AFT C.G. AND/OR FLOW IMPINGEMENT CONCERNS
- ALTERNATE OPERATIONAL SCENARIOS
  - ACC: EXCELLENT CONFIGURATION FOR LIFTING BRAKE; NO SIGNIFICANT IMPACT ON AEROBRAKE; AEROMANEUVERING IS NOT APPLICABLE
  - SPACEBASING/MANNED MISSION: NOT A SIGNIFICANT DISCRIMINATOR EXCEPT FOR PERFORMANCE
  - SDCLV: ATTRACTIVE OPTION WITH AEROASSIST
- FOR ALL CONCEPTS THE MAJOR UNRESOLVED ISSUES CONCERN REAL GAS FLOW EFFECTS AND THE DYNAMICS OF FLEXIBLE STRUCTURE AT THESE CONDITIONS
  - MORE TESTING REQUIRED TO PROVIDE DESIGN DATA
  - FLIGHT EXPERIMENTS NEEDED TO RESOLVE ALL DOUBTS
- FOR ALL CONCEPTS UPPER ATMOSPHERIC DISPERSIONS ARE A MAJOR DESIGN DRIVER
  - DESIGN DATA NEEDED – SOME TESTING REQUIRED
  - FLIGHT EXPERIMENTS IMPORTANT TO PROVE GN&C SYSTEMS
- NO SIGNIFICANT COST DISCRIMINATORS FOUND BETWEEN LOW L/D CONCEPTS

Figure 8

Technology Drivers

	TECHNOLOGY	ISSUE	COMMENTS
ENABLING	THERMAL PROTECTION	PEAK TEMPERATURE CAPABILITY	NEED TO ACCELERATE TECHNOLOGY GROWTH OF FLEXIBLE SURFACE INSULATION (FSI)
	AEROTHERMAL METHODS	THERMAL ENVIRONMENT PREDICTION	INCREASED ACCURACY IS REQUIRED TO FULLY CHARACTERIZE THERMAL ENVIRONMENT
	GN&C	ATMOSPHERIC GUIDANCE	AEROPASS REQUIRES MORE ADVANCED ADAPTIVE GUIDANCE SYSTEM
HIGH PAYOFF	PROPULSION	HIGHER PERFORMANCE ENGINE	DEVELOPMENT OF AN ADVANCED LH_2/LO_2 HIGHER I_{SP} ENGINE IS COST EFFECTIVE
	ATMOSPHERIC PHYSICS	HIGH ATMOSPHERE DESCRIPTION	BETTER UNDERSTANDING OF THE UPPER ATMOSPHERE SIMPLIFIES GN&C AND THERMAL PROBLEMS
	AERODYNAMICS	RAREFIED FLOW EFFECTS	ENHANCE GUIDANCE SYSTEM ACCURACY
	STRUCTURES	STRUCTURAL WEIGHT REDUCTION	UTILIZING ACCELERATED TECHNOLOGY GROWTH IS COST EFFECTIVE

Figure 9

Technology Ranking

RANK	TECHNOLOGY ITEM	BENEFIT	COST (FY 84→88) (MILLIONS OF DOLLARS)
1	THERMAL PROTECTION SYSTEM	• INCREASE FLEXIBLE & RIGID INSULATION TEMPERATURE CAPACITY • IMPROVE OPTICAL COATINGS • DEVELOP TRANSPIRATION COOLING	4.2
2	AEROTHERMAL METHODS	• BLUNT BODY FLOW UNDERSTANDING WITH AND WITHOUT JET COUNTERFLOW • BOUNDARY LAYER TRANSITION CRITERIA • NON-EQUILIBRIUM RADIATION	4.2
3	GN&C	• OPTIMAL GUIDANCE APPROACHES • CONTROL FUNCTION DEVELOPMENT • SYSTEM VALIDATION • FLEXIBLE MATERIAL CONTROL TESTS	5.3
4	PROPULSION	• RL10 IIB ENHANCEMENT • ADV EXPANDER ENGINE DEVELOPMENT • ADV ENGINE IMPROVEMENTS	8 TO 15.7
5	ATMOSPHERIC PHYSICS	• TETHER DATA ANALYSIS • LASER RALEIGH BACKSCATTER	2.4
6	AERODYNAMICS	• DETERMINE RAREFIED FLOW EFFECTS • FLEXIBLE BALLUTE DYNAMICS	3.8
•	FLIGHT DEMONSTRATION EXPERIMENT	• INTEGRATED SYSTEMS VERIFICATION • DEMONSTRATE & GN&C ALGORITHMS • PROVIDE AERODYNAMIC/AEROTHERMAL DATA • VERIFY DYNAMIC STABILITY OF FLEXIBLE BALLUTE • VERIFY TPS PERFORMANCE IN ACTUAL FLIGHT ENVIRONMENT	30

Figure 10

Technology Plan Summary

- REASONABLE DEFINITION OF TECHNOLOGY REQUIREMENTS AND OBJECTIVES
- CLEAR DISCRIMINATION BETWEEN REQUIRED VERSUS ENHANCED TECHNOLOGIES
- ENABLING TECHNOLOGY PROGRAM CAN BE ACCOMPLISHED TO SUPPORT PROGRAM START IN LATE 1980's FOR APPROXIMATE TOTAL $ = 65.6 MILLION
- FLIGHT DEMONSTRATION EXPERIMENT(S) EXTREMELY DESIRABLE—POWERFUL BENEFITS—
  - DEMONSTRATES GN&C CONCEPTS AND ALGORITHMS
  - PROVIDES NEEDED AERODYNAMICS/AEROTHERMAL DATA
  - VERIFIES DYNAMIC STABILITY OF FLEXIBLE BALLUTE
  - VERIFIES TPS PERFORMANCE IN ACTUAL FLIGHT ENVIRONMENT
- ENHANCING TECHNOLOGIES APPEAR TO HAVE HIGH PAYOFF (NOT QUANTIFIED IN ALL CASES)
- THIS IS STILL A "FIRST CUT" PLAN AND NEEDS ITERATION

Figure 11

AOTV Thermal Criteria

SURFACE TEMPERATURE – 100 REUSES

TPS MATERIAL	MAXIMUM TEMPERATURE		
	CURRENT TECHNOLOGY	1990 TECHNOLOGY (1995 IOC)	
		NORMAL GROWTH	ACCELERATED GROWTH
FLEXIBLE SURFACE INSULATION (FSI)	1500°F – 1800°F (AFRSI)	2500°F	3000°F
RIGID SURFACE INSULATION (RSI)	2700°F (FRCI)	3000°F	3500°F
HIGH DENSITY REFRACTORY (HDR)	3200°F (ACC)	3500°F	4000°F

BACKWALL TEMPERATURE

MATERIAL	MAXIMUM TEMPERATURE
GRAPHITE/POLYIMIDE *	600°F
KEVLAR CLOTH	600°F

HIGHER TEMPERATURE STRUCTURES ARE POSSIBLE, BUT ARE NOT CONSIDERED ADVANTAGEOUS BECAUSE OF THERMAL CONTROL CONSTRAINTS

Figure 12

Flexible Surface Insulation Technology Assessment

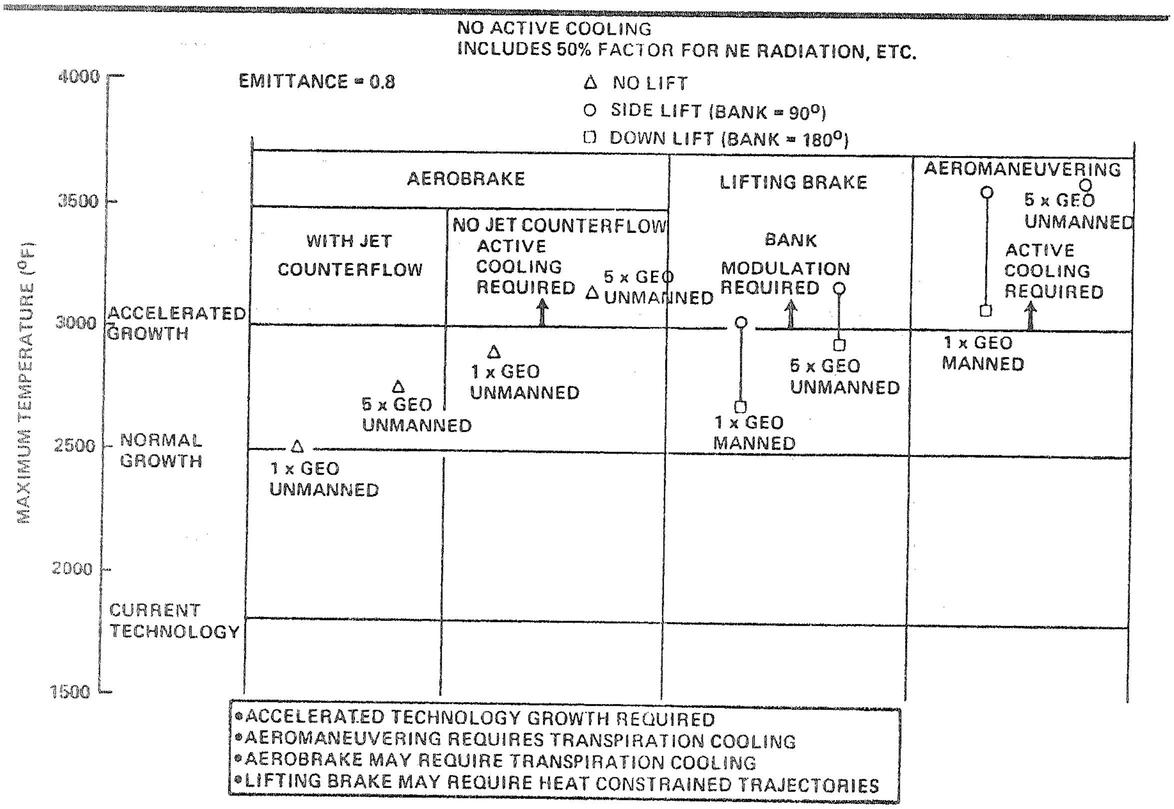


Figure 13

Rigid Surface Insulation Technology Assessment

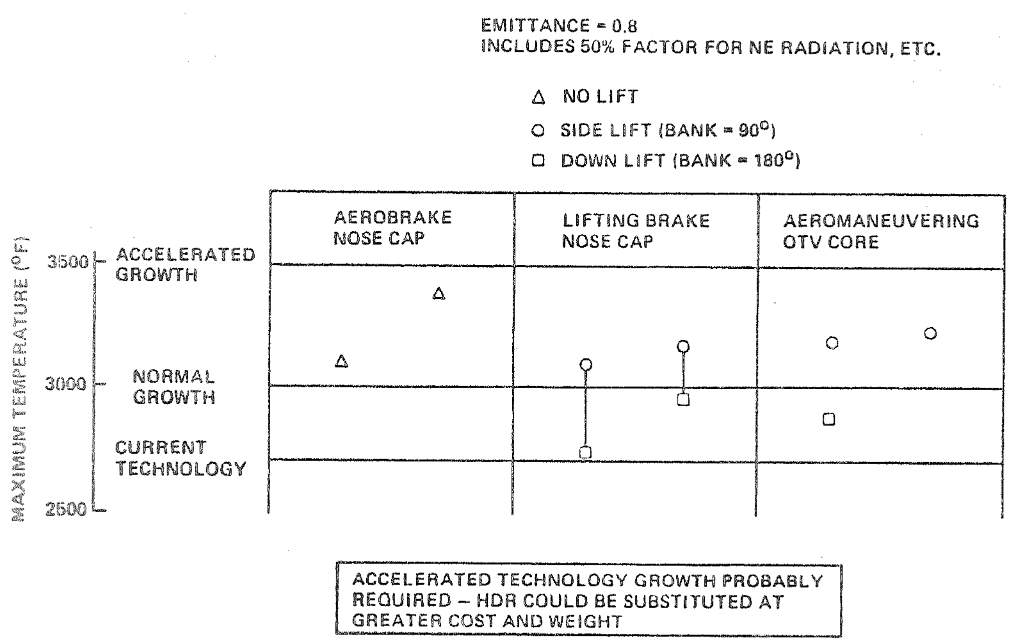


Figure 14

High Density Refractory Technology Assessment

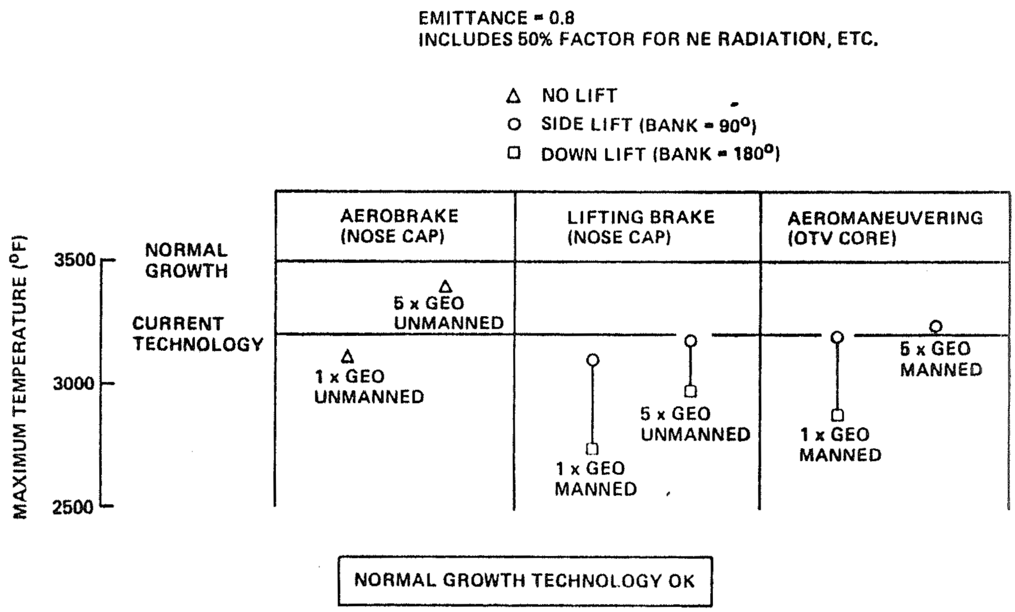


Figure 15

Examples of High Payoff Technology Assessment

SUBSYSTEM	DELTA GEO PAYLOAD	DELTA TECHNOLOGY DEV COST	DELTA DDT&E COST	GEO PAYLOAD COST [1]	RELATIVE SAVINGS [2]
ACCELERATED GROWTH STRUCTURAL COMPONENTS (10% WEIGHT REDUCTION)	280 LB	[3]	$ 35M	10,490 $/LB	210 $/LB
NORMAL GROWTH EXPANDER CYCLE ENGINE (480 SEC ISP)	1,330 LB	$ 8M	$430M	10,013 $/LB	687 $/LB
ACCELERATED GROWTH EXPANDER CYCLE ENGINE (490 SEC ISP)	1,903 LB	$17M	$630M	9,802 $/LB	898 $/LB

ACCELERATED GROWTH TECHNOLOGY COST EFFECTIVE FOR AOTV'S

[1] COSTS BASED ON 6 FLIGHTS/YEAR FOR TEN YEARS (NOM MISSION COST = $81.8M)

[2] BASELINE LCC = $5,632M, BASELINE GEO PAYLOADS = 526,400 LB (10,700 $/LB)

[3] TECHNOLOGY DEVELOPMENT FINANCED BY OTHER PROGRAMS

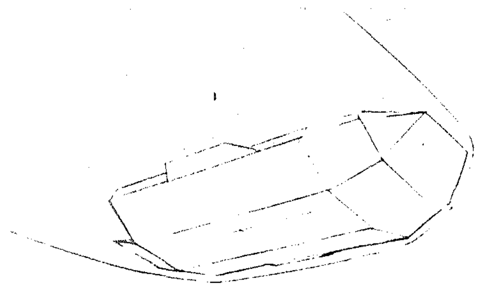
Figure 16

Recommendation

- AEROBRAKED AOTV IS RECOMMENDED CONCEPT IF DEVELOPMENT WERE TO START TODAY
  - BEST PERFORMANCE, LEAST COST, MOST STS COMPATIBLE, ETC.
  - CONTROL METHOD REQUIRES MORE DEVELOPMENT (NEEDS TESTING)
- BEGIN DEVELOPMENT OF TECHNOLOGIES SUITABLE FOR GENERIC AOTV
  - REUSABLE TPS (RIGID AND FABRIC) WITH CAPABILITY TO 3000°F
  - MORE ACCURATE AEROTHERMAL PREDICTION METHODS
  - GN&C SYSTEMS SUITABLE FOR AEROASSIST REENTRY TRAJECTORIES
  - ADVANCED EXPANDER CYCLE ENGINE TECHNOLOGY
- IF SPACE BASING BECOMES PRIMARY OPERATING MODE THEN A SPACE ASSEMBLED LIFTING BRAKE/AEROMANEUVERING CONCEPT SHOULD ALSO BE PURSUED
  - DESIGN FOR COMPLETE REUSABILITY
  - USE L/D TO REDUCE PROPULSIVE ΔV, PEAK HEATING, AND EFFECT OF ATMOSPHERIC DISPERSION

Figure 17

Fresh Look Lifting Brake Designed for Space Assembly



- STS COMPATIBLE OTV MOUNTED USING SHUTTLE FIXTURES
- OTV CAN BE EITHER GROUND BASED OR SPACE BASED
- NO NOZZLE RETRACTION REQUIRED
- GROSS TRIM ACCOMPLISHED BY SLIDING OTV ON RAILS
- CONTROL WITH AERODYNAMIC SURFACES & RCS

- COMBINE BEST FEATURES OF LIFTING BRAKE & AMOTV TO INCREASE L/D AND REDUCE SCAR WEIGHT
- SPACE ASSEMBLED PREFABRICATED COMPOSITE PANELS
- RIGID OR FABRIC REUSABLE TPS
- LARGE PLANFORM AREA REDUCES TEMPERATURES
- NO IMPINGEMENT PROBLEM

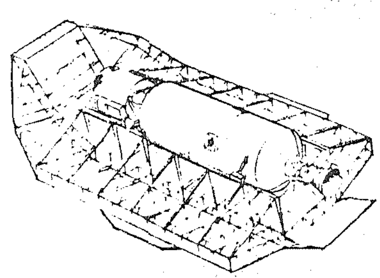


Figure 18

Objectives of Follow-on Study

- DESIGN DEVELOPMENT OF SPACE ASSEMBLED LIFTING BRAKE CONCEPT
- FURTHER APPLICATION OF OPTIC GUIDANCE CONCEPT TO GN&C TRADES
  - AEROBRAKE CONCEPT
  - LIFTING BRAKE CONCEPT
- PROPULSION SYSTEM TRADES
  - SIZE AND NUMBER OF ENGINES OPTIMUM FOR AOTVs
  - TECHNOLOGY LEVELS OPTIMUM FOR LCC
- TECHNOLOGY DEVELOPMENT PLANNING
  - ATMOSPHERIC DISPERSION TESTING
  - TPS .
  - AEROTHERMAL
  - AERODYNAMICS

Figure 19

MODERATE LIFT-TO-DRAG AEROASSIST

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Numerous potential technology advances have been identified and evaluated that provide significant mission enabling and mission enhancing features to a wide variety of mid L/D AOTVs. In this paper, those advances associated with propulsion subsystems will be highlighted.

INTRODUCTION

Significant performance benefits can be realized via aerodynamic braking and/or aerodynamic maneuvering on return from higher altitude orbits to low Earth orbit, Reference 1-5. This approach substantially reduces the mission propellant requirements by using the aerodynamic drag, D, to brake the vehicle to near circular velocity and the aerodynamic lift, L, to null out accumulated errors as well as change the orbital inclination to that required for rendezous with the Space Shuttle Orbiter. A study has been completed where broad concept evaluations were performed and the technology requirements and sensitivities for aeroassisted OTV's over a range of vehicle hypersonic L/D from 0.75 to 1.5 were systematically identified and assessed. The aeroassisted OTV is capable of evolving from an initial delivery only system to one eventually capable of supporting manned roundtrip missions to geosynchronous orbit. Concept screening has been conducted on numerous configurations spanning the L/D = 0.75 to 1.5 range, and several with attractive features have been identified.

Initial payload capability has been evaluated for a baseline of delivery to GEO, six hour polar, and Molniya (12 hours x 63.4°) orbits with return and recovery of the AOTV at LEO. Evolutionary payload requirements that have been assessed include a GEO servicing mission (6K up and 2K return) and a manned GEO mission (14K roundtrip).

AOTV Performance

Previous studies, References 3 and 4, have considered only missions from LEO to Geosynchronous orbit and return. In this study, missions were defined to higher inclination orbits, where an aeromaneuvering vehicle was expected to become more attractive due to its ability to provide orbital plane change.

Performance studies have been conducted for return of mid L/D vehicles from GEO, 5 x GEO, and 6-hour Polar circular orbits. Steering laws have been employed that include constant deceleration cruise at the overshoot and undershoot bounds, and constant bank angle cruise. Orbital plane change obtained is summarized in

Figure 1, where it is shown that plane change capability increases with hypersonic L/D and entry velocity (maximum for the 5 x GEO return) for a specific steering law. The 90° bank angle provides the maximum plane change.

The insensitivity of an L/D = 1.5 AOTV to variations from the nominal in the atmosphere density or to errors in the apriori estimate of the drag coefficient have been evaluated by personnel from NASA JSC and are illustrated in Figure 2.

Configuration Development

Several classes of configurations exist that meet the hypersonic performance requirements. Thse include axisymmetric and elliptical cross section cones, biconics, cone cylinders and arbitrary bodies. Generally, the sphere cones are too long to meet the length constraint and package the required propellant tanks and payloads. Arbitrary bodies are generally geometrically more complex than necessary for this aeromaneuver vehicle and exhibit poor propellant tank packaging efficiency.

Biconic and cone cylinders were selected for this study because they were the best compromise on L/D and packaging efficiency; there is a large aerodynamic and design data base; the basic maneuvering concept has been flight proven for this class of vehicles. This concept was thoroughly evaluated for the planetary aerocapture mission and presents a feasible, well characterized, solution.

The aerodynamic configuration selected must: 1) meet the external dimensional constraints of the launch vehicle, and 2) provide packaging room for the propellant tanks and other subsystems so that the launch configuration with tanks full meets the launch vehicle center-of-mass requirement and the entry configuration with tanks empty meets the center-of-mass requirement to trim the vehicle at the desired angle of attack during the aeromaneuver. The desired angle of attack is obtained by placing the entry center-of-mass at the AOTV center-of-pressure location for that angle-of-attack. The selected angle of attack for the baseline vehicles will be that for which L/D is a maximum, thus insuring maximum plane change capability for the vehicle.

The aerodynamic configurations of mid L/D AOTV's evolved from review of an existing computational aerodynamic data base supplemented with additional calculations. The initial data base consisted of existing flow field calculations for aft frustum angles down to 4° and the AMOOS results for frustum angles of 0 and 1/2°. This data base was supplemented with new HABP, Reference 8, calculations for a frustum angle of 2°.

The effect of increased nose length or increased vehicle length on increasing the vehicle hypersonic L/D is illustrated in Figure 3. Note the large effect that increased nose length makes.

For packaging or aerodynamic reasons, a full nose bend, δ_n, may not be desirable. The effect of lesser nose bend on $(L/D)_{max}$ is also illustrated in Figure 3.

Several major configuration classes are possible by employing different staging techniques. Single stage vehicles were evaluated recently, References 1, 3 and 4, where the propellant tanks are enclosed within the AOTV and the entire vehicle makes the round trip. Stage and-a-half vehicles, AMOS, Reference 6, 9, MOTV, Reference 7, have been evaluated and were shown to offer payload delivery and cost advantages over the single stage vehicles. Two-stage vehicles have been evaluated and shown to offer payload delivery advantages. Specific configurations employing each of the above staging techniques have been evaluated.

For the single stage vehicles, propulsion stage packaging trends have been evaluated to determine vehicle center of mass possibilities for combinations of total vehicle length, Lv, and nose length, Ln. Two propulsion stages were used; one representing an extremely short stage, (utilizes torroidal oxygen tank) and one representing probably the longest stage possible (spherical tanks). Using these results, in combination with the parametric center of pressure locations, three configurations were defined, Figure 4, that span the range of L/D from 0.75 to 1.5 for further evaluation.

MAJOR FACTORS FOR IMPROVING MID L/D PAYLOAD DELIVERY PERFORMANCE

The performance capability of a mid L/D AOTV can now be enhanced considerably by combining many of the effects that incrementally improve performance of the AOTV into one vehicle. The improvements can be categorized into: 1) those that fall within current state-of-the-art, and 2) those that result from improvements in state-of-the-art, and are summarized in Figure 5.

Considering all of these effects, a representative ideal Geosynchronous delivery vehicle was defined for evaluation, Figure 6.

PROPULSION SUBSYSTEM TECHNOLOGY ADVANCES

As part of the Advanced OTV Propulsion System Program currently underway, improvements in specific impulse for LOX-H_2 fueled engines are projected to reach 480 to 490 seconds, References 10, 11 and 12. The potential improvement in AOTV payload delivery capability is illustrated for GEO and Polar delivery in Figure 7. Note that the payoff for increased specific impulse is about 60-65 pounds of payload for each second of specific impulse improvement.

The advantage of variable mixture ratio (MR) operation to maximize the specific impulse of a throttable engine was identified, Reference 10. In addition, increase of the mixture ratio reduces the size of the hydrogen tank by one foot for the 65K STS and 1.8 feet for the 100K STS at only a small loss of payload delivery capability.

The wide range of engine size and thrust level possibilities have been identified, Reference 10. The packaging advantages and the shorter (hence lighter) vehicles that result from use of multiple small engines have been evaluated. One to six engines, providing a total thrust of 15,000 lbs, and man-rating requirements have been considered. The results of this AOTV-engine weight trade are summarized in Figure 8 where it is seen that for a representative Mid L/D AOTV, six engines result in nearly a 5 foot shorter and 260 lbs lighter vehicle.

Some of the AOTV configuration-engine location interactions that were found are summarized in Figure 9.

SEVERAL ATTRACTIVE MID L/D AOTVs

Examples of several configuration classes were evaluated including both single and multiple stage vehicles, unmanned delivery and manned vehicles. Examples of these configurations employing some growth technology are illustrated in Figures 10 and 11 and their primary features enumerated.

Flight performance and payload delivery sensitivities across the mid L/D range for a single stage AOTV are summarized in Figure 12. The incremental increase in payload delivery capability, given a reduction in vehicle dry weight, or an increase in vehicle L/D is illustrated for vehicles at both ends of the mid L/D range. The incremental loss of payload delivery capability is illustrated for each degree of plane change generated propulsively in the initial mission orbit. Note the large differences in the effect of incremental L/D on payload delivery capability, Δ W P/L/ Δ L/D, between the GEO and 6 hr polar delivery missions.

ADVANCED TECHNOLOGY PAYOFFS

A detailed review of the current state-of-the-art in the various technology and subsystems areas was conducted to serve as a baseline point of departure for this study. Technology advancement possibilities identified in numerous recent studies of OTV, AOTV, SDV, and STS were reviewed. These results are compared with our in-house data base and parameters selected that represent improvements due to nominal expected growth resulting from normal funding of these technology areas. A number of these improvements resulting in from 10 to 70% reduction of subsystem weight are summarized in Figure 13. Other improvements include such items as increase of maximum operating temperature of the thermal protection system elements and increased confidence in the hypersonic aerodynamic characteristics.

Various techniques exist for ranking the technology benefits. The method selected for this study is as follows: given a subsystem weight reduction or other performance improvement possibility, the effect on increased payload weight was determined and this payload gain was converted to a customer cost benefit, given a nominal delivery cost to GEO of $8000 per lb. The mid L/D AOTV payload delivery sensitivities of Figure 12 have been combined with the delivery cost and the subsystem weight reduction possibilities to generate the results summarized in Figure 14 for the 38 ft and OH-3 delivery vehicles. Note that the 38 ft single stage vehicle has very different technology payoffs from the small OH-3 staged vehicle.

Additional technology advance benefits are summarized in Figure 15 for both vehicles. Aerodynamic uncertainties due to viscous and rarefaction effects will exist and could amount to as much as ± 0.1 of Δ L/D. This uncertainty requires a propellant contingency which in turn decreases the payload delivery capability. Flight vehicles have typically flown initially with a safety margin in the thermal protection system of as much as 25%. This translates into a very large payload loss (and hence cost benefit if it is decreased or eliminated) for the 38 ft delivery vehicle but a much smaller effect for the OH-3 vehicle due to its much smaller size. In the GN&C subsystem area, the ability to obtain aerodynamic plane change is translated into payload gain and hence customer cost benefit. The value of an "optimum" guidance system that has been selected because it is capable of obtaining the most aerodynamic plane change from a given vehicle configuration is illustrated for one degree of incremental plane change. The value of an "adaptive" guidance system that has the capability of updating during the early portion of entry is illustrated for each additional one degree of plane change that can be generated. The effect of encountering a 30% density shear (pocket) similar to that experienced by a recent STS flight has been demonstrated to have no effect on vehicle with L/D = 1.5 but to have a small effect on a vehicle with L/D = 0.6.

CONCLUDING REMARKS

The major conclusions of this study include the following:

- Use of mid L/D AOTV provides significant aerodynamic plane change capability and control authority over trajectory dispersions and off nominal atmospheres.

- All mid L/D AOTV enabling technology is ready today.
- Substantial performance improvements and hence cost benefit can be obtained by developing enhancing technologies.
- Six fixed, low thrust ($\approx$2000 to 3000 lb), advanced expander, LOX-hydrogen engines operating at a MR $>$6.0 offer attractive packaging possibilities.
- Manned mission to GEO with delivery of one ton payload is possible with the 65K STS, mid L/D AOTV, an advanced cryofueled engine and lightweight ASE (3000 lbs).
- Delivery of very long payloads (45 ft) is possible by use of very short AOTVs with drop tank.

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AOTV PLANE CHANGE CAPABILITY

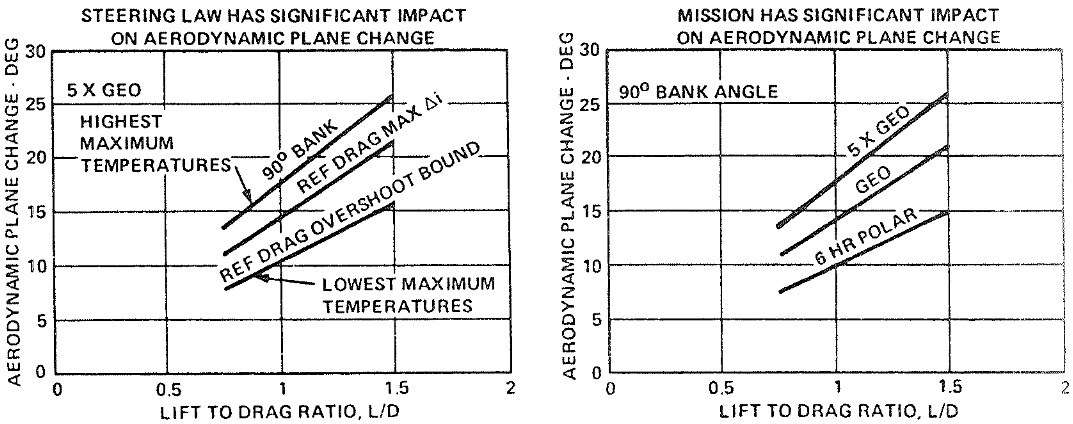


FIGURE 1

MID L/D AOTV IS RELATIVELY INSENSITIVE TO ATMOSPHERIC DENSITY AND DRAG COEFFICIENT UNCERTAINTIES

L/D = 1.5 W/C_D S = 97 PSF

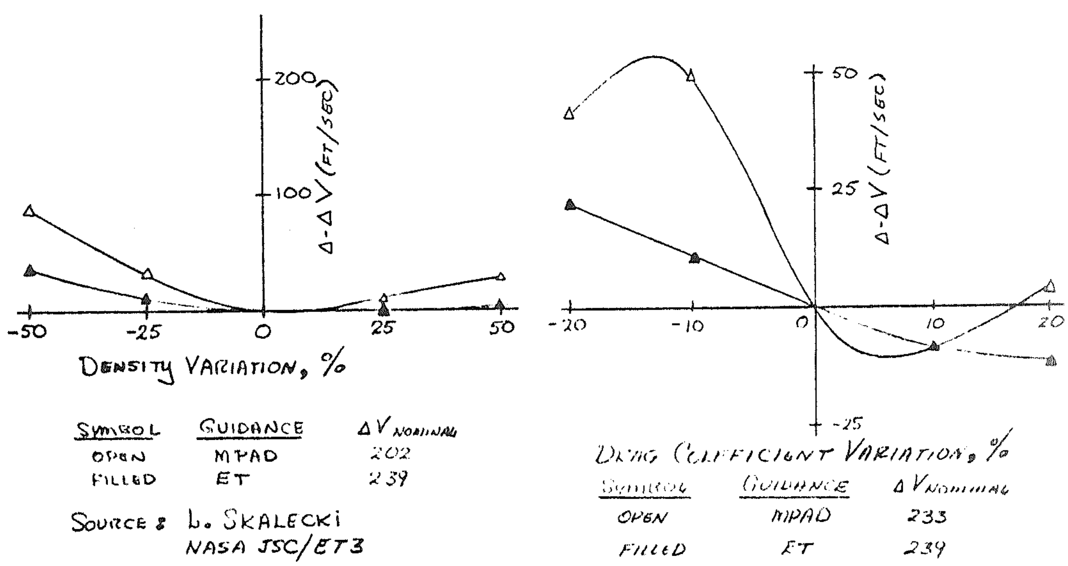
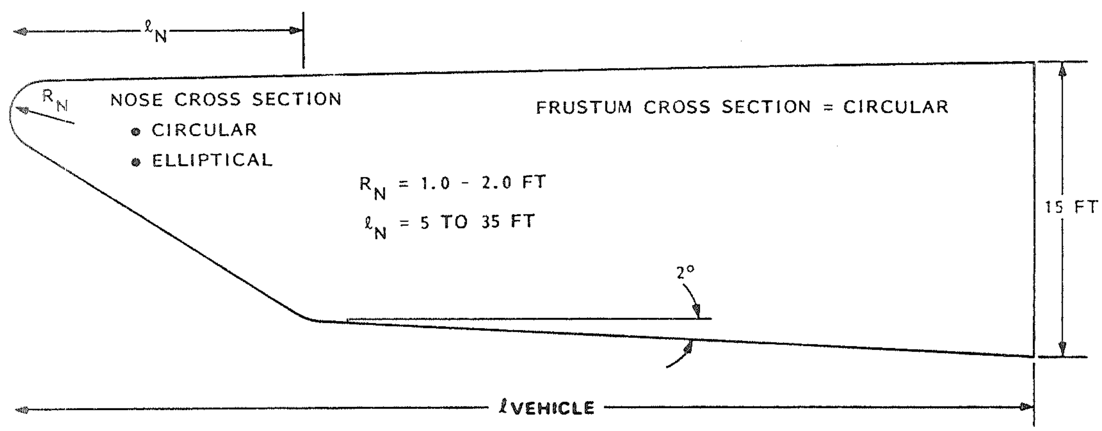


FIGURE 2

EFFECT OF NOSE BEND ON MAXIMUM L/D



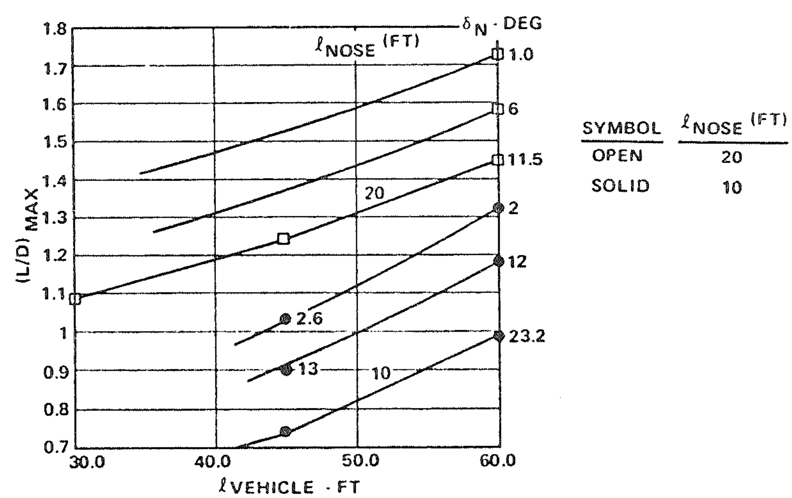


FIGURE 3

AOTV CONFIGURATIONS SELECTED FOR FURTHER SENSITIVITY STUDIES

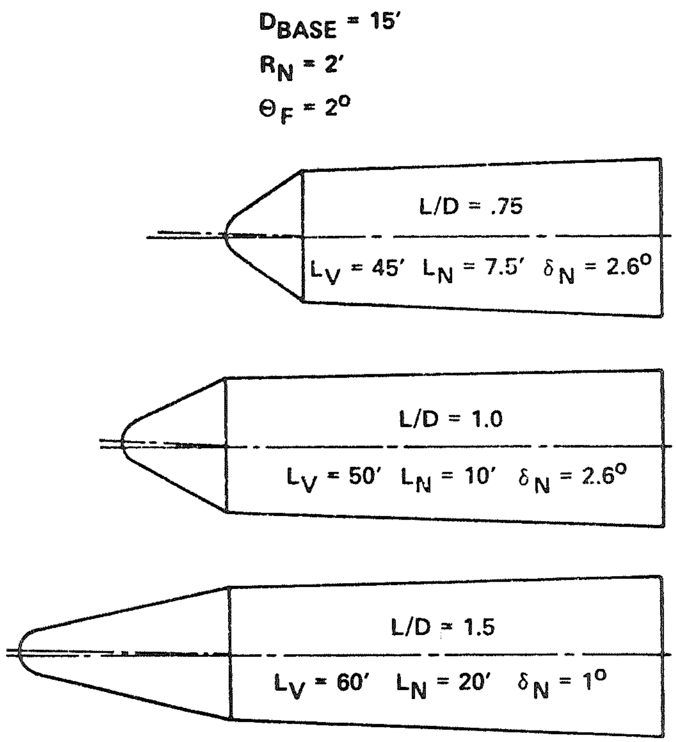


FIGURE 4

MAJOR FACTORS FOR IMPROVING MID L/D PAYLOAD DELIVERY PERFORMANCE

WITHIN STATE OF ART

REDUCE AOTV DRY WEIGHT
- SHORTEN VEHICLE
- COLD SOAK TPS PRIOR TO ENTRY

INCREASE L/D
- LENGTHEN NOSE
- STEEPEN FRUSTUM CONE ANGLE (BETTER X_{CP})
- DECREASE NOSE BEND ANGLE
- DECREASE NOSE RADIUS (HEATING LIMITATIONS?)

} LOSE PACKAGING VOLUME

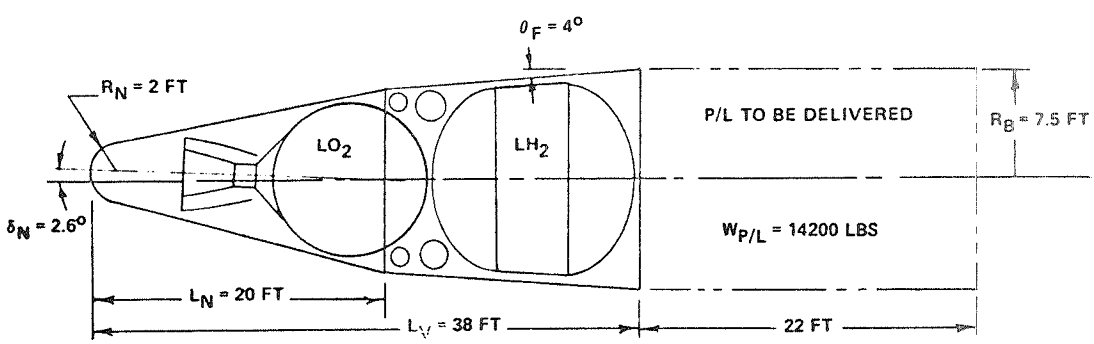
IMPROVEMENTS IN STATE OF ART

REDUCE SIZE OF PROPULSION CORE
- INCREASE I_{SP}
- INCREASE MR - REDUCES LH_2 TANK SIZE
- INCORPORATE MULTIPLE SMALL ENGINES

REDUCE AOTV DRY WEIGHT
- STRUCTURAL SHELL, FRAMES AND SUPPORT
- FLAPS
- AVIONICS
- EPS
- PROPELLANT TANKS, TPS, ACS AND PROPULSION

FIGURE 5

A 38 FT GEO DELIVERY VEHICLE



L/D = 1.5 INV $\quad X_{CM}/L_V = 0.52$

$W_P = 45$ K $\quad$ MR = 7

FIGURE 6

INCREASED SPECIFIC IMPULSE PROVIDES
MAJOR AOTV PERFORMANCE PAYOFFS FOR
BOTH GEO AND POLAR MISSIONS

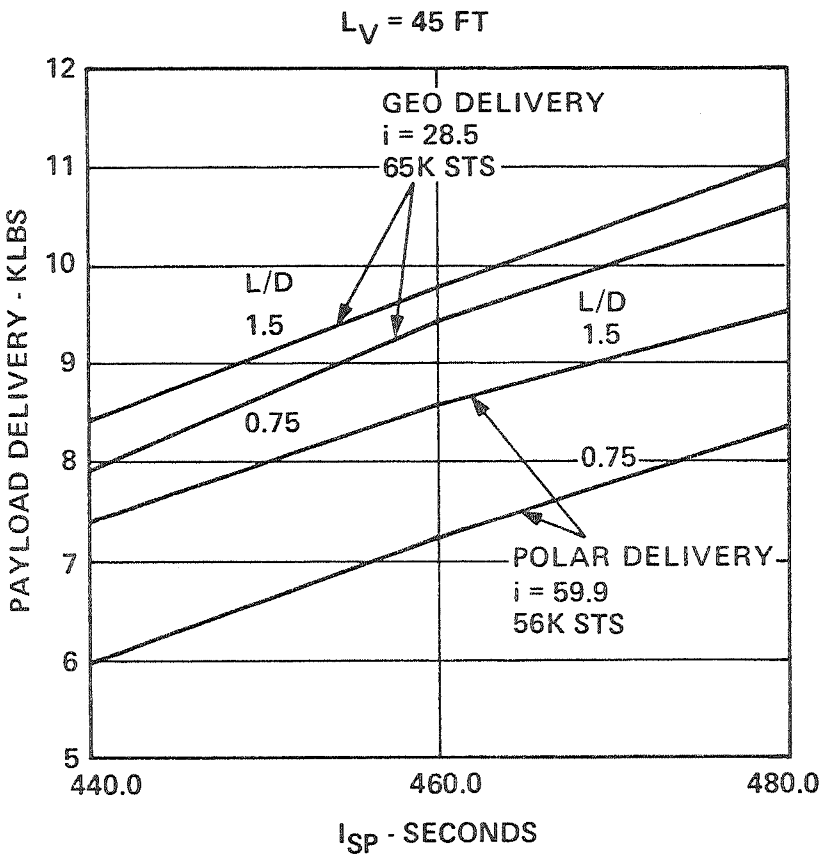


FIGURE 7

NUMBER OF ENGINES vs AOTV WEIGHT (MAN RATED)

- REPRESENTATIVE LARGE AOTV (e.g., H-1M)
  - 14.5' ϕ AT AFT END
  - AEROSHELL (TPS + STRUCTURE) WEIGHT ≈ 80 LB/FT OF LENGTH
- ADJUST PROPULSION SYSTEM TRADE FOR RETRACTABLE NOZZLES
  - ADD ≈ 10 LB/ENG FOR NOZZLE EXTENSION
- INCORPORATE RESULTS OF ENGINE/VEHICLE LENGTH TRADE
  - 15° GIMBAL ANGLE FOR 1 → 5 ENGS
  - MAXIMIZE ENG RADIAL LOCATION WITHIN AOTV
  - WITH ENG ℄ PARALLEL TO VEHICLE ℄, NOZZLE EXIT PLANE DEFINES END OF AEROSHELL

	GIMBALED					FIXED
NUMBER OF ENGINES	1	2	3	4	5	6
Δ VEHICLE LENGTH (FT)	0	- 0.25	- 2.25	- 3.17	- 4.83	- 4.92
Δ VEHICLE WEIGHT (LB)	0	-20	-180	-253	-387	-393
Δ PROPULSION SYS WT (LB)	0	+14	+ 1	+ 93	+185	+134
NOZZLE RETRACT ADJ (LB)	0	+24	+ 33	+ 40	+ 50	0
Σ = AOTV Δ WT (LB)	0	+18	-146	-120	-152	-259

MIN AOTV WEIGHT WITH SIX ENGINES

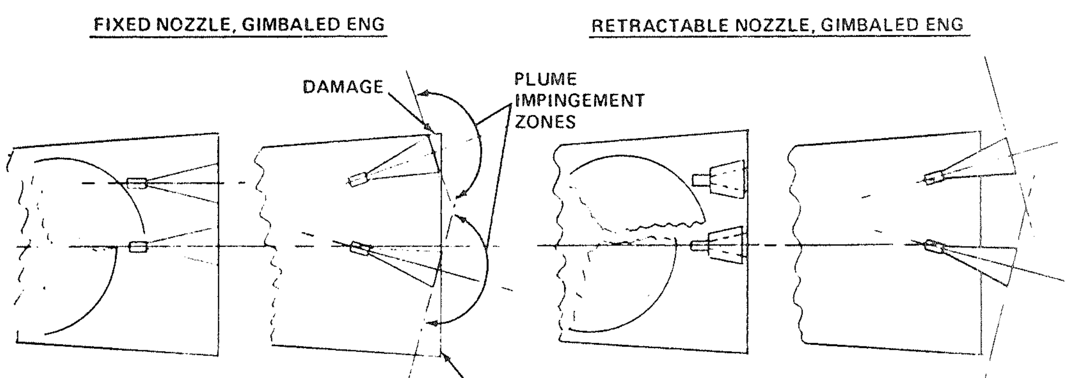
PREFERRED

FIGURE 8

SOME BI-CONIC AFT END & ENGINE INTERACTIONS

- CURRENT AOTV GROUNDRULE: "ALL REUSEABLE AOTV COMPONENTS MUST BE PROTECTED BY AEROSHELL"

FIXED NOZZLE, FIXED ENGINE
- REQUIRES MULTIPLE ENGINES ⇒ "LOW THRUST" PER ENGINES ⇒ SHORT ENGINES
- SMALL ENGINES FIT INTO "CORNERS & HOLES"
  - SHORT AOTVs RESULT



- APPEARS TO BE UNACCEPTABLE

- SHORTER VEHICLES
- NO PLUME IMPINGEMENT DAMAGE

FIGURE 9

SMALL MANNED AOTV "H-1M'

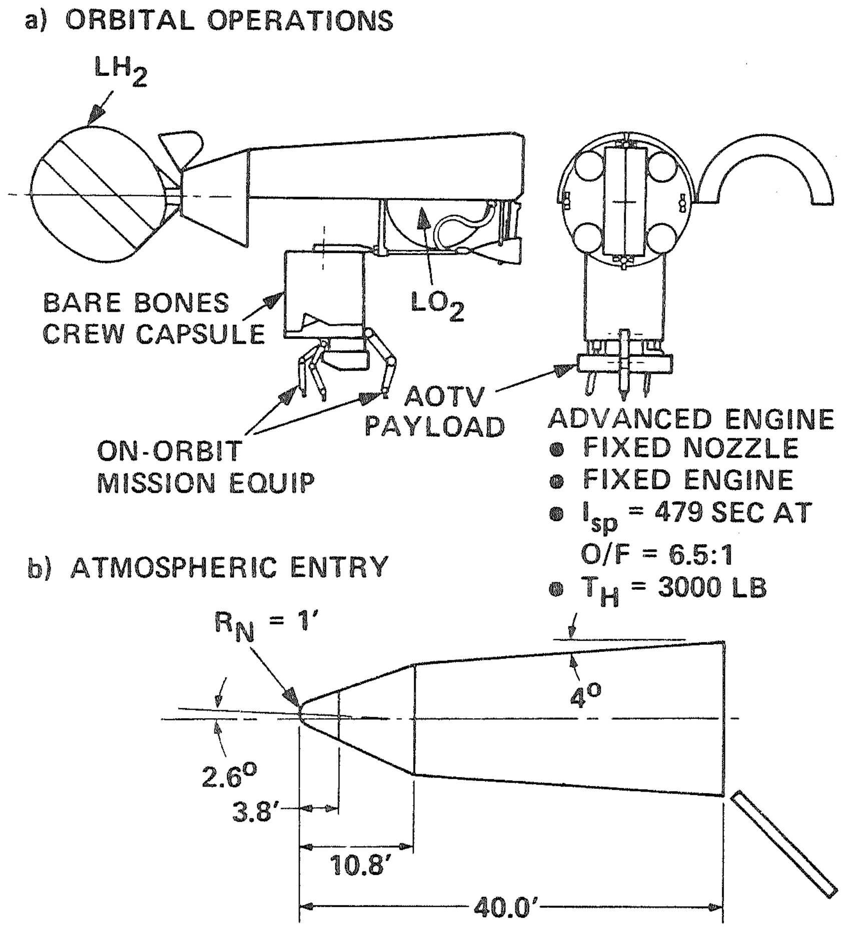
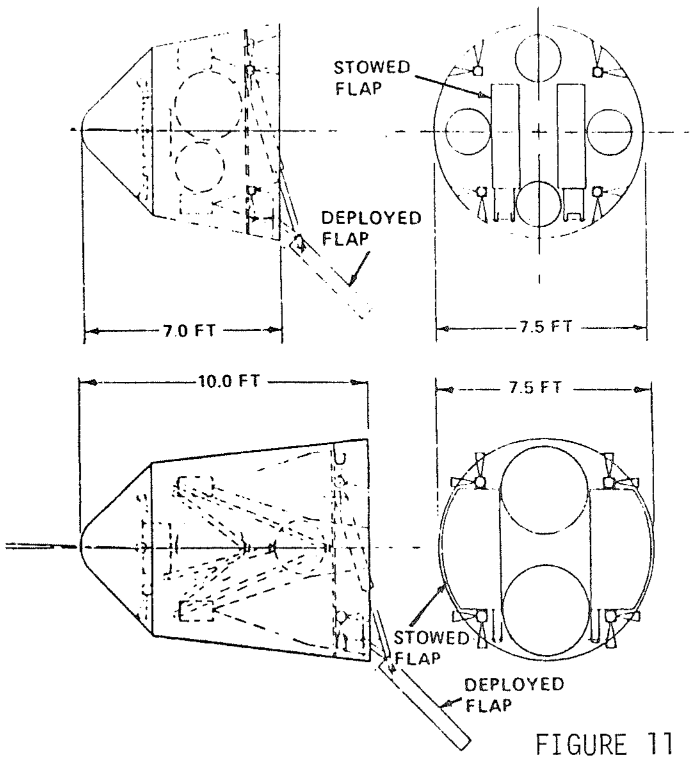


FIGURE 10

PERFORMANCE COMPARISON OF OH-3 & OH-1



OH-3

- FOUR ADVANCED EXPANDER ENGINES
  ϵ = 400: 1
  I_{SP} = 475 SEC
- TOTAL THRUST = 12,000
- EFFICIENT FOR OFFSET C.G.
- POSSIBLE ENGINE OUT CAPABILITY
- 4000 LB DRY WEIGHT
- PAYLOAD DELIVERY TO GEO TRANSFER ORBIT WITH 11,930 LB OF PROP: W_{PAY} – 11,400 LB

OH-1

- TWO ADVANCED EXPANDER ENGINES
  ϵ = 1000: 1
  I_{SP} = 480 SEC
- TOTAL THRUST = 6000 LB
- 4000 LB DRY WEIGHT
- PAYLOAD DELIVERY TO GEO TRANSFER ORBIT WITH 11,930 LB OF PROPELLANT: W_{PAY} = 11,400 LB

FIGURE 11

SUMMARY OF PAYLOAD DELIVERY SENSITIVITIES FOR A SINGLE STAGE AOTV-65K STS

PARAMETER		MISSION	P/L SENSITIVITIES	
			L/D = 0.75	1.5
AOTV DRY WEIGHT	$\frac{\Delta W_{P/L}}{\Delta WT_{DRY}}$ (LB/LB)	GEO DELY	-1.65	-1.65
		6 HR POLAR	-1.7	-1.5
ENGINE I_{SP}	$\frac{\Delta W_{P/L}}{\Delta I_{SP}}$ (LB/SEC)	GEO DELY	64	64
LIFT-DRAG RATIO	$\frac{\Delta W_{P/L}}{\Delta L/D}$ (LB)	GEO DELY	430	430
		6 HR POLAR	2000	1700
		GEO MANNED RT	800	800
PROPULSIVE PLANE CHANGE AT MISSION ALTITUDE	$\frac{\Delta W_{P/L}}{\Delta i_{PROP}}$ (LB/o)	GEO DELY	-34	-34
		6 HR POLAR	-183	-183

FIGURE 12

TECHNOLOGY ADVANCEMENT POTENTIAL

AOTV SUBSYSTEM ELEMENT	EXPECTED IMPROVEMENT
STRUCTURE (SHELL, FRAMES, SUPPORTS & FLAPS)	10 TO 30% WEIGHT REDUCTION
THERMAL PROTECTION SYSTEM	UP TO 69% WEIGHT REDUCTION
TRANSPIRATION COOLED NOSE	7° PLANE CHANGE INCREASE FOR 5 X GEO RETURN
AVIONICS	50 TO 70% WEIGHT REDUCTION
ELECTRICAL POWER SUPPLY	20 TO 38% WEIGHT REDUCTION
NEW CRYOFUELED ENGINE	Isp UP TO 480 SEC

FIGURE 13

EFFECT OF TECHNOLOGY ADVANCES ON CUSTOMER COST BENEFIT

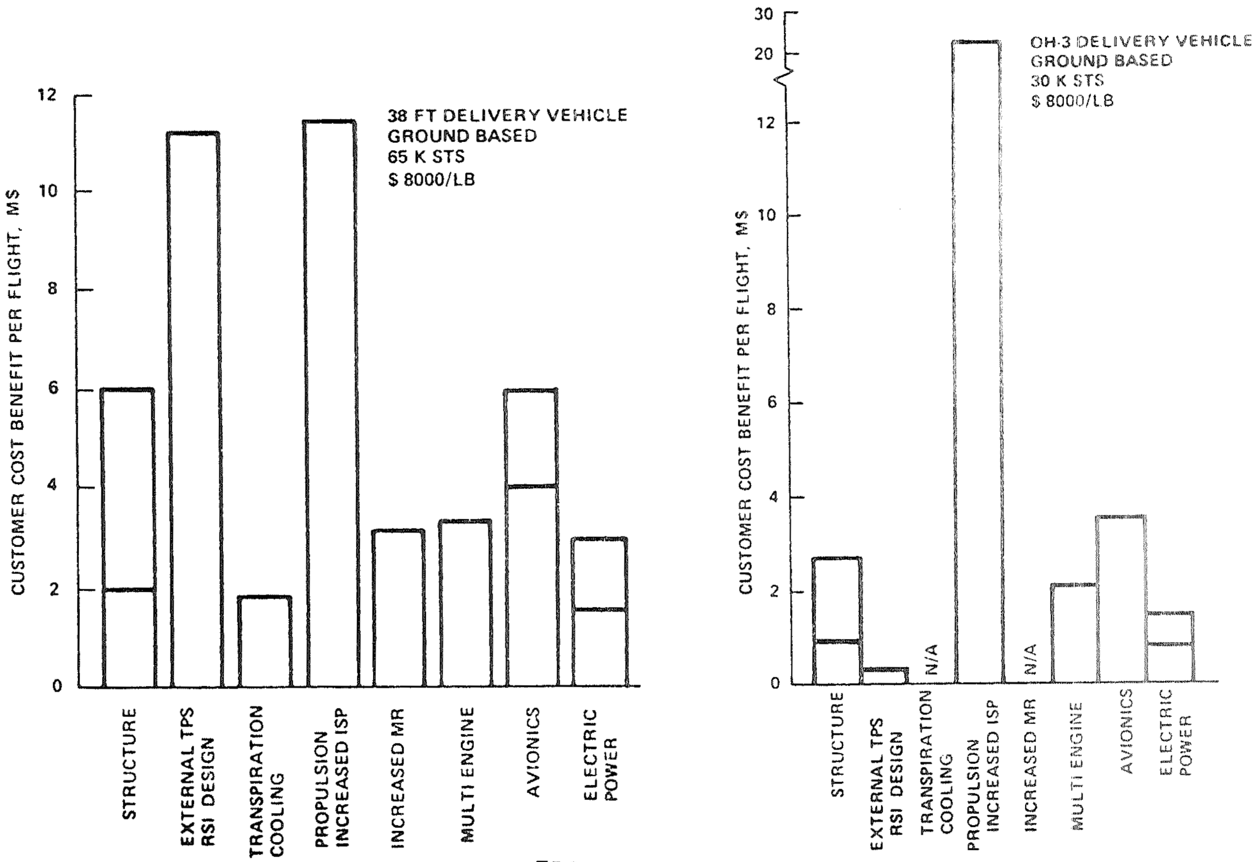


FIGURE 14

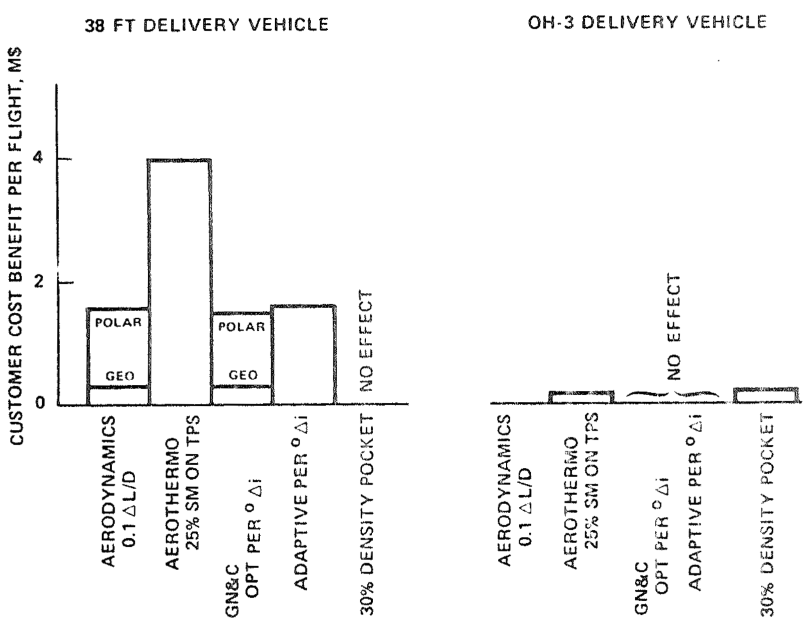
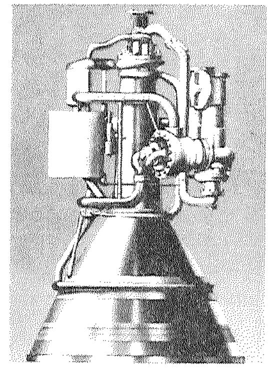


FIGURE 15

OTV PROPULSION SYSTEM CHALLENGES

GOALS



VACUUM SPECIFIC IMPULSE lbf-sec/lbm	520
VACUUM THROTTLE RATIO	30:1
NET POSITIVE SUCTION HEAD, lbf-ft/lbm	0
WEIGHT, lbm	360
LENGTH (STOWED), INCH	40
RELIABILITY	1.0
SERVICE LIFE	
BETWEEN OVERHAULS, CYCLES/hr	500/20
SERVICE FREE, CYCLES/hr	100/4

REQUIREMENTS

PROPELLANTS	HYDROGEN/OXYGEN
TOTAL VACUUM THRUST, lbf	10,000 - 25, 000
ENGINE MIXTURE RATIO	6 ± 1

CD-83-13814

FIGURE 16

N85-16996

D7

BENEFITS OF HIGH AERODYNAMIC EFFICIENCY TO ORBITAL TRANSFER VEHICLES*

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The Boeing Company

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S. W. Paris
The Boeing Company

An analysis of the benefits and costs of high aerodynamic efficiency on aeroassisted orbital transfer vehicles (AOTV) is presented. These results show that a high lift-to-drag (L/D) AOTV can achieve significant velocity savings relative to low L/D aerobraked OTV's when traveling round trip between low Earth orbits (LEO) and alternate orbits as high as geosynchronous Earth orbit (GEO). Trajectory analysis is used to show the impact of thermal protection system technology and the importance of lift loading coefficient on vehicle performance. The possible improvements in AOTV subsystem technologies are assessed and their impact on vehicle inert weight and performance noted. Finally, the performance of high L/D AOTV concepts is compared with the performances of low L/D aeroassisted and all-propulsive OTV concepts to assess the benefits of aerodynamic efficiency on this class of vehicle.

*Work supported by U.S. Air Force Wright Aeronautical Laboratories and Boeing Aerospace Company.

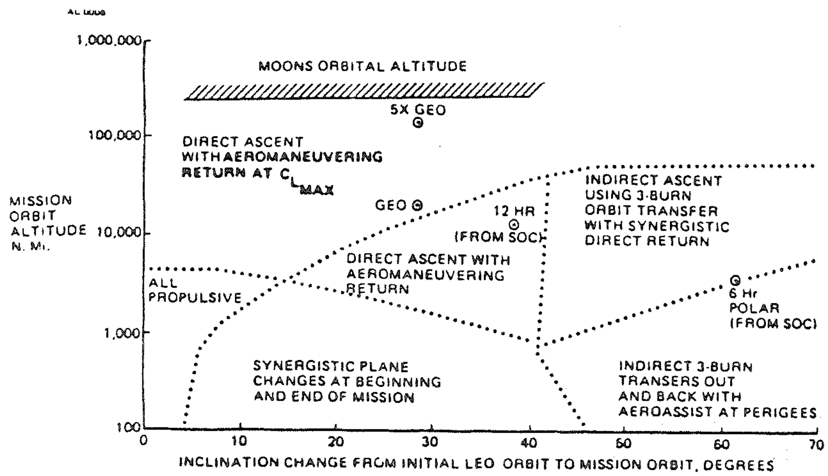

Figure 1. Synthesis of Optimum AOTV Mission Scenarios

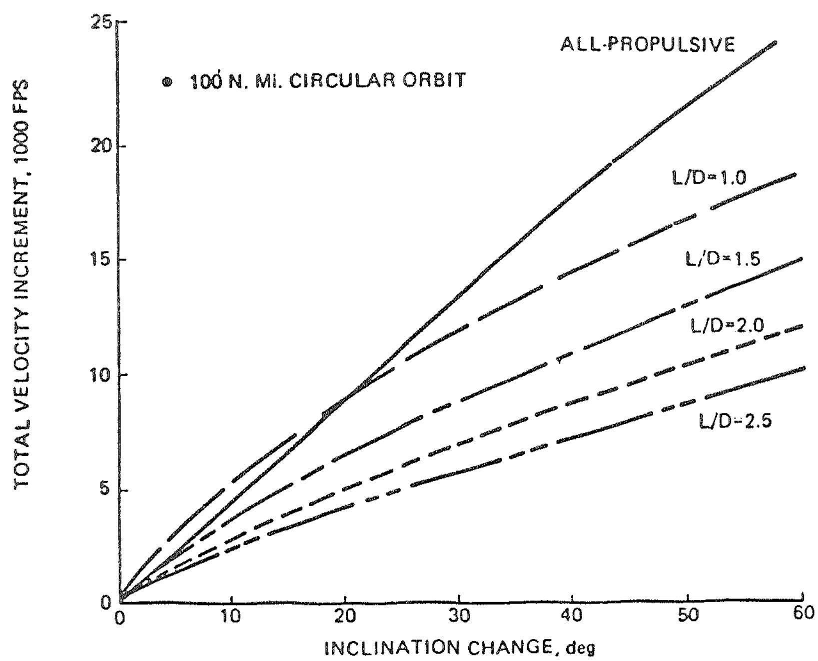

Figure 2. Effect of Vehicle L/D on Plane-Change Capability

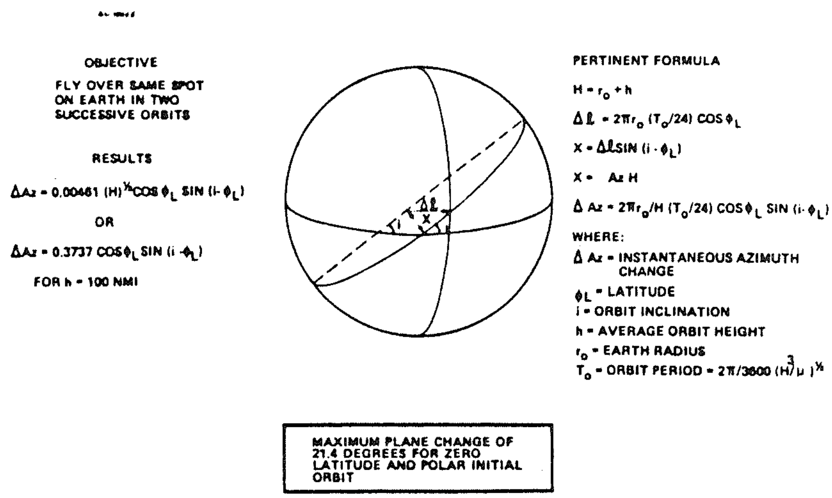

Figure 3. Synergistic Plane-Change Maneuver for Ground-Based AOTV

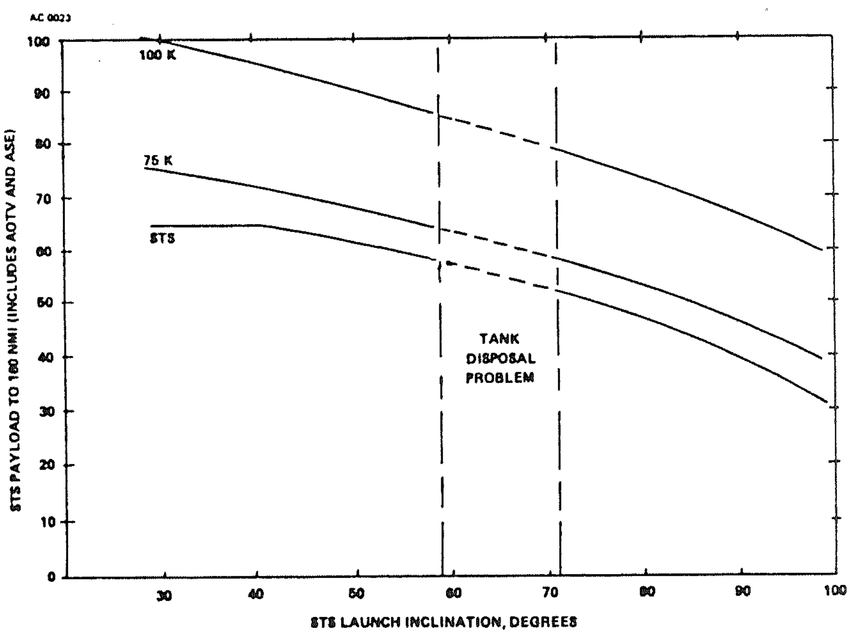

Figure 4. Ground-Based AOTV Insertion Weight

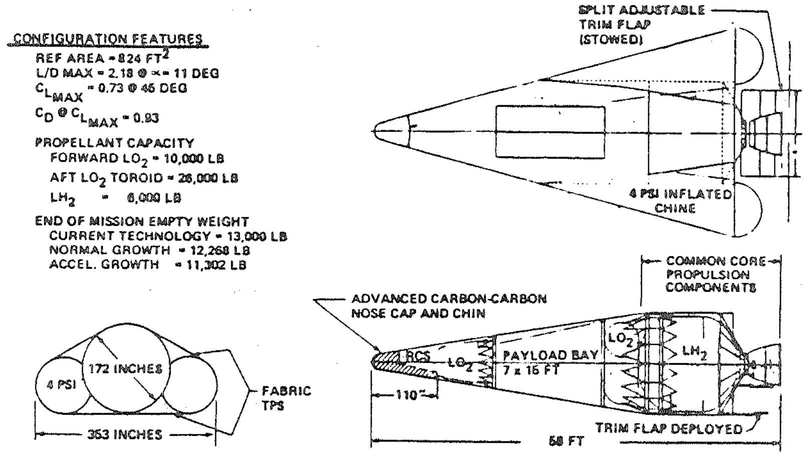

Figure 5. Ground-Based AOTV Configuration

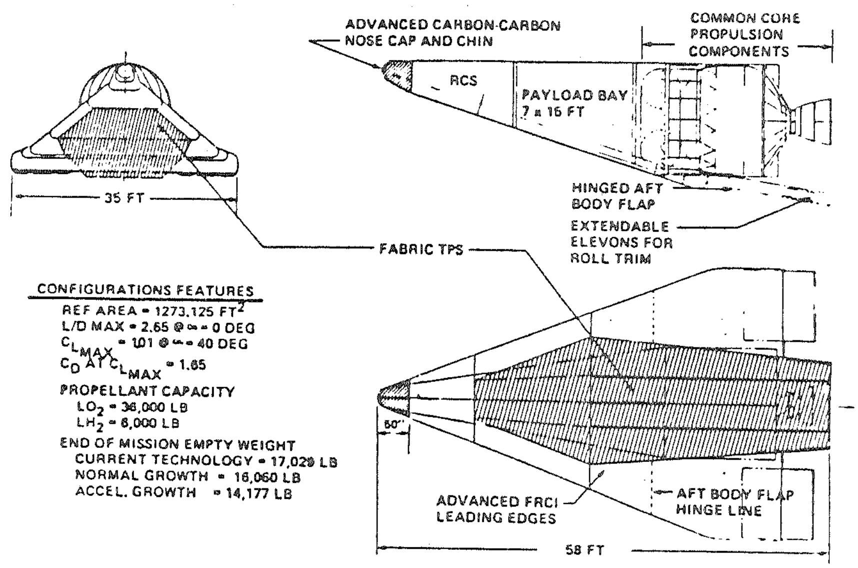

Figure 6. Space-Based AOTV Configuration

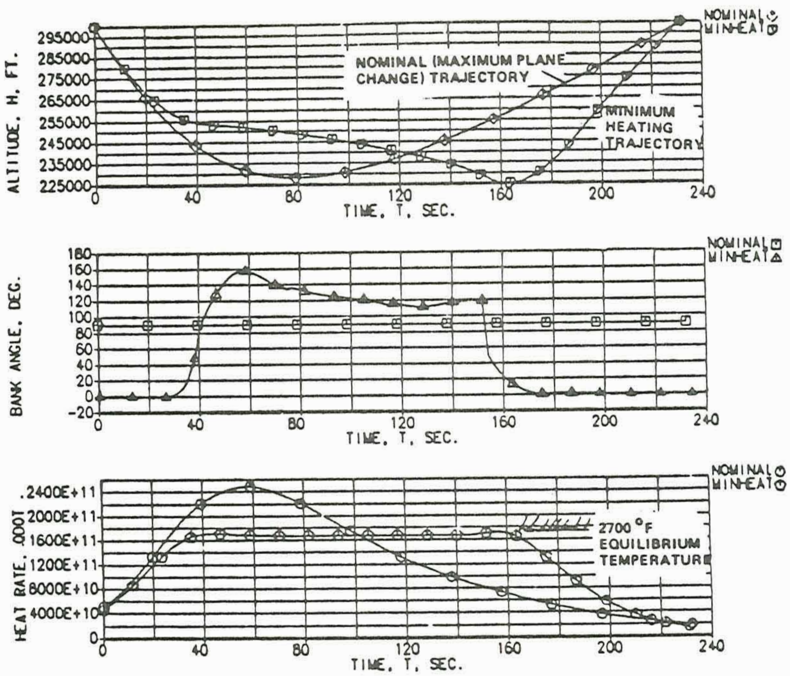

Figure 7. Inflatable Chine AOTV 5xGEO Reentry Trajectory Comparison

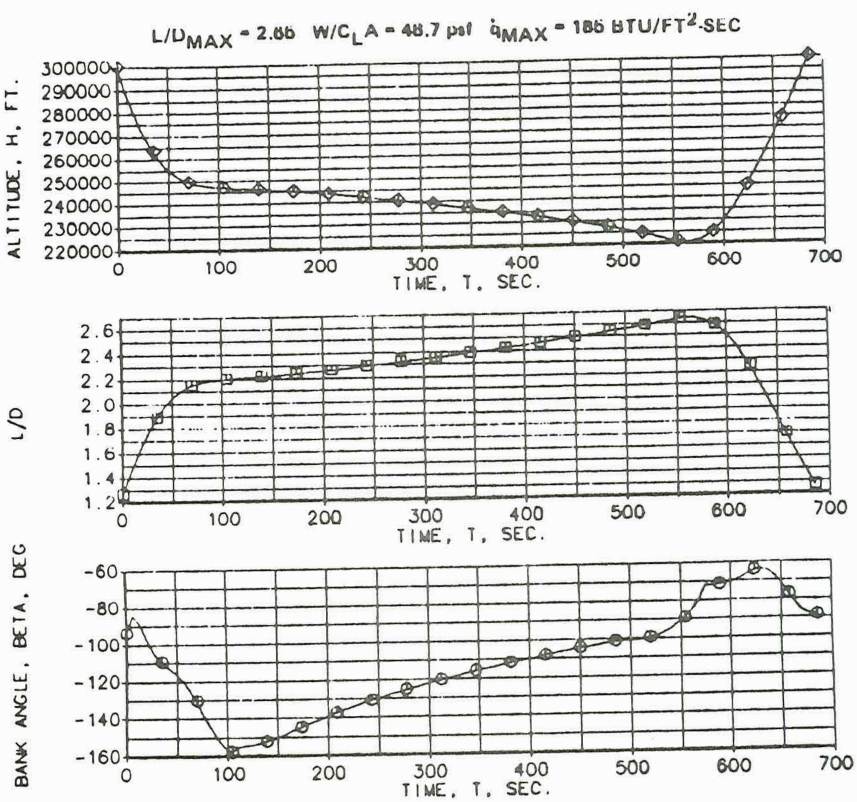

Figure 8. Space-Based High L/D AOTV Optimized GEO Reentry Trajectory

C-2

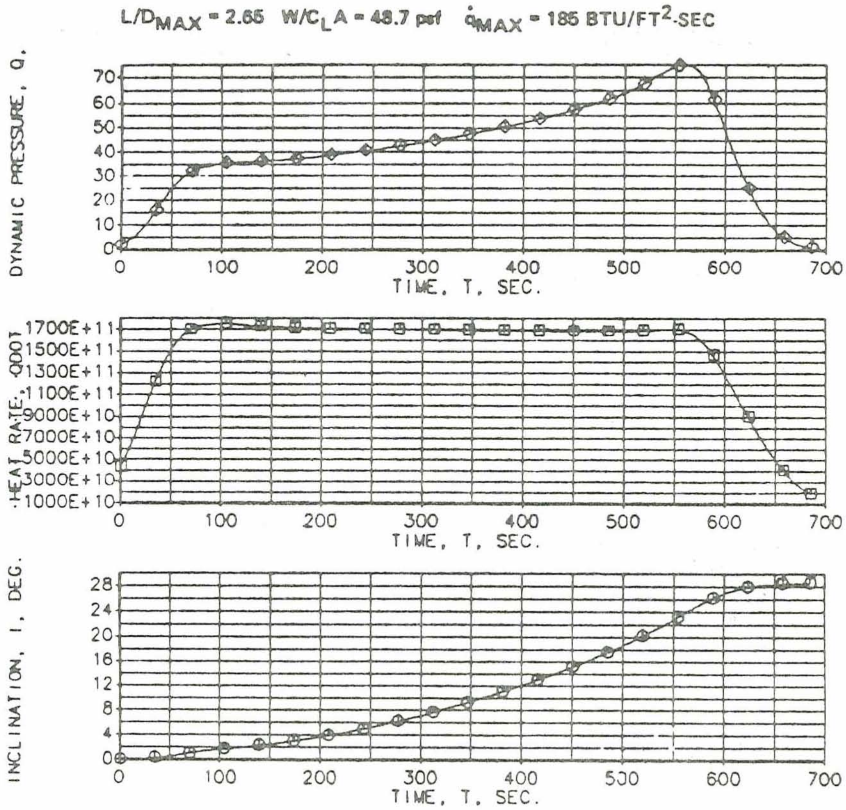

Figure 8. Space-Based High L/D AOTV Optimized GEO Reentry Trajectory (Cont'd)

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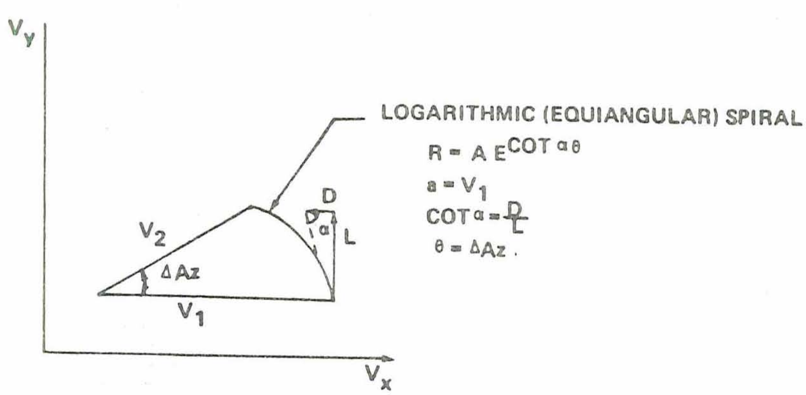


Figure 9. Aeromaneuver in Velocity Space

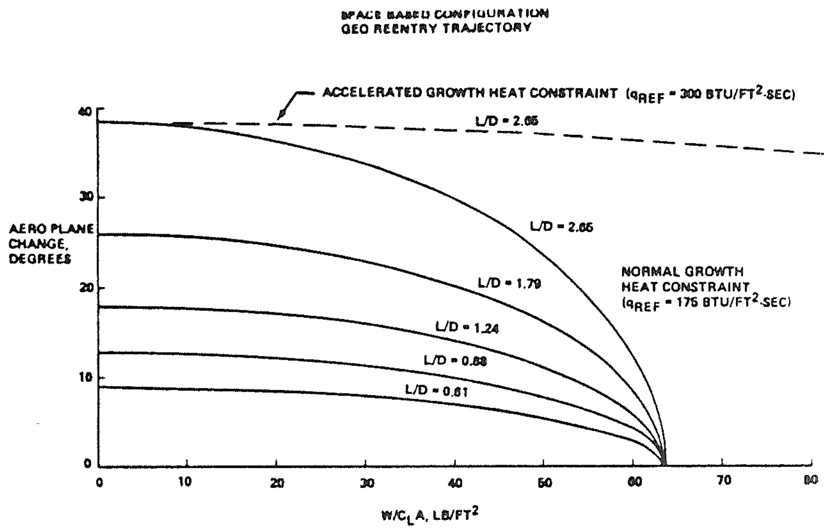


Figure 10. Impact of Heating Constraints on Aero Plane-Change Capability

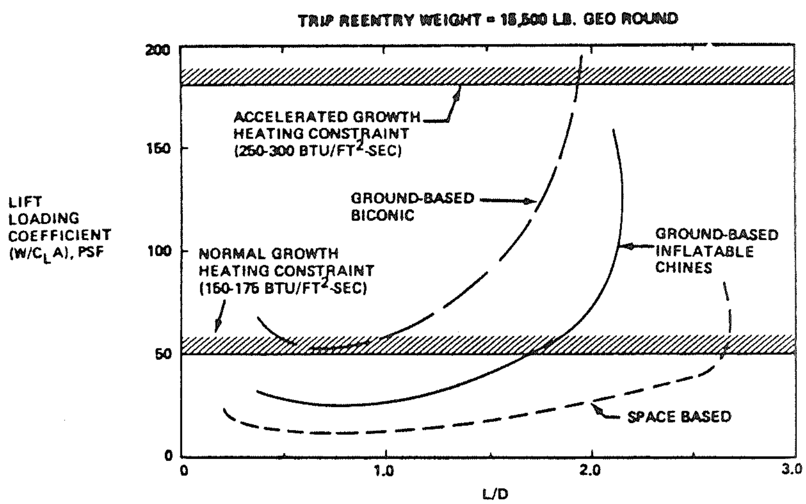


Figure 11. Impact of Heating Constraints on Reentry L/D Ratio

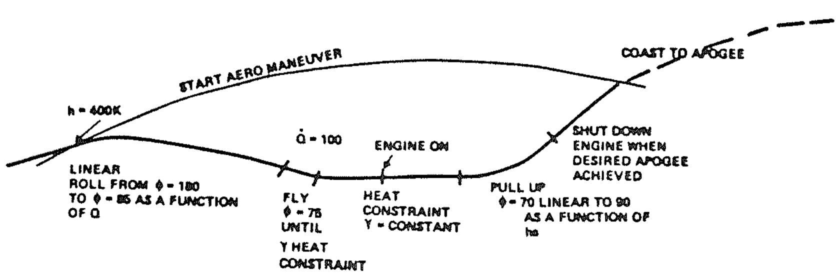


Figure 12. Synergistic Atmospheric Phase

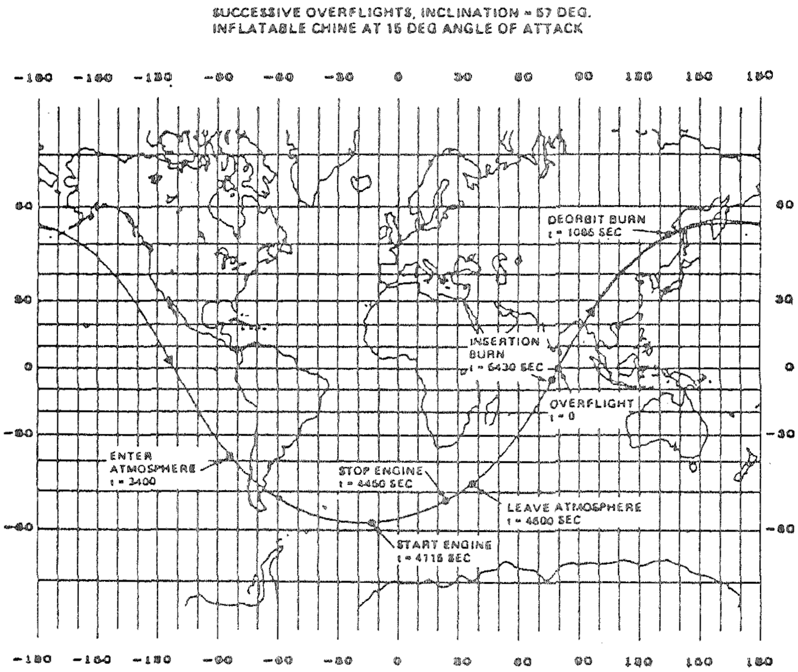


Figure 13. Ground-Based Sortie Mission

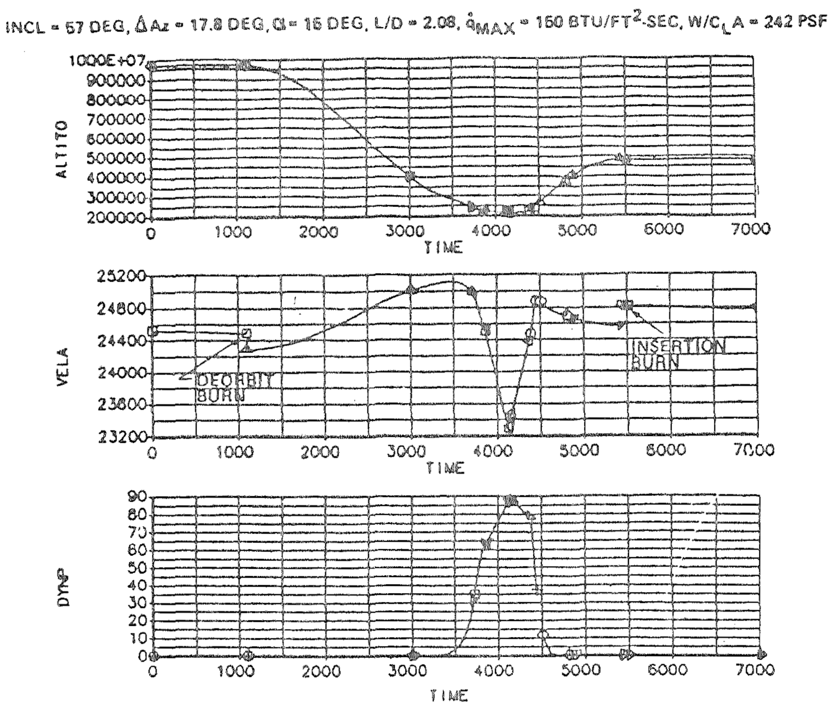


Figure 14. Ground-Based Sortie Mission

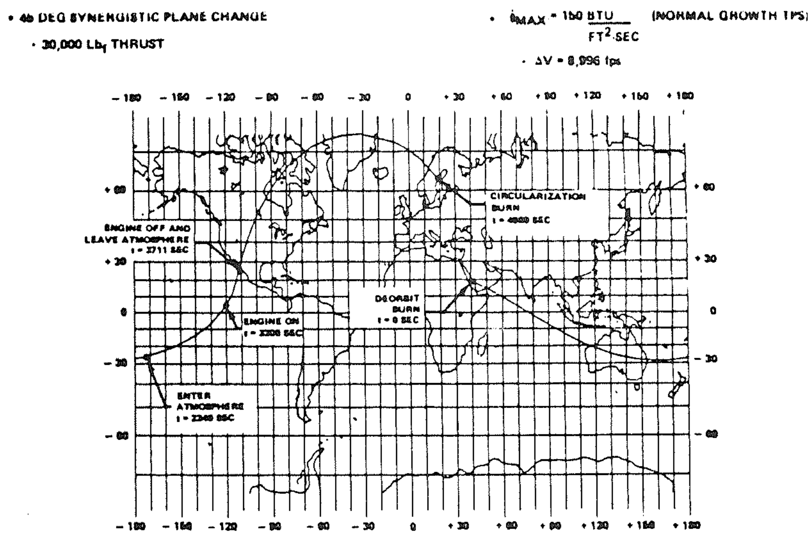


Figure 15. Space-Based Sortie Mission

- CONSTANT ANGLE OF ATTACK FOR L/D MAX
- 150 $\frac{BTU}{FT^2\text{-}SEC}$ MAXIMUM HEATING RATE
- 45 DEGREE PLANE CHANGE USING SPACE-BASED AOTV

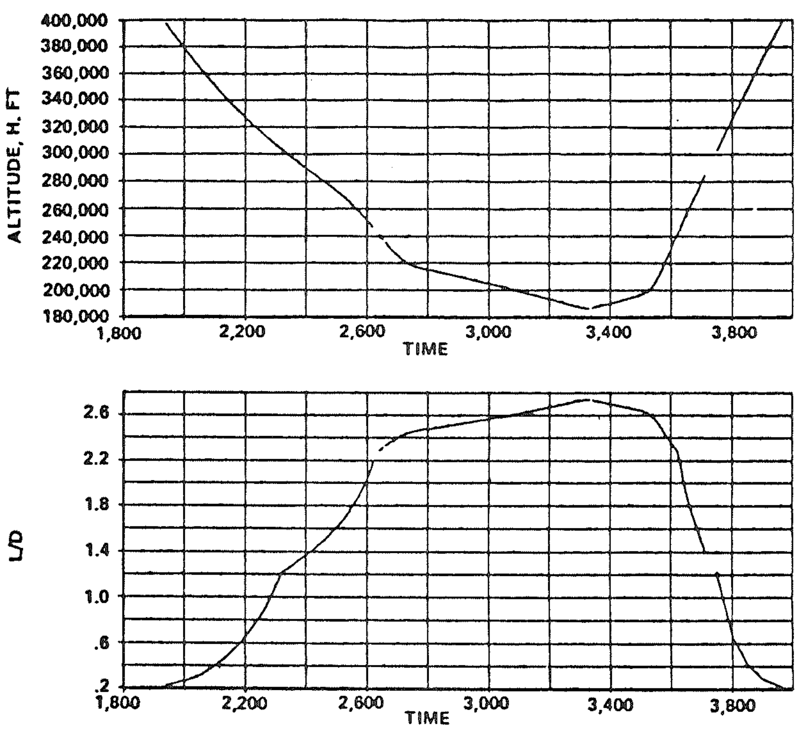


Figure 16. Variation in L/D_{Max} Over Synergistic Plane-Change Trajectory

Δ INC = 45 DEG, L/D_{MAX} = 2.65, $\left(\frac{W}{C_L A}\right)_{INITIAL}$ = 205 psf

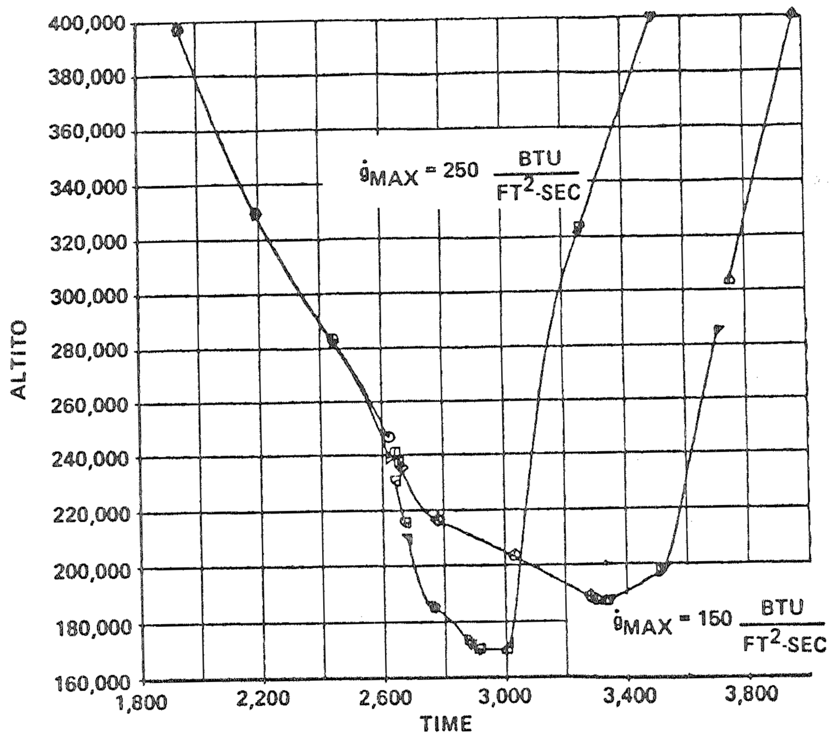


Figure 17. Space-Based Sortie Mission

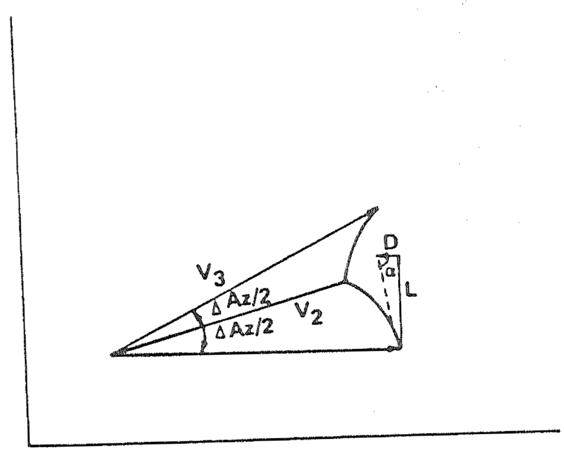


Figure 18. Synergistic Maneuver in Velocity Space

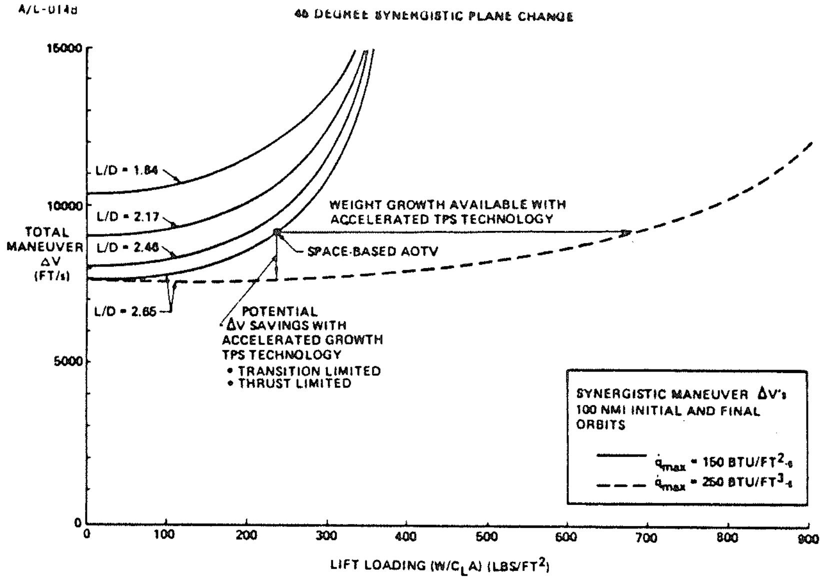


Figure 19. Impact of TPS Technology on Space-Based AOTV

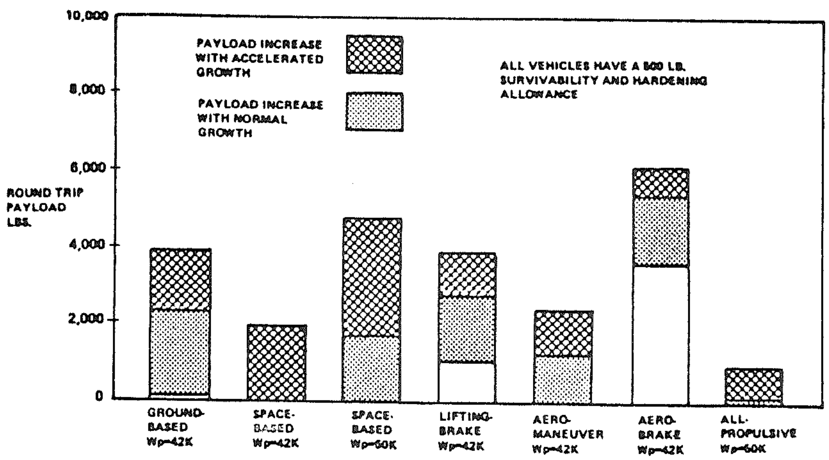


Figure 20. Round Trip GEO Mission Performance Comparison

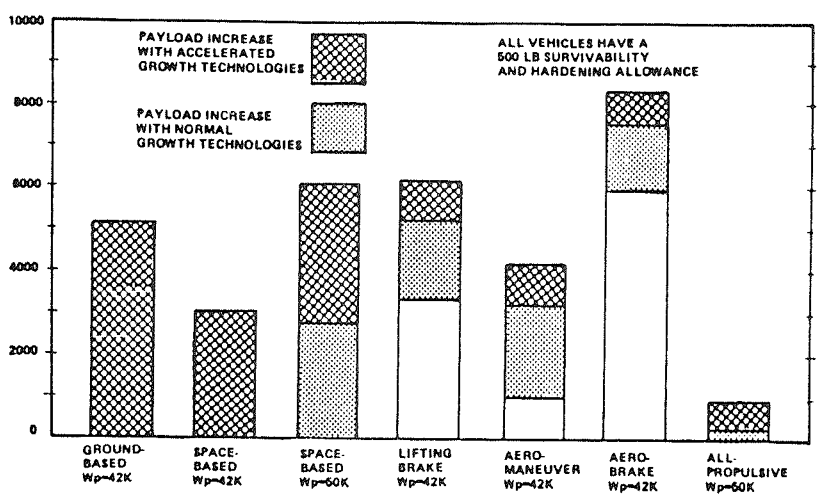


Figure 21. Round Trip 5xGEO (Polar) Mission Performance Comparison

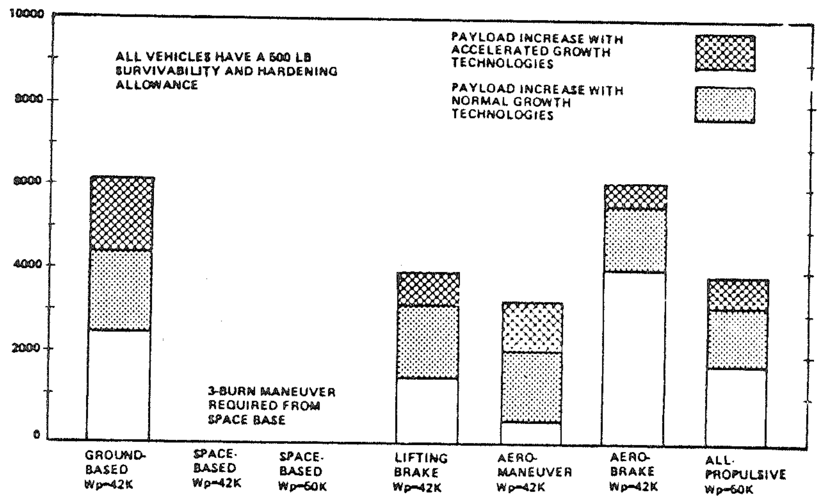


Figure 22. Round Trip 6-Hour Polar Orbit Mission Comparison

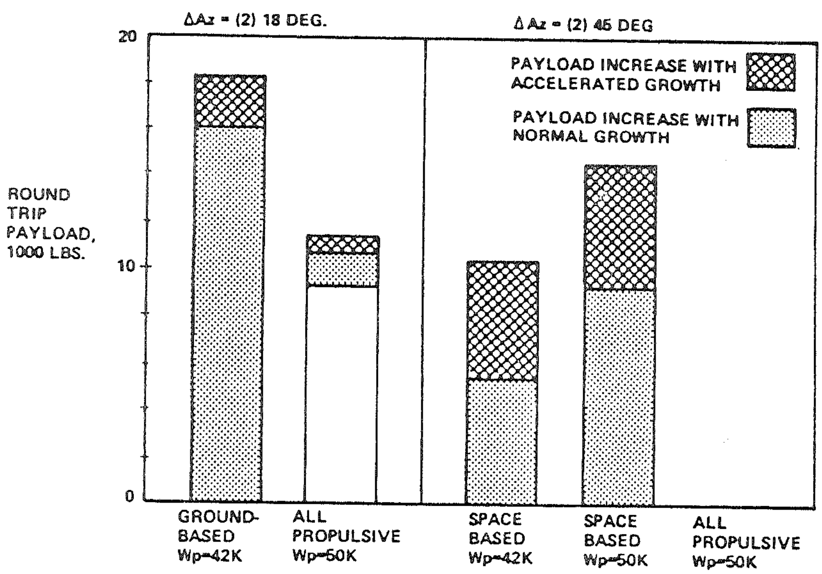


Figure 23. Synergistic Plane-Change Mission Performance Comparison

D8

N85-16997

OTV PROPULSION TECHNOLOGY PROGRAMMATIC OVERVIEW

Larry P. Cooper
NASA Lewis Research Center

To meet the propulsion needs for future Orbit Transfer Vehicles (OTV), NASA has established the Advanced OTV Propulsion Technology Program. An overview of this program is presented.

For the 1990's and beyond it is envisioned that an advanced OTV will be an integral part of the National Space Transportation System (fig. 1), carrying men and cargo between low Earth orbit and geosynchronous orbit as well as performing planetary transfers and delivering large acceleration limited space structures to high Earth orbits. This OTV will be driven by the need to achieve significant reductions in the operational costs for orbit transfer.

To support this scenario, the Advanced OTV Propulsion Technology Program was initiated in 1981. Its objective (fig. 2) is to establish an advanced propulsion technology base for an OTV for the mid 1990's. The program supports technology for three unique engine concepts. Efforts are being conducted in generic technologies which benefit all three concepts as well as specific technology which benefits only one of the concepts.

Figure 3 shows the program goals and requirements. These goals have been established as technology challenges to generate options and tradeoffs although they may not be achievable singularly or concurrently.

NASA Lewis Researach Center has responsibility (fig. 4) for the overall accomplishment of the program's objective, under the cognizance of the Transportation Systems Office of the Office of Aeronautics and Space Technology and with assistance from other NASA Centers. An Advisory Committee has been established to provide technical support to the Program Managers at Lewis Research Center and Marshall Space Flight Center. The overall management structure is shown in figure 5.

The program elements (fig. 6) include concept and technology definition to identify propulsion innovations and subcomponent research to explore and validate their potential benefits. Approximately one-half of the program resources support the three engine manufacturers and the remainder supports university grants, in-house work at NASA Centers, and generic research with industry.

An expansion of the program is being proposed for 1986 (fig. 7) by NASA OAST to enable validation of component and systems level engine capabilities in a realistic operating environment. Support for this research engine proposal has been established through a series of reviews (fig. 8) with government and industry. The total program (fig. 9) will extend into 1992 with approximately one-quarter of the additional resources supporting component evaluation at NASA Centers and the remainder being expended by the three engine manufacturers for hardware, software, and testing of the components and integrated research engines. This expansion of the program to include research engines is designed to be a precursor to a development program (fig. 10) and will allow the latest technology to be incorporated in the advanced engine while providing a low risk, minimum cost development program.

INTEGRATED SPACE TRANSPORTATION

1990's SCENARIO

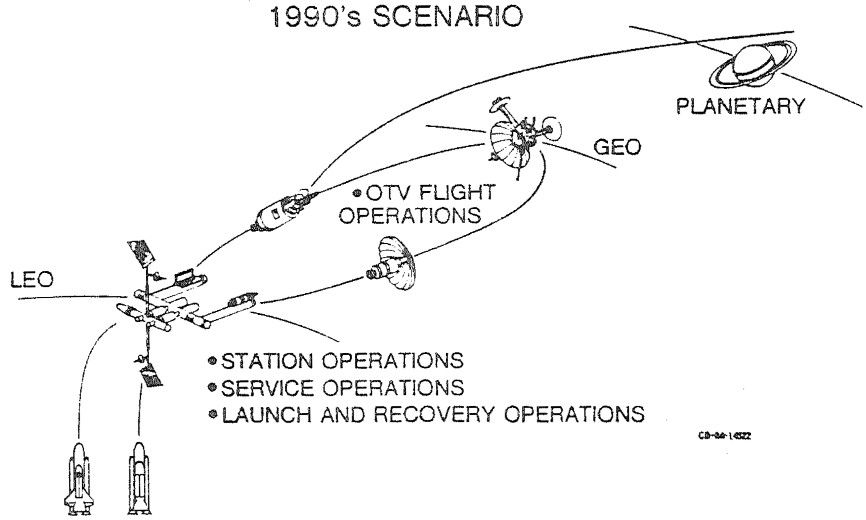


Figure 1

ADVANCED ORBITAL TRANSFER PROPULSION

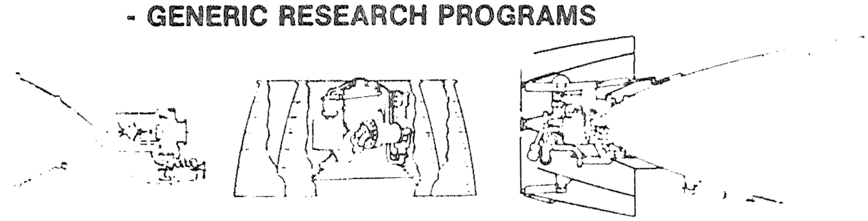
OBJECTIVE

TO ESTABLISH THE TECHNOLOGY BASE FOR A HIGHLY VERSATILE, SPACE BASABLE, REUSABLE, MAN RATABLE ENGINE FOR ORBITAL TRANSFER VEHICLES FOR MID-1990's IOC

APPROACH

MULTI-ELEMENT PROGRAMS SUPPORTING TECHNOLOGIES FOR UNIQUE OTV ENGINE CONCEPTS AT AEROJET, PRATT & WHITNEY, AND ROCKETDYNE

- CONCEPT SPECIFIC PROGRAMS
- GENERIC RESEARCH PROGRAMS



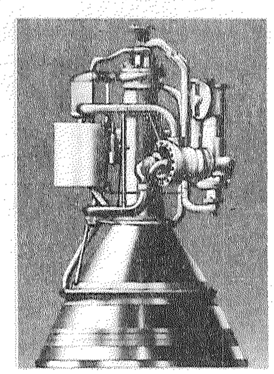
• PROPULSION/VEHICLE ANALYSIS • TURBOMACHINERY • NOZZLES
• THRUST CHAMBERS • HEALTH MONITORING • CONTROLS

Figure 2

OTV PROPULSION SYSTEM CHALLENGES

GOALS

VACUUM SPECIFIC IMPULSE lbf-sec/lbm	520
VACUUM THROTTLE RATIO	30:1
NET POSITIVE SUCTION HEAD, lbf-ft/lbm	0
WEIGHT, lbm	360
LENGTH (STOWED), INCH	40
RELIABILITY	1.0
SERVICE LIFE	
BETWEEN OVERHAULS, CYCLES/hr	500/20
SERVICE FREE, CYCLES/hr	100/4



REQUIREMENTS

PROPELLANTS	HYDROGEN/OXYGEN
TOTAL VACUUM THRUST, lbf	10,000 - 25, 000
ENGINE MIXTURE RATIO	6 ± 1

CD-83-13874

Figure 3

MANAGEMENT

- NASA OAST - TRANSPORTATION SYSTEMS OFFICE
- LeRC LEAD CENTER
- LeRC AND MSFC ARE FIELD CENTERS RESPONSIBLE FOR PROGRAM TASKS
- ADVISORY COMMITTEE FROM LeRC, LaRC, MSFC, & AFRPL

Figure 4

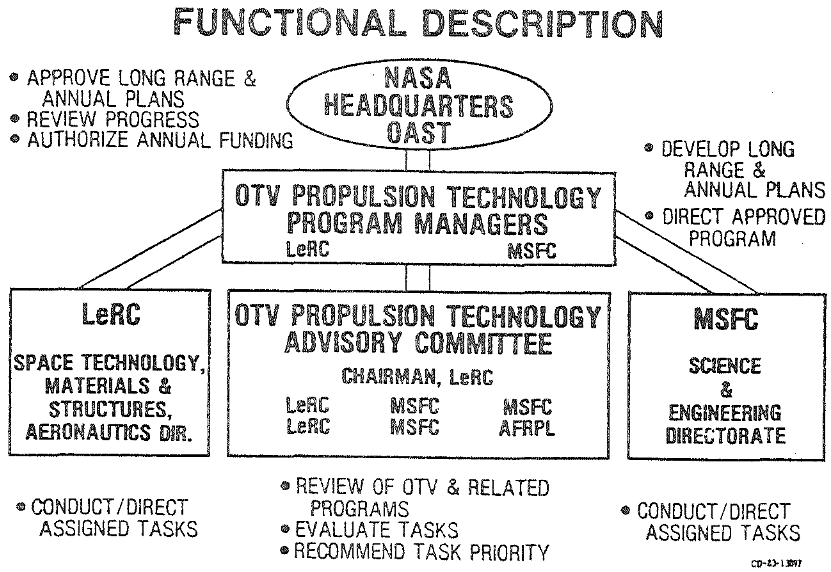


Figure 5

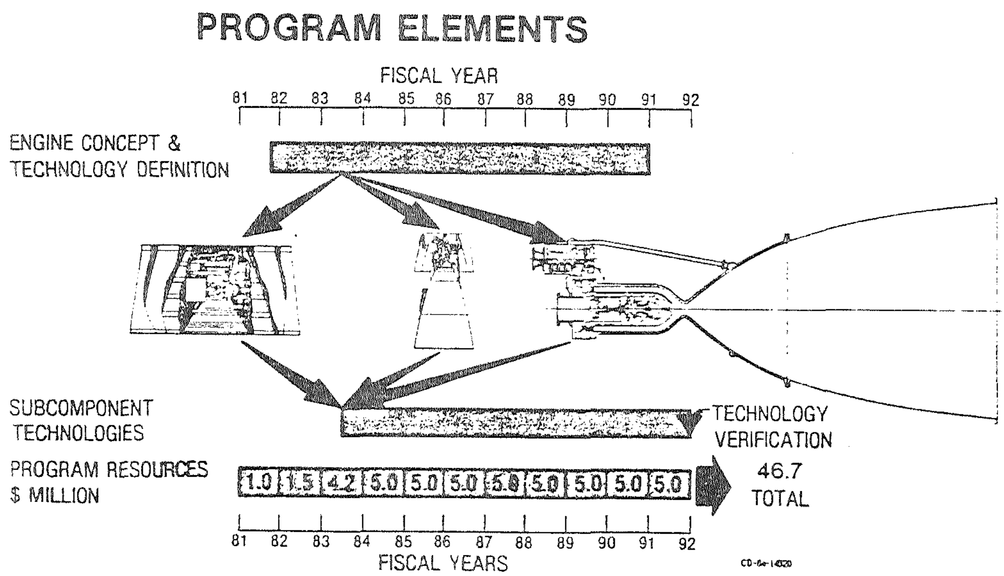


Figure 6

RESEARCH ENGINE INITIATIVE

OBJECTIVE

TO VALIDATE COMPONENT AND SYSTEMS LEVEL PERFORMANCE/ CAPABILITY FOR A HIGHLY VERSATILE OTV ENGINE WITH IOC IN THE MID 1990's

APPROACH

EVALUATE WITH EACH ENGINE MANUFACTURER SUBCOMPONENTS, COMPONENTS, AND SYSTEM LEVEL INTERACTIONS FOR RESPECTIVE ENGINE CONCEPTS

PAYOFF

ASSESSMENT OF INTEGRATED ENGINE CONCEPTS IN REALISTIC OPERATING ENVIRONMENT

Figure 7

BACKGROUND & REVIEWS

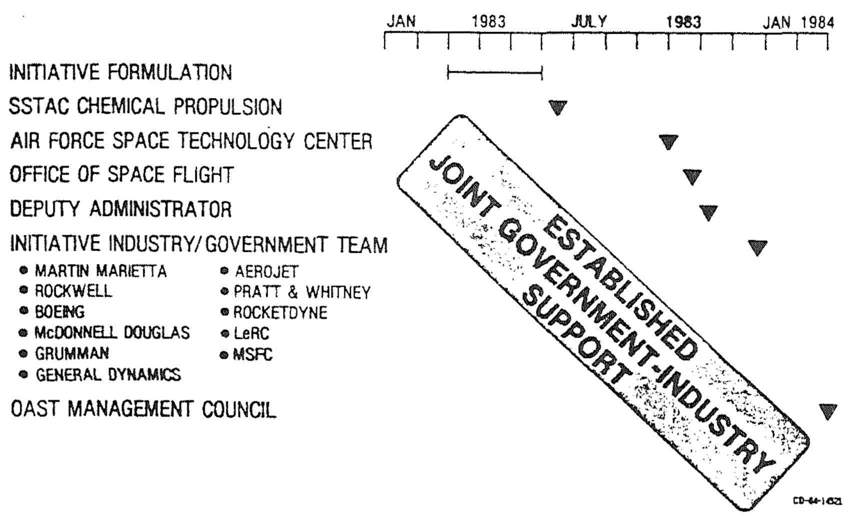


Figure 8

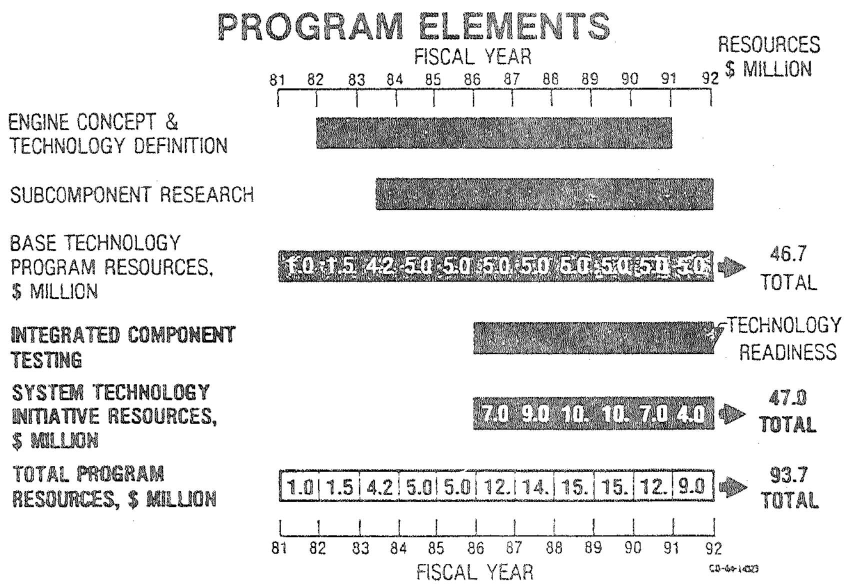


Figure 9

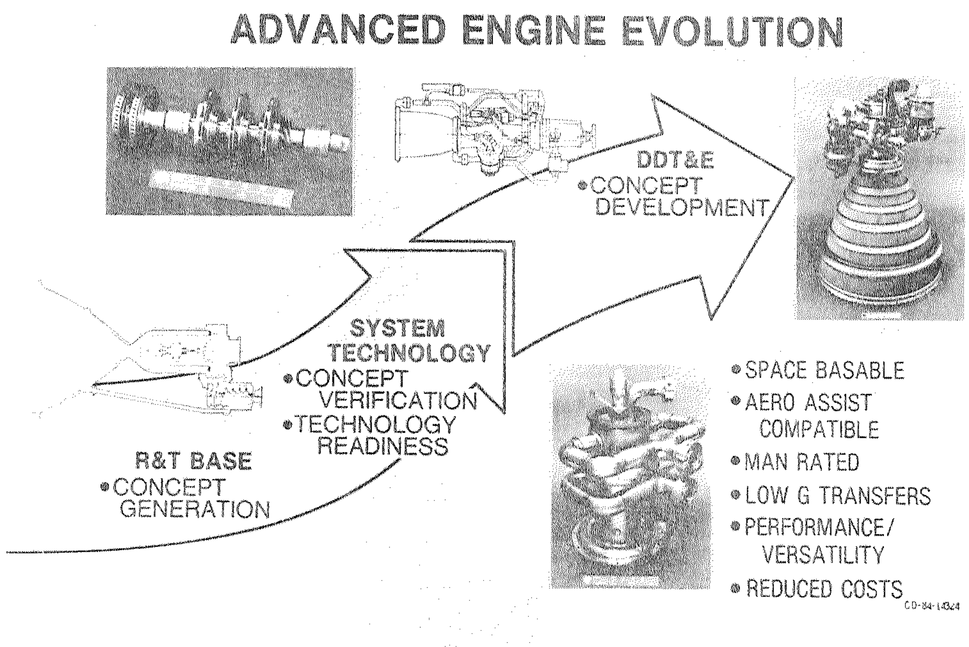


Figure 10

N85-16998

DRIVES AND BENEFITS OVERVIEW

S. D. McIntyre
NASA Marshall Space Flight Center

The presentation covers the major technology issues for an advanced OTV engine to be used in conjunction with a space based, reusable orbit transfer vehicle. A brief summary of the results of the space station studies conducted in 1983 as they relate to the OTV is given as well as a brief review of ground rules and guidelines for a reusable OTV vehicle study which is currently being initiated at MSFC. The technology drives are presented and related to benefit categories i.e., mission versatility, increased reliability or reduced cost. The technology drivers and the associated benefits are then covered in detail with regard to relative significance and impact on the on-going OTV engine technology program. The concluding summary recommends that based on the maintenance opportunity afforded by the Space Station, the broad range of mission requirements and the long term potential cost benefits a new engine is needed for the space based reusable OTV.

OTV PROPULSION ISSUES
DRIVERS & BENEFITS

- IN THE MID 1990'S THE U. S. WILL NEED A NEW SPACE BASED OTV WHICH WILL BE PART OF AN INTEGRATED SPACE TRANSPORTATION SYSTEM CONSISTING OF:
  - SHUTTLE ORBITER (STS) (EXISTING)
  - SPACE STATION (SS) (1990)
  - SPACE BASED ORBIT TRANSFER VEHICLE (OTV) (1995)
  - ORBITAL MANEUVERING VEHICLE (OMV) (1990)
- OTV VEHICLE STUDIES ARE BEING INITIATED NOW AT MSFC TO DEFINE REQUIREMENTS AND CONCEPTS. A SUMMARY OF GROUND RULES AND GUIDELINES FOR THESE STUDIES ARE AS FOLLOWS:
  - INVESTIGATE USE OF SYSTEMS/SUBSYSTEMS FROM EXISTING AND PLANNED VEHICLES
  - ALL CONFIGURATIONS SHALL <u>EVOLVE</u> TO BECOME: (OR BE THAT WAY FROM OUTSET)
    - REUSABLE
    - SPACE–BASED
    - INCORPORATE AERO ASSIST (OR ALTERNATE APPROACHES)
    - MAN RATABLE
    - CRYOGENIC (OR ALTERNATE APPROACHES)
  - SUFFICIENT DETAIL TO DETERMINE COST AND VIABILITY OF EVOLUTIONARY APPROACH
  - GROUND BASED CONCEPTS SHALL INCLUDE OPERATION IN CONJUNCTION WITH THE SPACE STATION

Figure 1

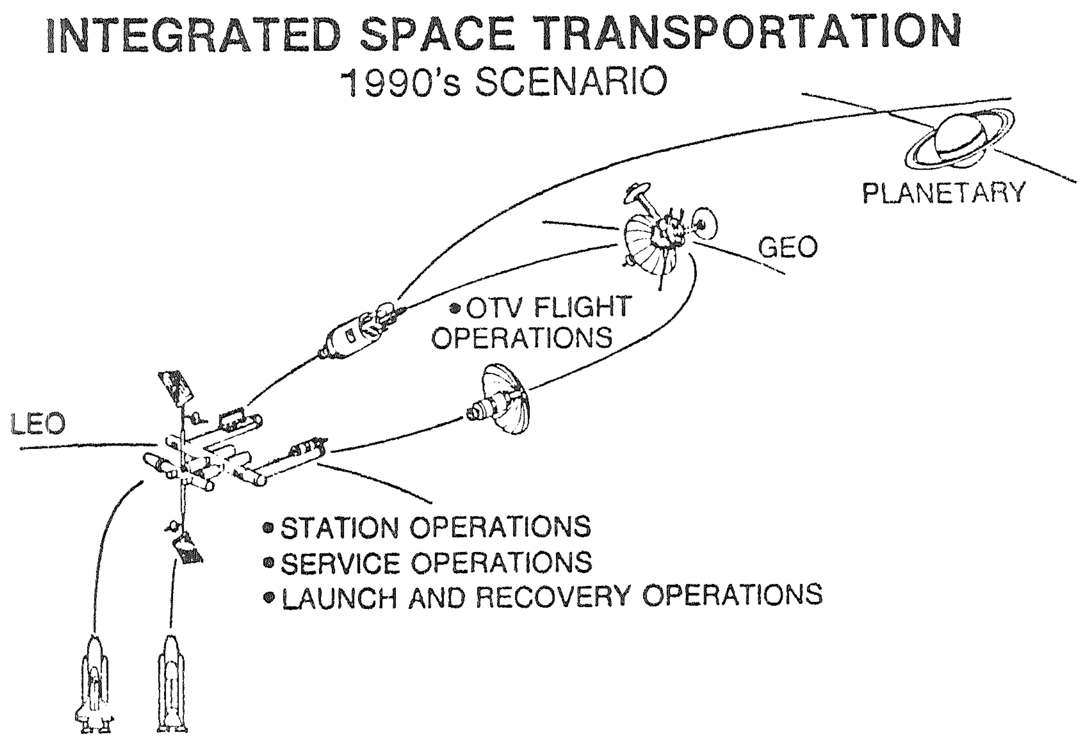


Figure 2

Mature Configuration (ACC)

Modular ACC Option

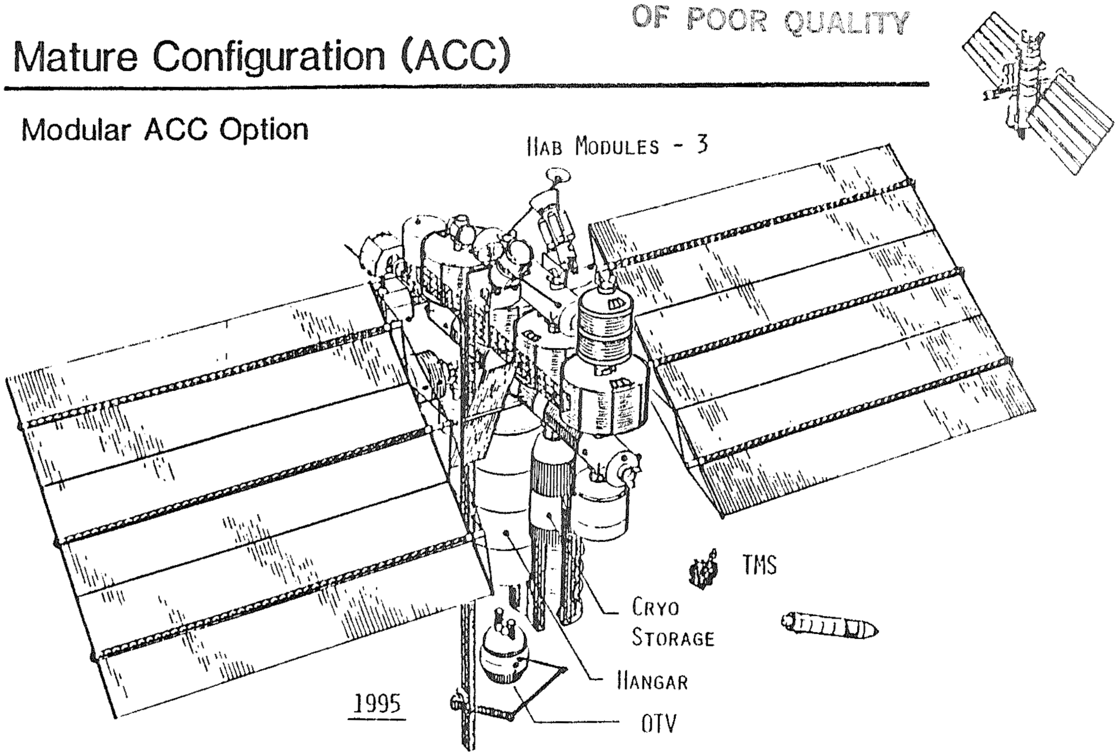
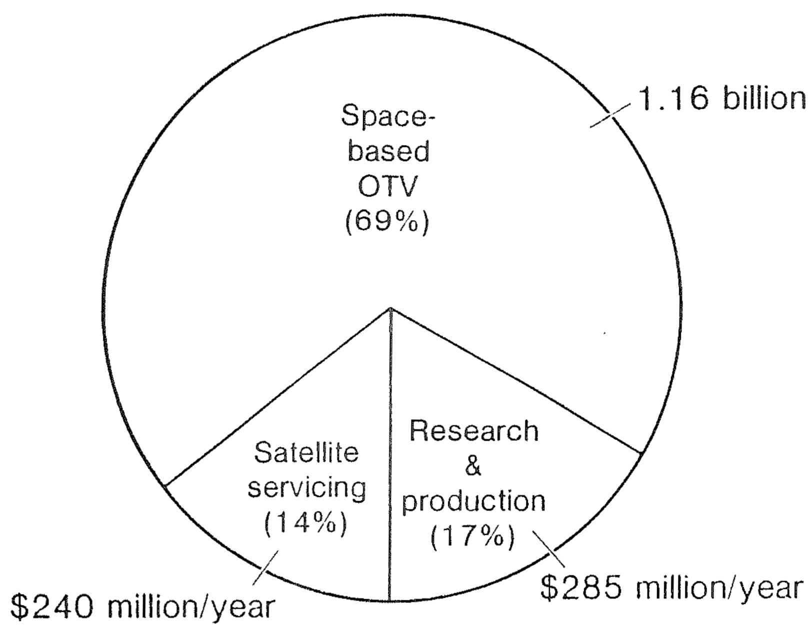


Figure 3

SPACE STATION ECONOMIC BENEFITS
(1984 $)



Total economic benefit: $1.685 billion

Figure 4

OTV PROPULSION ISSUES
DRIVERS & BENEFITS

TECHNOLOGY DRIVERS AND BENEFITS

TECHNOLOGY DRIVER	BENEFITS: MISSION VERSATILITY	INCREASED RELIABILITY	REDUCED COST
SPACE BASED	√	√	√
MAN RATED	√	√	
THRUST LEVEL	√		
AERO ASSIST COMPATIBLE	√		√
PERFORMANCE	√		√
REUSABILITY/LIFE		√	√

Figure 5

OTV PROPULSION ISSUES
DRIVERS & BENEFITS

SPACE BASING BENEFITS FOR OTV MISSIONS

- SPACE STATION SERVES AS A HOLDING AREA FOR:
  - PROPELLANTS
  - OTV SYSTEMS
  - PAYLOADS

  THESE ITEMS CAN BE LAUNCHED, STORED AND ASSEMBLED IN MOST COST EFFECTIVE WAY
- POTENTIAL ECONOMIC BENEFIT FOR OTV MISSIONS IS ESTIMATED TO BE $5B THRU CY2000
- SPACE STATION CREATES OR ENHANCES OPPORTUNITY FOR:
  - LOW COST DELIVERY OF OTV PROPELLANT TO SS
  - MULTIPLE PAYLOADS ON OTV
  - MISSION VERSATILITY WITH MODULAR OTV SYSTEMS

IMPACT OF SPACE BASING ON EXISTING OTV PROPULSION TECHNOLOGY DRIVERS

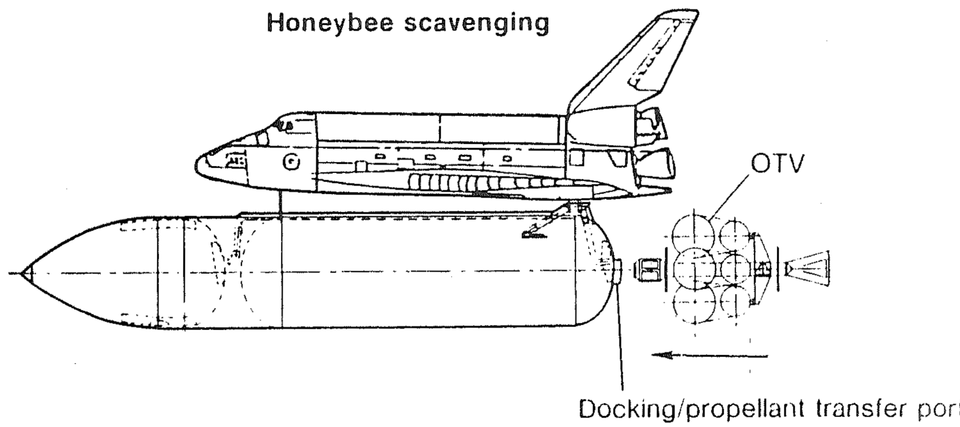
- PERFORMANCE – REDUCED PROPELLANT DELIVERY COST
- LIFE – SERVICE AND MAINTENANCE OPTIONS CREATED
- SIZE – FINAL ASSEMBLY AT SPACE STATION

NEW TECHNOLOGY DRIVERS INTRODUCED

- MODULAR DESIGN – ON-ORBIT ASSEMBLY
- HEALTH MONITORING, DIAGNOSTICS AND IN-FLIGHT CHECKOUT

Figure 6

PROPELLANT DELIVERY SYSTEM CONCEPTS



Performance

- Propellant delivered to station per mission —
  11,300 pounds
- Propellant delivered to station per year —
  230,000 to 270,000 pounds
- Propellant delivery cost —
  $250/pound

Figure 7

ROTV PROPELLANT REQUIREMENT

CY 1990 – 2000

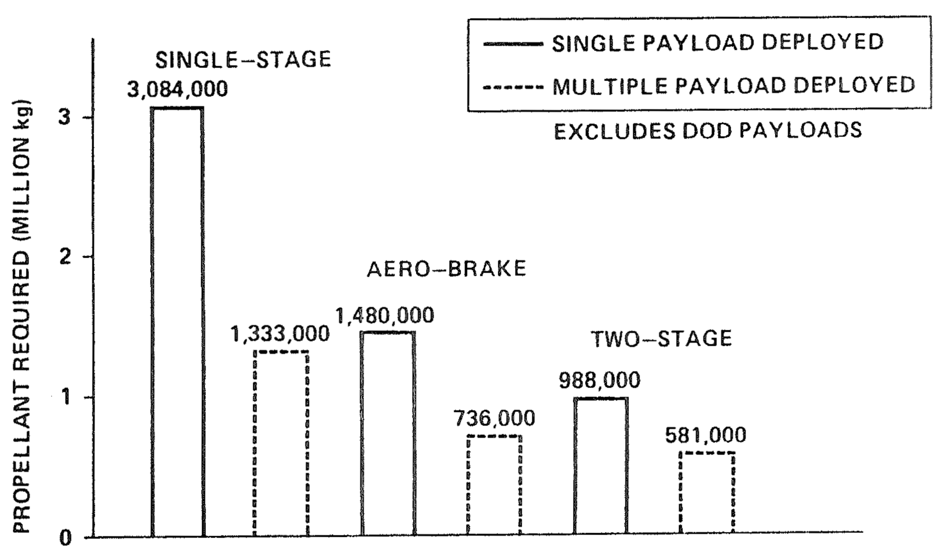


Figure 8

STS/SPACE BASEABLE PACKAGING CONCEPTS

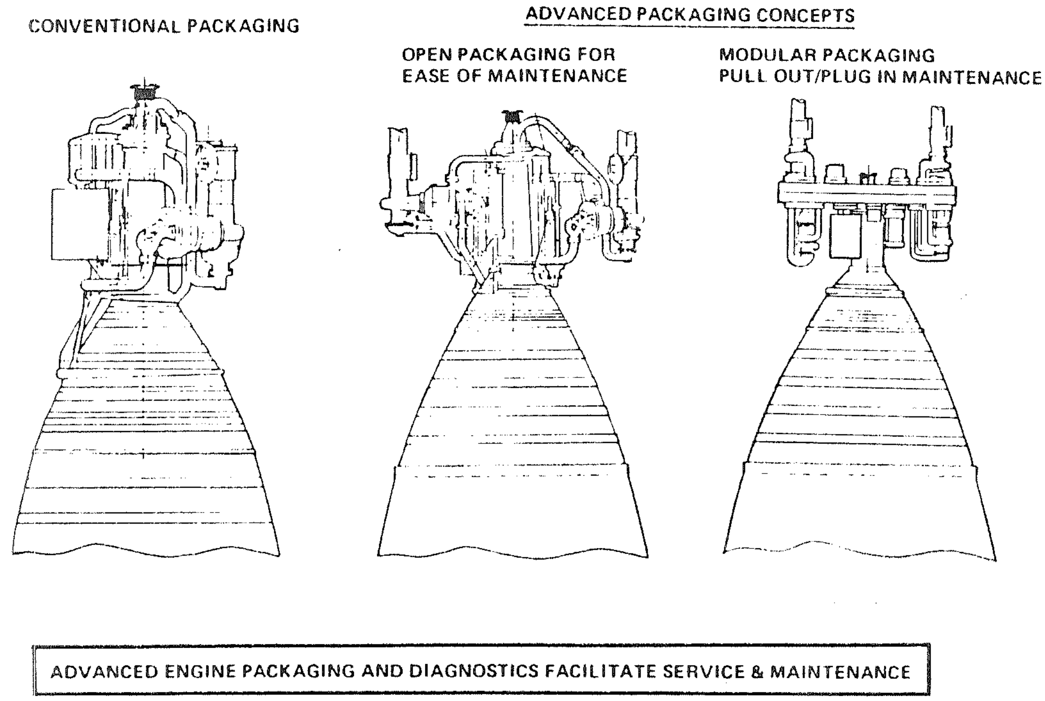


Figure 9

DIAGNOSTICS FOR MAINTAINABILITY APPROACH

ACHIEVED BY USING A BETWEEN-FLIGHT AND/OR IN-FLIGHT CONDITION MONITORING SYSTEM CONSISTING OF STATE OF THE-ART AND/OR NOVEL AUTOMATED DETECTION TECHNOLOGIES AND TAILORED DATA PROCESSING AND COMPUTERS

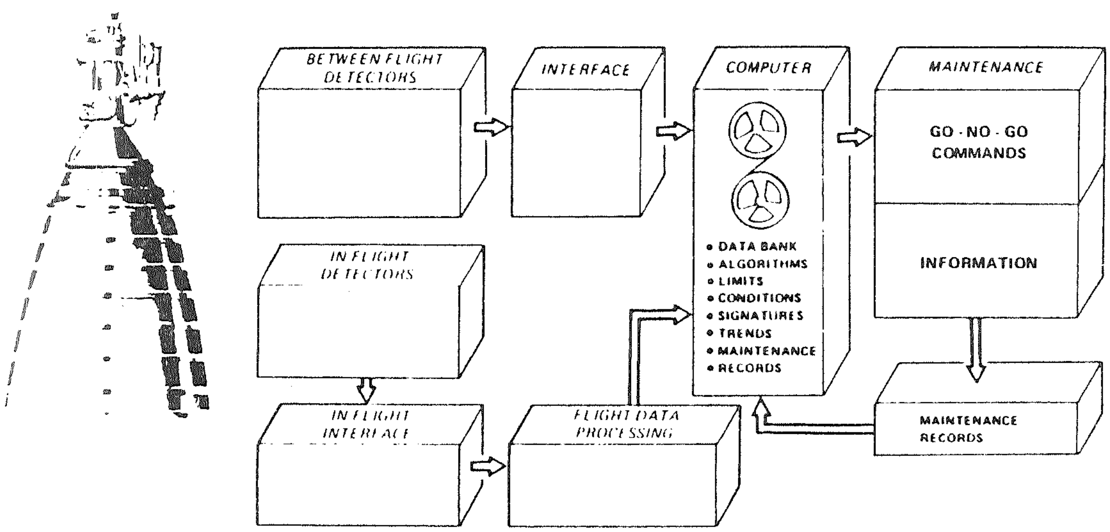


Figure 10

OTV PROPULSION ISSUES
DRIVERS & BENEFITS

MAN RATABLE OTV BENEFITS

- ENABLES COMPLEX SATELLITE SERVICING TASKS
- ENHANCES MISSION VERSATILITY AND RELIABILITY
- COULD SAVE REPLACEMENT OF A COMPLEX PAYLOAD

MAN RATABLE OTV PROPULSION TECHNOLOGY DRIVERS

- MAN RATING MEANS REDUNDANCY OR THE ELIMINATION OF ALL SINGLE POINT FAILURES
- LARGER MARGINS OF SAFETY ON STRUCTURAL COMPONENTS
- CONCENSUS IS MAN RATED OTV WOULD HAVE 2 ENGINES
  - MAN RATING THEN LEADS TO ENGINE THRUST LEVEL ISSUE
  - TOTAL THRUST FOR OTV ESTIMATED TO 10 – 20 K LB.
  - SINGLE ENGINE THRUST ON 2 ENGINE VEHICLE IS THEN 5 – 10 K LB.

Figure 11

OTV PROPULSION ISSUES
DRIVERS & BENEFITS

THRUST LEVEL BENEFITS

- MISSION VERSATILITY WHICH INCLUDES LOW g AND MANNED MISSIONS
- CONTINUOUS LOW THRUST THROTTLING TO MAINTAIN CONSTANT T/W RATIO ON LARGE DEPLOYED STRUCTURE PAYLOADS
- SELECTION OF A PROPULSION/ENGINE SYSTEM WHICH BEST ACCOMMODATES THE RANGE OF KNOWN OR ANTICIPATED MISSION REQUIREMENTS

THRUST LEVEL RELATED OTV PROPULSION TECHNOLOGY DRIVERS

- THE SMALL LH_2/LO_2 PUMP FED ENGINE (5 – 10K) IS ITSELF APPROACHING A NEW TECHNOLOGY AREA
  - DESIGN AND MANUFACTURING TECHNIQUES FOR SMALL DIAMETER HIGH SPEED PUMP COMPONENTS IS A NEW TECHNOLOGY AREA
  - BOUNDARY LAYER EFFECTS ON HEAT TRANSFER AND PERFORMANCE BECOME MORE SIGNIFICANT AS THE SIZE OF ENGINE IS REDUCED
- SPACE BASING AND MODULAR ENGINE DESIGN ENHANCES OPPORTUNITY TO USE "KITS" TO ACHIEVE A LOW THRUST ENGINE CONFIGURATION, i.e., PUMPS, INJECTORS, THRUST CHAMBER OR NOZZLES DESIGNED FOR LOW THRUST OPERATION AND MAXIMUM PERFORMANCE.

Figure 12

TECHNOLOGY SCALING

GOAL	EASE OF ACHIEVEMENT	IMPACTS
PERFORMANCE	MORE DIFFICULT AT LOWER THRUST	• LOW PUMP AND TURBINE EFF. • HIGH COOLANT JACKET ΔP • LOWER TURBINE ADMISSION
SYSTEM CONTROL	MORE DIFFICULT AT LOWER THRUST	• LOWER FLOWRATES • GREATER INSTRUMENT PRECISION REQ.
ENGINE CONDITIONING	MORE DIFFICULT AT LOWER THRUST	• HIGHER MASS PER UNIT FLOWRATE • LARGER SURFACE AREA PER UNIT FLOW
MANUFACTURABILITY	MORE DIFFICULT AT LOWER THRUST	• LESS METAL TO CUT • TIGHTER TOLERANCES • TURBINE, PUMP, AND T/C FAB LIMITS
MAINTAINABILITY	MORE DIFFICULT AT LOWER THRUST	• SMALLER PASSAGES • MORE DIFFICULT INSPECTION • DIAGNOSTIC SENSORS NOT SCALEABLE • SMALLER TOOLING
SYSTEM DESIGN	MORE DIFFICULT AT LOWER THRUST	• BETTER MATERIALS REQUIRED
LIFE	MORE DIFFICULT AT LOWER THRUST	• HIGHER ΔP FOR EQUAL LIFE • HIGHER SURFACE TEMPERATURES

Figure 13

OTV PROPULSION ISSUES
DRIVERS & BENEFITS

AERO ASSIST BENEFITS

- REDUCED PROPELLANT REQUIREMENT BY USING ATMOSPHERE AS BRAKE
- TWO STAGE OTV IS EQUALLY EFFECTIVE FOR DELIVERY ONLY MISSIONS
- AERO ASSIST MOST EFFECTIVE FOR DELIVERY AND RETURN MISSIONS, i.e., MANNED MISSIONS OR PAYLOAD SERVICING AT SPACE STATION

AERO ASSIST RELATED OTV PROPULSION TECHNOLOGY DRIVERS

- MEDIUM TO HIGH L/D CONCEPTS HAVE STRONG PREFERENCE FOR SMALL ENGINES WHICH CAN BE CONTAINED WITHIN THE VEHICLE PROFILE.
- LOW L/D BALLUTE CONCEPT REQUIRES THE ENGINE TO PROVIDE A GAS LAYER THERMAL BARRIER OVER THE INFLATED BALLUTE MATERIAL.
- LOW L/D RIGID SHIELD CONCEPT IS SENSITIVE TO ENGINE LENGTH SINCE DEPLOYMENT OVER THE ENGINE IS REQUIRED FOR THERMAL PROTECTION.

Figure 14

OTV PROPULSION ISSUES
DRIVERS & BENEFITS

PERFORMANCE BENEFITS

- REDUCED PROPELLANT REQUIREMENT AND PAYLOAD DELIVERY COST
- ADVANCED CONCEPTS ARE PREDICTED TO DELIVER ~ 480 SEC ISP OR A 40 SEC. INCREASE OVER RL-10-3A REF. ENGINE (440 SEC)
- INCREASED PERFORMANCE (440 – 480 SEC) REPRESENTS 20% REDUCTION IN PROPELLANT REQUIRED OR 20% INCREASE IN PAYLOAD
- THE OVERALL SIGNIFICANCE OF THIS IMPROVED PERFORMANCE IS A FUNCTION OF:
  - PROPELLANT DELIVERY AND STORAGE COST AT LEO OR SS
  - USE OF AERO ASSIST OR STAGING CONCEPTS

PERFORMANCE RELATED OTV PROPULSION TECHNOLOGY DRIVERS

- HIGHER CHAMBER PRESSURE AND PUMP SPEEDS
- ENHANCED HEAT TRANSFER TO PROVIDE MORE POWER TO PUMPS
- HIGH AREA RATIO NOZZLES WHICH, DEPENDING ON ENGINE SIZE, MUST BE SEGMENTED IF ENGINE LENGTH IS CONSTRAINED.

Figure 15

EFFECT ON SPECIFIC IMPULSE ON OTV PROPELLANT REQUIREMENTS

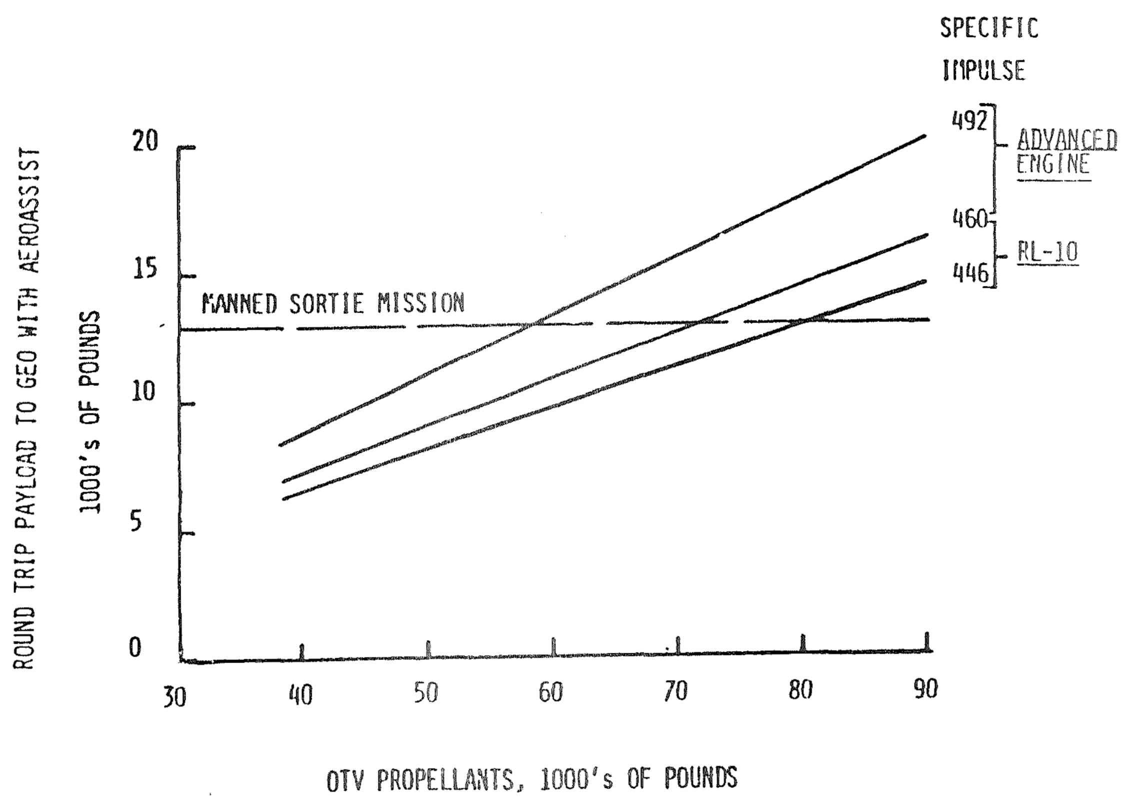


Figure 16

ADVANCED ENGINE PROVIDES PERFORMANCE FLEXIBILITY

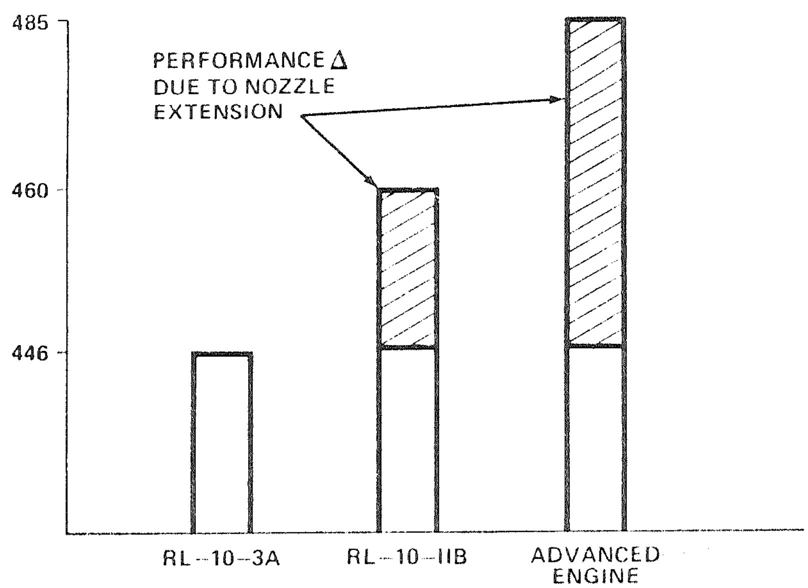


Figure 17

OTV PROPULSION ISSUES
DRIVERS & BENEFITS

REUSABILITY AND EXTENDED LIFE BENEFITS

- REDUCED OPERATING COST

LIFE RELATED OTV PROPULSION TECHNOLOGY DRIVERS

- IMPROVED BEARING, SEAL AND GEAR MATERIALS AND DESIGNS
- IMPROVED THRUST CHAMBER MATERIALS AND DESIGNS
- IMPROVED NOZZLE MATERIALS AND DESIGNS
- NEW CONTROL SYSTEMS, HEALTH MONITORING AND DIAGNOSTIC SYSTEMS TO CONTINUOUSLY MONITOR ENGINE STATUS OVER THE LIFE OF ENGINE

OTV PROPULSION TECHNOLOGY PROGRAM PERSPECTIVE

- THE 1995 SPACE BASED OTV WILL NEED A NEW HYDROGEN/OXYGEN FUELED ENGINE
  - PROVIDES FOR BROAD RANGE OF MISSIONS FROM LOW g TO MANNED FOR NEXT 20 – 40 YEARS
  - WILL TAKE FULL ADVANTAGE OF SERVICING OPTIONS AFFORDED BY THE SPACE STATION
  - PROVIDES IMPROVED RELIABILITY WITH INTEGRATED CONTROL AND HEALTH MONITORING SYSTEMS
  - EXTENDED LIFE AND INCREASED PERFORMANCE WILL PROVIDE A LONG TERM COST BENEFIT

Figure 18

N85-16999

AEROJET ADVANCED ENGINE CONCEPT

L. Schoenman
Aerojet TechSystems Company

The future OTV requirements defined in figure 1 dictate the need for a highly versatile, highly reliable, reusable propulsion module. Aerojet's engine design approach (fig. 2) will provide a total thrust capability of 500 to 18 000 lbF, using one to six continuously throttleable engines. The selection of a nominal thrust level of 3000 lbF best fulfills the overall OTV requirements with a single propulsion system.

In order to attain maximum operational economy, space-basing will be essential. This requires a reausable, maintenance-free engine. The design features of this space-based engine are defined in figure 3.

A new engine cycle and its advantages, shown in figure 4, allow all the maintenance goals of figure 3 to be attained. Rubbing contact and interpropellant seals and purges, etc., are eliminated when GO_2 is used to drive the LO_2 pump, as shown in figure 5. The TPA design shown has only one moving part.

The use of both GH_2 and GO_2 to drive the turbines lowers the turbine temperatures to the values shown in figure 6. In addition, lower GH_2 temperatures and pressures allow improved chamber cooling and longer life.

The use of GO_2 as a turbine drive fluid, even at the low temperature of 400° F, is a concern which is being addressed through extensive materials testing. Friction rubbing and aluminum particle impact test results (fig. 7) indicate that proper selection of materials can eliminate the metals ignition experienced in the past. Stainless steel alloys are a notably poor choice for oxygen service.

Space-based engines will require an integrated control and health monitoring system (fig. 8) to improve system reliability and eliminate all scheduled maintenance.

Engine length is a major consideration when aero-assist return from GEO is employed. Examples of the importance of length are shown in figures 9 to 12.

Figures 13 and 14 show that the use of multiple engines has only minor impact on total propulsion system weight.

The issues associated with low-G transfers are presented in figures 15 to 17. Significant performance losses will develop when a single 15 000 lbF engine is operated at 500 to 1000 lbF. Also, the optimum mixture ratio shifts to the fuel-rich direction during throttling. This, in turn, increases stage volume and dry weight. Figure 17 indicates the relative performance benefit of one or two 3000 lbF engines operated at reduced thrust in comparison to one or two 15 000 lbF engines operating at the same thrust level. Figure 17 also demonstrates that the installation of three of four smaller engines versus two 15 000 lbF engines for a fail-operational capability always results in higher performance during nominal operation.

The superiority of multiple engines for mission success and man rating is shown in figures 18 and 19.

Figure 20 summarizes the advantages of the Aerojet 3000 lbF thrust engine design concept, which is shown in figure 21. Photographs of test hardware that closely parallels the designs and technology required for this engine are shown in figures 22 to 24.

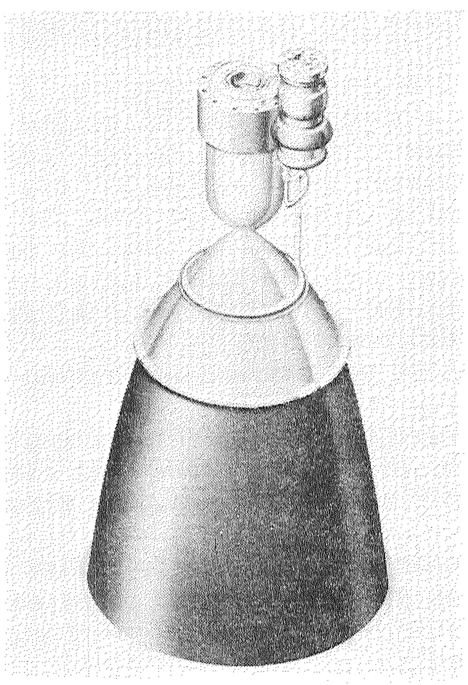
Aerojet believes that all OTV propulsion requirements can be fulfilled with a single engine. Our program is designed to develop the technologies required to demonstrate that engine.

ADVANCED OTV REQUIREMENTS

- SPACE-BASING
- AERO-ASSIST
- LOW G TRANSFERS
- MANNED MISSIONS
- LOW COST PAYLOAD DELIVERY

Figure 1

AEROJET'S NEW CORE ENGINE CAN



- 3000 lbF THRUST MODULES
- THRUST SELECTIVITY: 200-18,000 lbF

Figure 2

NEW FEATURES ENABLE SPACE-BASING

- NON-WEARING SEALS AND BEARINGS
- NO INTERPROPELLANT SEALS OR PURGES
- NO GEAR BOXES
- LOWER OPERATIONAL TEMPERATURES
- INTEGRATED HEALTH MONITORS
- SPACE-REPLACEABLE UNITS

REUSEABILITY

NO MAINTENANCE

Figure 3

DUAL PROPELLANT EXPANDER CYCLE DELIVERS

- LOWER OPERATING TEMPERATURES
- CONTINUOUS THROTTLING
- SMOOTH STARTS
- LESS WEAR
- LONGER LIFE

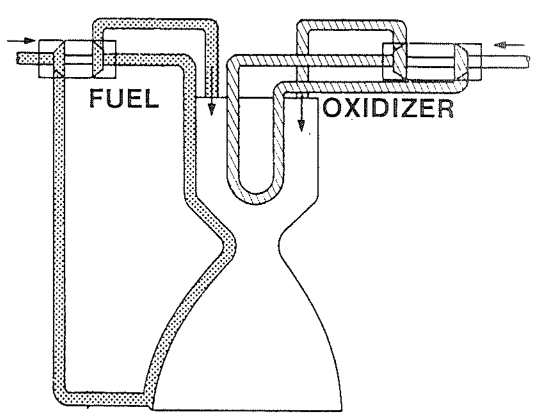
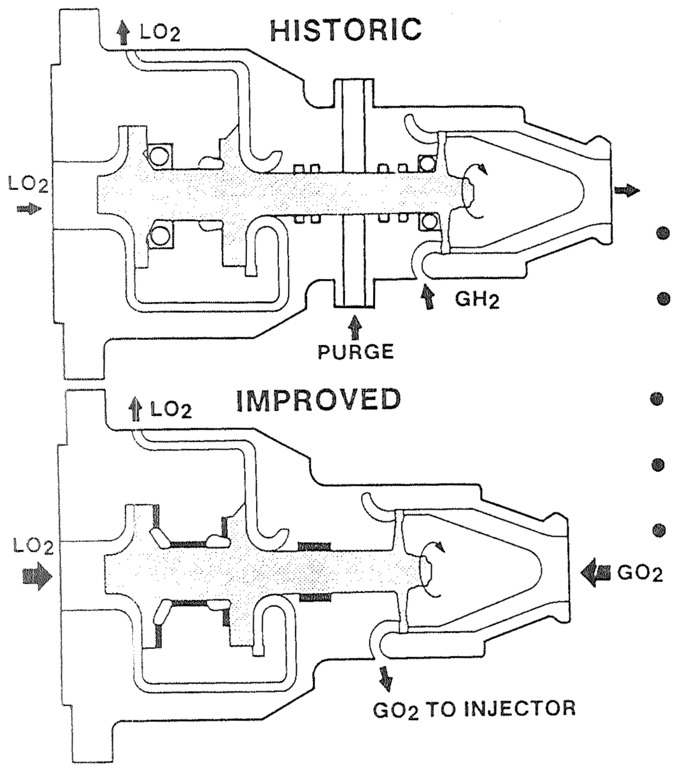


Figure 4

OXIDIZER PUMP ➡ ZERO MAINTENANCE



- CONTACTING BEARINGS
- INTERPROPELLANT SHAFT SEALS
- VENTED CAVITIES
- INTERPROPELLANT PURGES
- CRITICAL SPEED TRANSITION

Figure 5

LOW TEMPERATURES YIELD GREATER MARGIN

- TURBINE TEMPERATURE 400° F
- THROAT TEMPERATURE 600° F
- RADIATION SKIRT TEMPERATURE <2000° F

Figure 6

Burn Factor Provides A Ranking Criterion For The Selection Of Materials For High Pressure, Gaseous Oxygen Applications

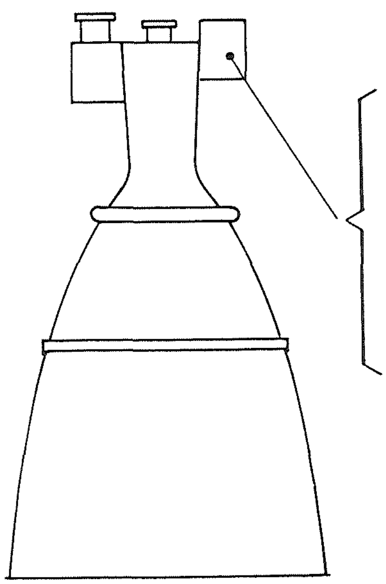
MATERIAL	BURN FACTOR	OBSERVATIONS
Zr Cu	35	NO IGNITION IN ANY TESTS > (790/1800°F)*
NICKEL 200	550	NO IGNITION WITHIN EXPERIMENTAL RANGE (825/2200°F)
SILICON CARBIDE	1145	NO IGNITION IN LIMITED TESTING
MONEL 400	1390	IGNITION ABOVE 1200°F FRT ONLY (800/1200°F)
K MONEL 500	2090	IGNITION ABOVE 1500 FRT (750/1500°F)
INCONEL 600	3226	IGNITION ABOVE 1100 (—/1100°F)
316 STAINLESS STEEL	4515	IGNITION IN ALL TESTS (450/800°F)
INVAR	5444	IGNITION IN ALL TESTS (675/340°F)
HASTELLOY X	7160	IGNITION IN ALL TESTS (725/750°F)

*(TEMPERATURES FROM PARTICLE INPINGEMENT TEST/FRICTION RUBBING TEST. (FRT))

*FRT AT 1000 PSI 17,000 RPM

Figure 7

INTEGRATED CONTROL AND HEALTH MONITORING



- CLOSED LOOP CONTROL
- DATA COLLECTION AND ANALYSIS
- DATA FEEDBACK TO CONTROLLER
  - PEAK PERFORMANCE
  - FAILURE PREVENTION
- LIFE PROJECTION

REDUCES OPERATIONAL COST

Figure 8

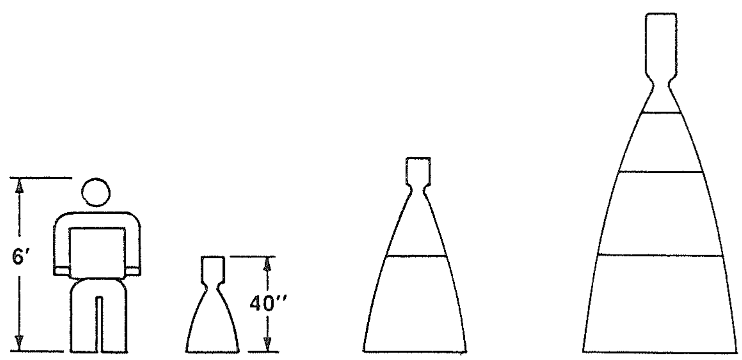
SMALL ENGINES FIT ALL AERO-ASSIST CONCEPTS

- SHORTER ENGINES ARE PREFERRED
- IDLE MODE OPERATION MAY BE REQUIRED

Figure 9

EQUAL PERFORMANCE IN A SMALLER PACKAGE

Pc = 2000
O/F = 6.0



THRUST, lbF	3000	3000	15,000
AREA RATIO	420	1500	1000
Isp $\frac{\text{lbF-sec}}{\text{lbm}}$	475	483	

Figure 10

LENGTH IS CRITICAL FOR ALL AEROMANEUVERING CONCEPTS

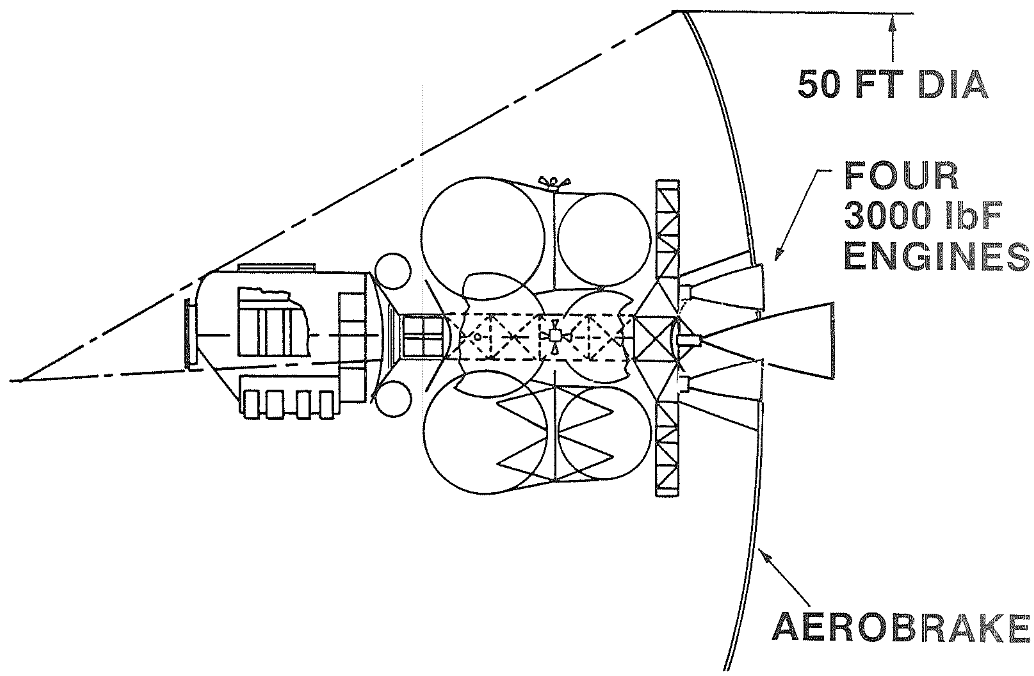


Figure 11

MINIMUM LENGTH AOTV

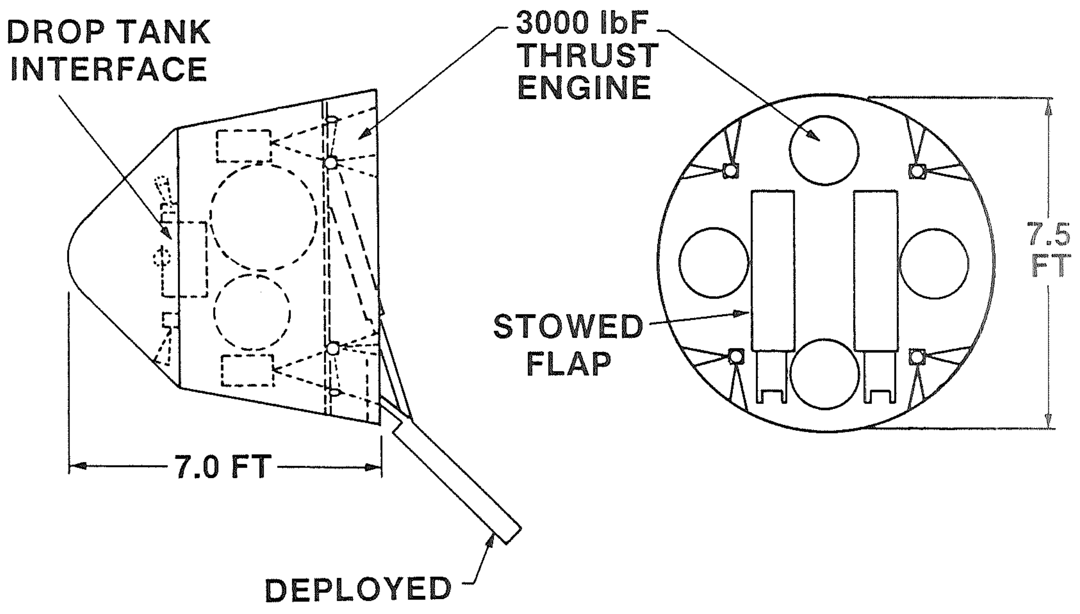


Figure 12

MULTIPLE ENGINES ALSO PROVIDE BOUNDARY LAYER CONTROL AND BOW SHOCK SKEWING

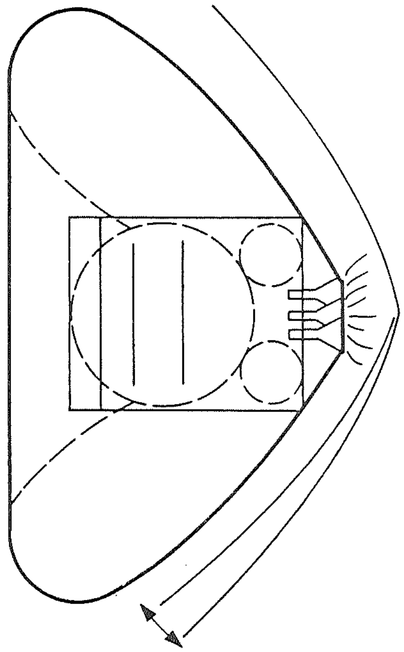


Figure 13

MAXIMUM WEIGHT DIFFERENCE FOR 6 ENGINES = 135 POUNDS

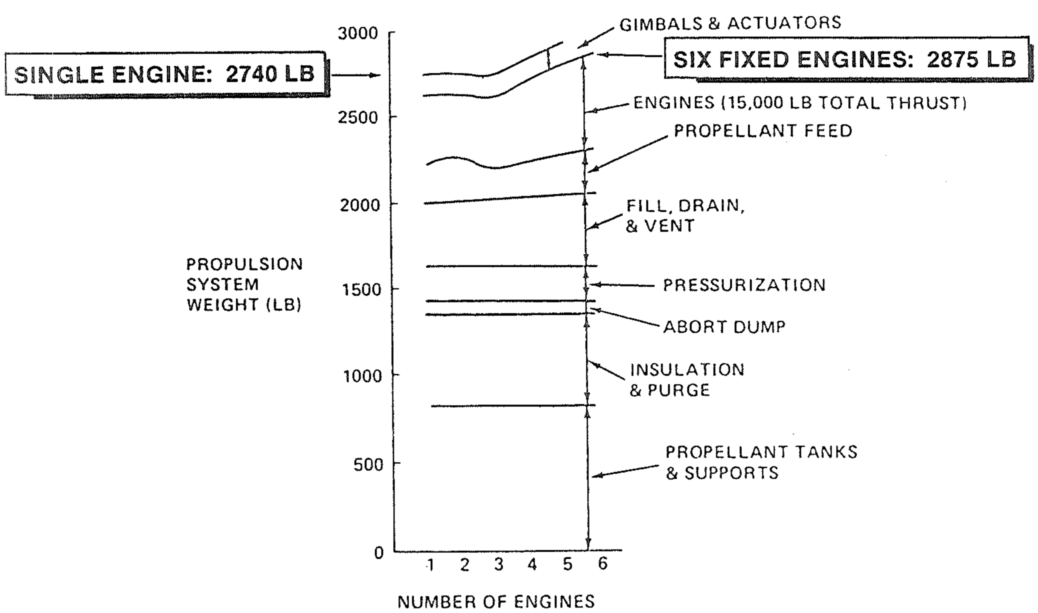


Figure 14

FOR LOW G TRANSFERS LOWER THRUST ENGINES OFFER

- SMALLER SIZE
- LOWER WEIGHT
- HIGHER PERFORMANCE
- SINGLE OR DUAL ENGINES
- THRUST SELECTIVITY 200 TO 3000 lbF

Figure 15

HIGH OPERATING PRESSURE → LOWER PROPELLANT VOLUME HIGHER PERFORMANCE

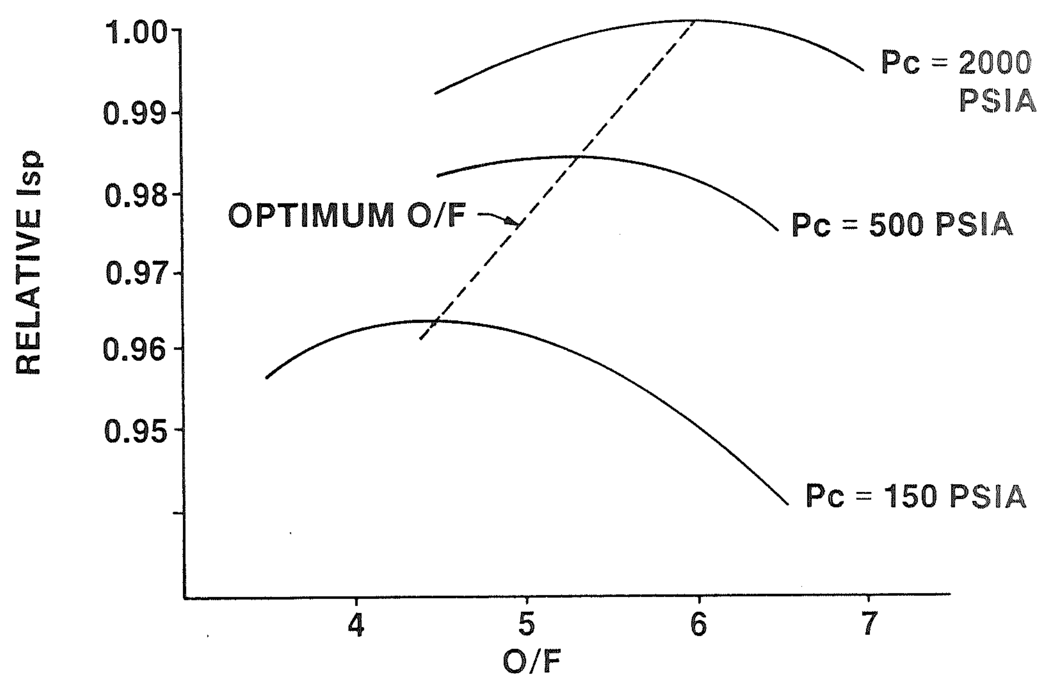


Figure 16

3000 LB THRUST MODULES YIELD HIGHER PERFORMANCE

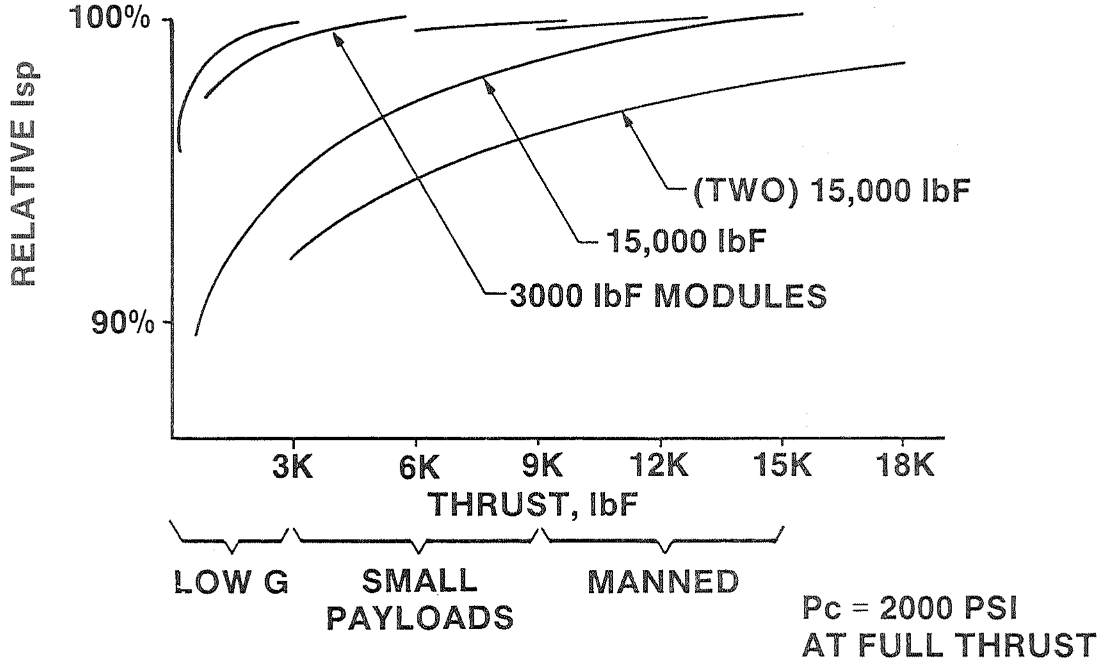


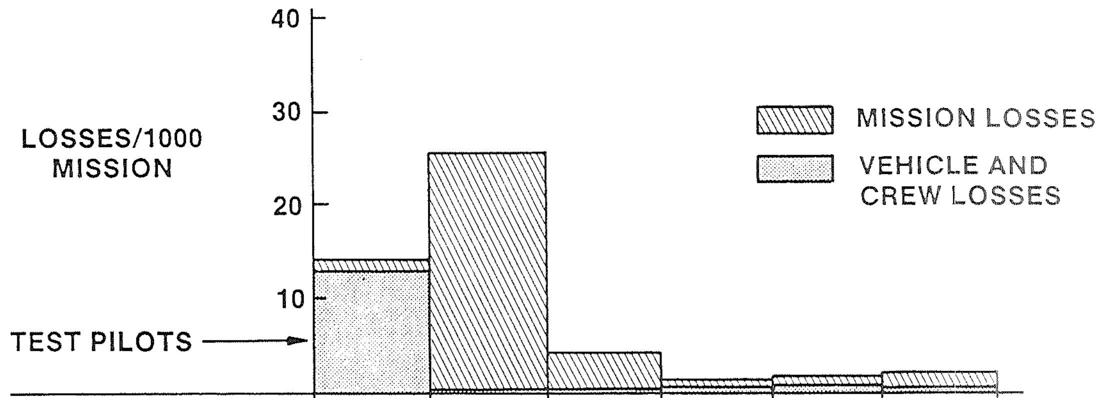
Figure 17

MULTIPLE ENGINES REQUIRED FOR MANNED MISSIONS

- MULTIPLE ENGINES =
  - ENGINE-OUT CAPABILITY
  - MISSION SUCCESS
  - MINIMIZES COMPONENT REDUNDANCY
  - MINIMIZES DEPENDENCE ON HEALTH MONITOR SYSTEM

Figure 18

MULTIPLE ENGINES ENHANCE MISSION SUCCESS AND CREW SAFETY



NO. OF ENGINES	1	2	2	3	4	6
NO. REQ'D FOR MISSION	1	2	2*/1	2	3	4
NO. REQ'D FOR SAFETY	1	1	1	1	2	2

*FIRST BURN

Figure 19

AEROJET APPROACH PROVIDES LOW COST PAYLOAD DELIVERY

- SMALL SIZE
  - LOWER DEVELOPMENT COST
  - LOWER UNIT COST
  - BETTER PACKAGING
  - HIGHER PERFORMANCE
- MODULAR APPROACH
  - OPTIMUM THRUST FOR ALL MISSIONS
  - ONLY ONE ENGINE DEVELOPMENT REQUIRED
- MULTIPLE ENGINES
  - MISSION SUCCESS

Figure 20

AEROJET'S NEW CORE ENGINE

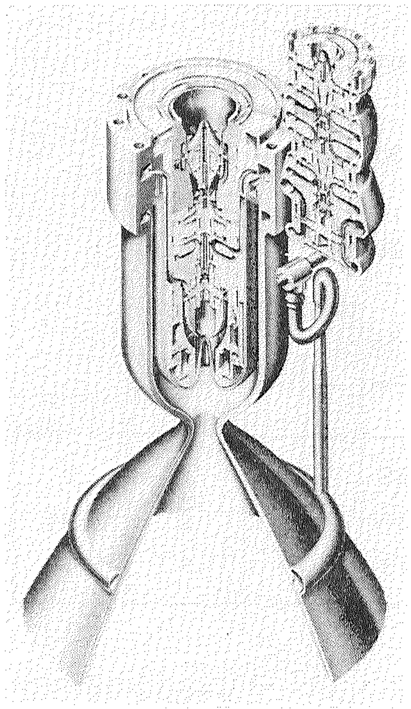


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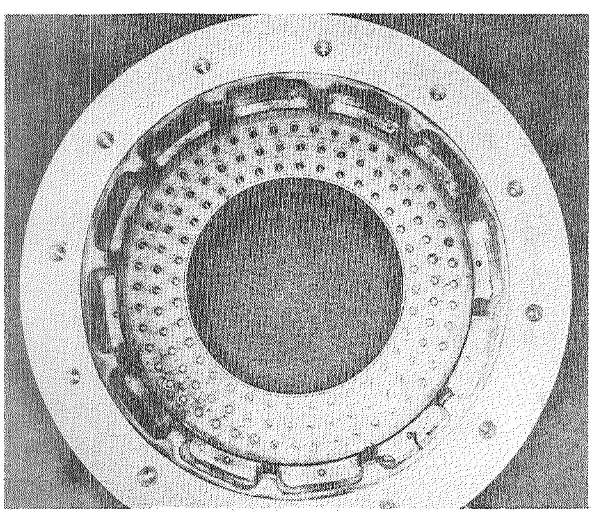
Figure 21

17 TESTS - GH_2/GO_2 ANNULAR TCA



- STABLE COMBUSTION
- HEAT TRANSFER AND PERFORMANCE DATA

C0184 007

Figure 22

CALORIMETER INNER CHAMBER

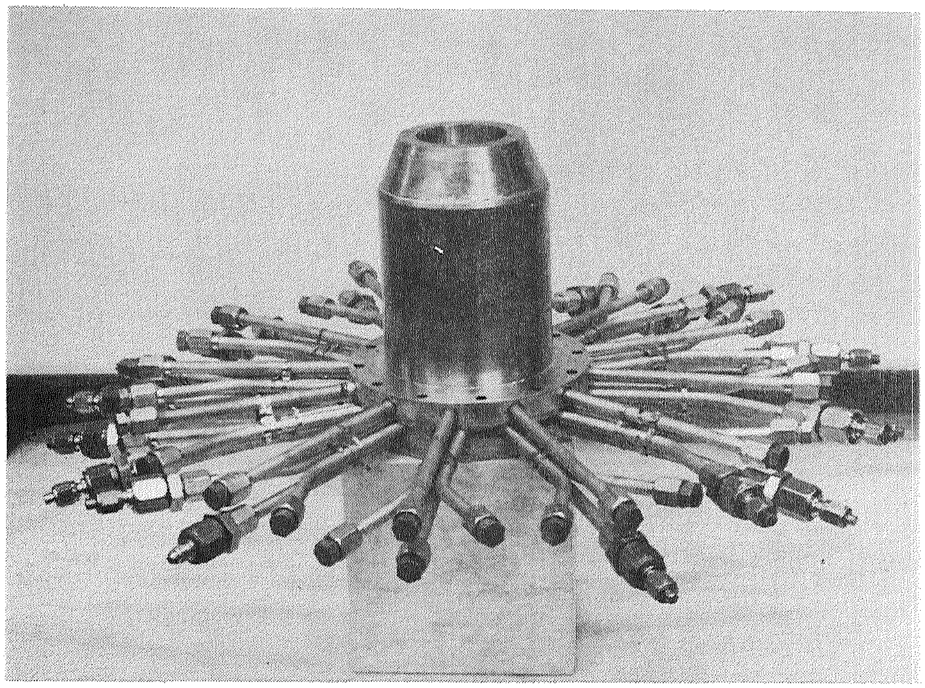


Figure 23

60,000 RPM LOW SPECIFIC SPEED PUMP. Ns = 700-1000

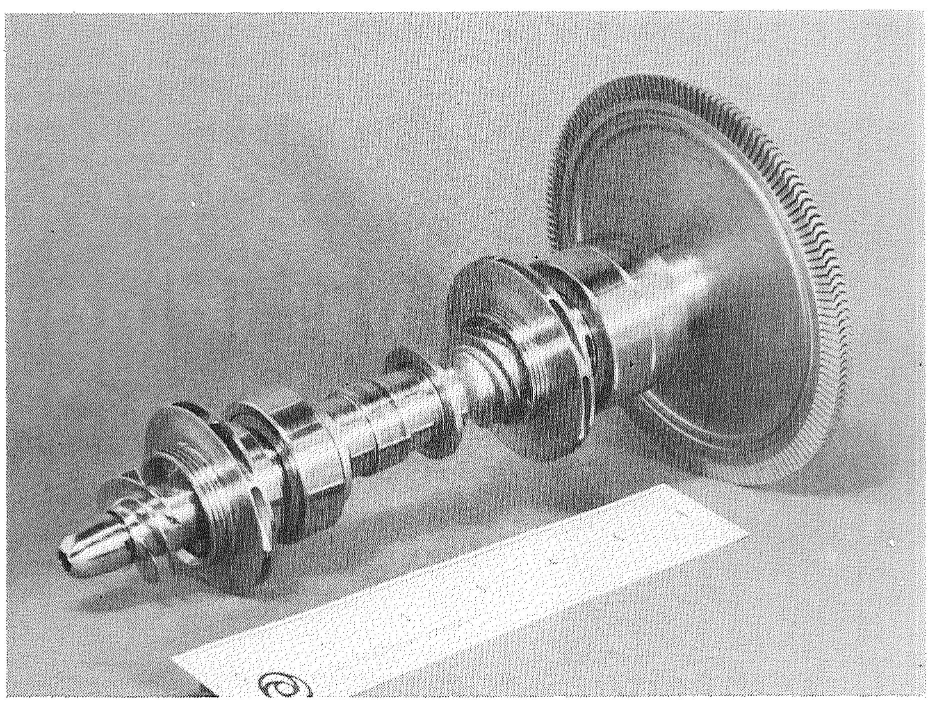


Figure 24

ADVANCED OTV ENGINES AND ISSUES

J. R. Brown
Pratt & Whitney Aircraft

Over the last decade Pratt & Whitney has studied the propulsion system requirements of Orbit Transfer Vehicles (OTV). Based on the current scenario for orbital operations in the late 1990's we have defined a baseline expander cycle engine which will meet those requirements.

The following presentation highlights the principal characteristics of our baseline engine and discusses some options which are available to accommodate OTV system optimization studies. A list of engine program issues are shown which are dependent on the mission scenario and the vehicle system configuration. Finally, a summary of the rationale for a new cryogenic OTV engine is given.

LATE 1990'S SCENARIO

- LEO space station with propellant depot
- Operational OMV
- Substantial LEO-GEO traffic
- Manned GEO sorties
- Reusable, cryogenic, aeroassisted OTV operational

Figure 1

REQUIREMENTS FOR OTV ENGINE

Must be compatible with:

- Space basing
- Aeroassist
- Man rating

Must have:

- High performance
- Long life
- High reliability
- Versatility
- Low operational cost

Figure 2

OTV ENGINE OPERATING MODES

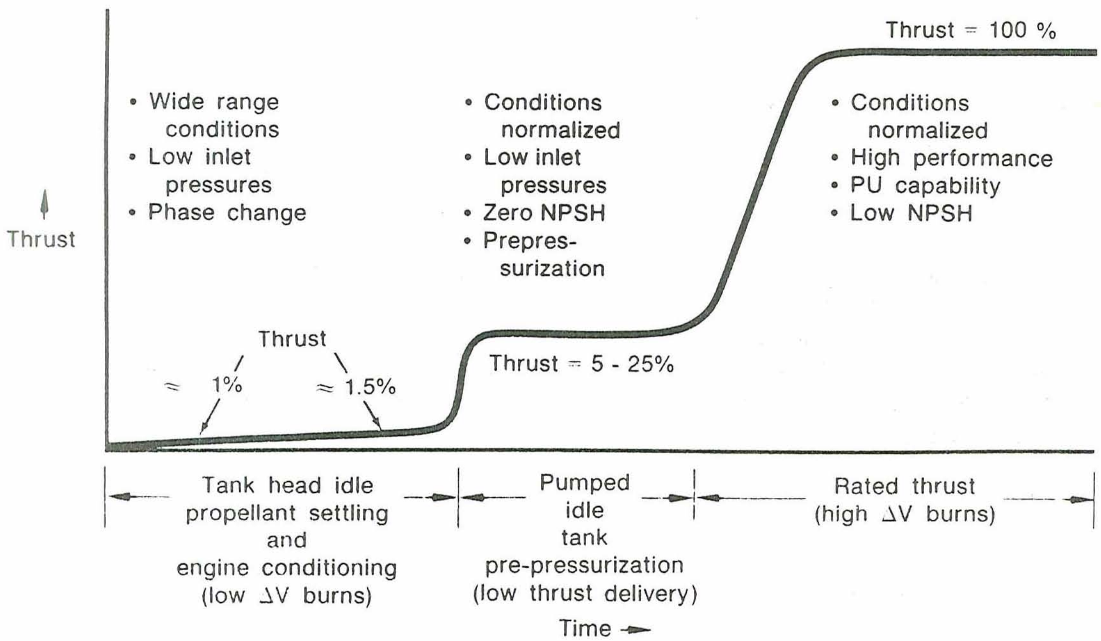
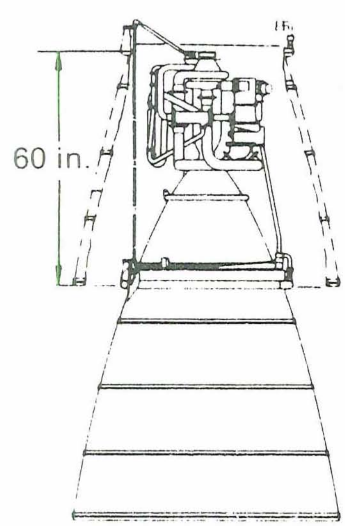


Figure 3

DESIGN CHARACTERISTICS

Advanced expander cycle engine



Thrust	: 15,000 lb
Mixture ratio	: 6.0:1 to 7.0:1
Chamber pressure	: 1500 psia
Area ratio	: 640
I_{sp}	: 482.0 sec at 6.0 MR
Operation	: Full thrust (low NPSH)
	: Pumped idle (1500 lb thrust) (saturated propellants)
Conditioning	: Tank head idle
Weight	: 427 lb
Life (design TBO)	: 300 firings/10 hr

Figure 4

INSTALLATION

Advanced expander cycle engine (1980)

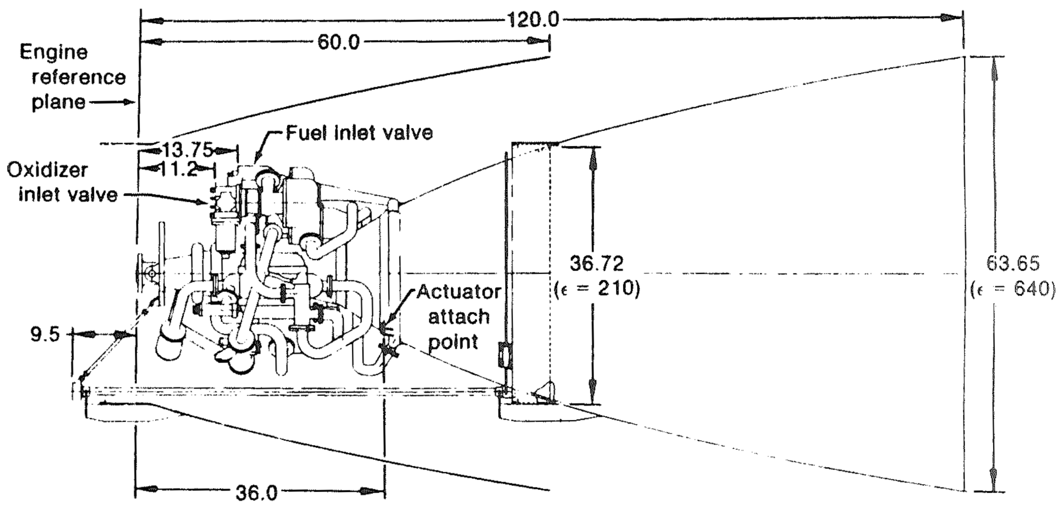


Figure 5

PROPELLANT FLOW SCHEMATIC

Advanced expander cycle engine at full thrust

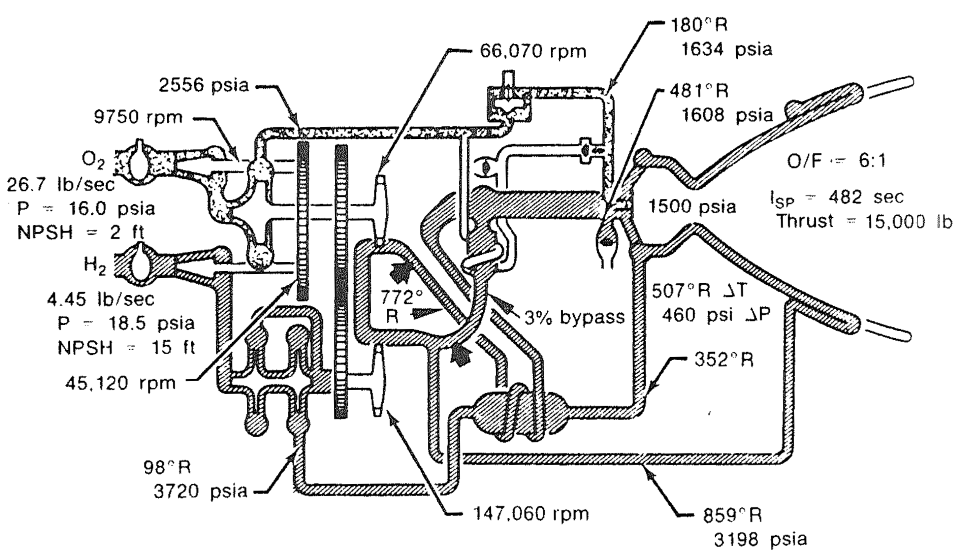


Figure 6

ADVANTAGES OF ADVANCED OTV ENGINE

- High reliability
  - Substantial design margins
  - Simple control system
- Adequate life
- Health monitoring relatively simple
- High performance
- Options available

Figure 7

ENGINE OPTIONS AVAILABLE

- Full thrust level
- Engine geometric size
- Throttling capability
- Mixture ratio range
- Special mission kits

Figure 8

FULL THRUST LEVEL AND SIZE

- Thrust levels evaluated: 500 to 30,000 lb
- Area ratios evaluated: 400 to 1,000
- I_{SP} (and size) proportional to thrust
- Optimum area ratio: 700 to 900
- DDT&E cost not significantly affected by thrust
- Limited applicability of technology for scaling

Figure 9

ADVANCED NOZZLE EXTENSION MECHANISM

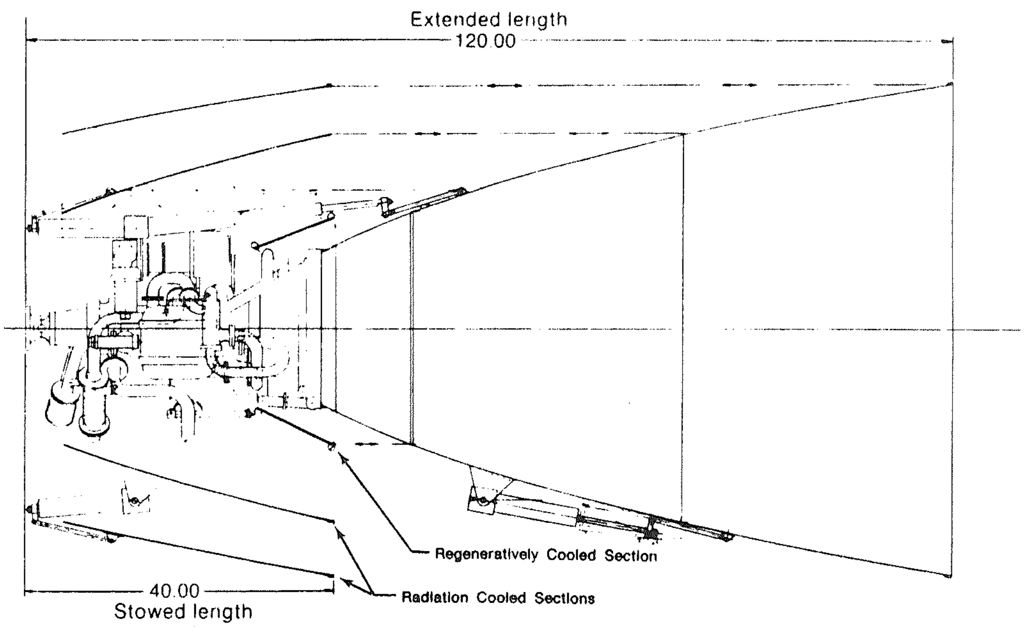


Figure 10

THROTTLING CAPABILITY

- Most missions
  - Full thrust for major burns
  - Very low thrust for trim burns (THI)
  - No requirement for intermediate levels
- Special G-level limited missions
  - "Continuous" throttling to hold max allowable T/W will yield higher average I_{SP}
  - Continuous throttling requires more complex engine system than a few discrete steps
- Potential throttled burn during aeroassist maneuver
  - Range and levels TBD

Figure 11

FULL THROTTLE RANGE CONCEPT

Continuous gaseous O_2 injection

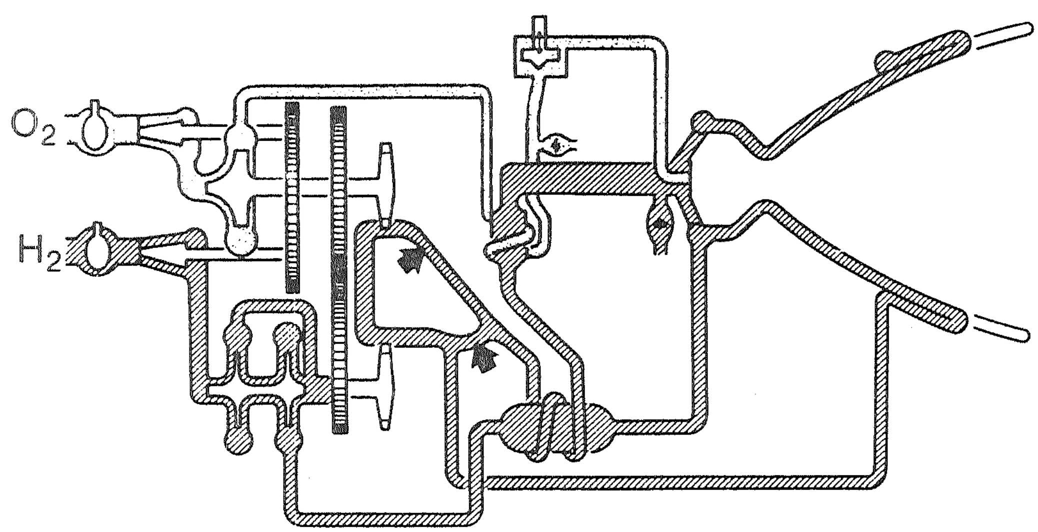


Figure 12

MIXTURE RATIO RANGE

- Optimum mixture range function of Pc (for $\epsilon \sim 750:1$)

  6:1 at Pc ~ 1000

  5:1 at Pc ~ 500

  4:1 at Pc ~ 50

- Off-nominal range required for vehicle considerations

  Boiloff

  Tanking uncertainty

  Special mission requirements

Figure 13

SPECIAL MISSION KITS

Examples

High thrust expendable mission

- Low thrust components might be eliminated
- Nozzle area ratio might be increased w/o aero assist

Low thrust expendable mission

- Control system might be mission specific
- Nozzle area ratio might be increased w/o aero assist

Figure 14

ADVANCED OTV ENGINE ISSUES

* • Engine thrust level(s)
  - Engine throttle requirements
* • Engine geometry constraints (including number of engines)
  - Engine operational life/servicing requirements
  - Basing mode
  - Engine/aeroassist mode interaction
  - Is "low thrust deployment" a viable OTV mission?
  - Is manned GEO mission viable?

Figure 15

SUMMARY

- A new cryogenic OTV engine will significantly enhance the capability of the OTV system in the following areas:

  Designed for space basing

  Designed for aeroassist compatibility

  Designed for man-rateability

  Designed for versatility/very high performance
- Commitment to DDT&E should be based on sound design, low risk approach which for an advanced technology engine includes key component demonstrations
- 6-10 year leadtime needed for high technology engine (depending on preceding component technology demonstration programs)
- OTV open issues exist
- Continuing iterations with NASA/systems contractors required to resolve issues and focus technology program

Figure 16

N85-17001 D12

ADVANCED OTV ENGINE CONCEPTS

A. T. Zachary
Rockwell International/Rocketdyne Division

Orbital transfer vehicles (OTVs) of the 1990 to 2000 period will deliver payloads for the more energetic of the NASA missions currently defined: large structure deployment, satellite servicing, and manned sorties to geosynchronous orbit. Along with advances in vehicle design, advances in engine technologies are required to improve overall engine capabilities, and thus vehicle performance, reliability, cycle fatigue life, maintainability, and cost. This paper briefly presents the results and status of NASA-LeRC-funded engine technology effort to date and related company-funded activities.

Advanced concepts in combustors and injectors, high-speed turbomachinery, controls, and high-area-ratio nozzles that package within a short length result in engines with specific impulse values 35 to 46 seconds higher than those now realized by operational systems. Equally, if not more important, will be the improvement in life, reliability, and maintainability.

INTRODUCTION

Studies conducted under NASA contracts have identified near-term, intermediate-term, and longer term technologies to meet the needs of a broad-based program of space utilization. As shown in figure 1, the evolution of the development process for the OTV leads to manrated service near the end of the century. The technology drivers in meeting the goals of a viable, space-based system are space basing, aeroassist, manned operation, and low-g transfers. As presented in figure 2, approaches have been identified; however, with specific challenges that must be met. These challenges of on-orbit servicing, increased life, reliability, maintainability, reduced length, and increased performance will be achieved through an evolutionary process as indicated in figure 3.

The NASA plan for technology acquisition for the orbit transfer rocket engines of the period 1990 to 2000 is a three-phase approach encompassing conceptual definition, preliminary experimental evaluation, and critical component technology verification stages. This approach is as follows, with the principal goals of the studies identified.

PHASE I Conceptual Designs and Technology Definition

- Identify, screen, evaluate, and select advanced technology concepts
- Provide engine conceptual designs and technology acquisition plans

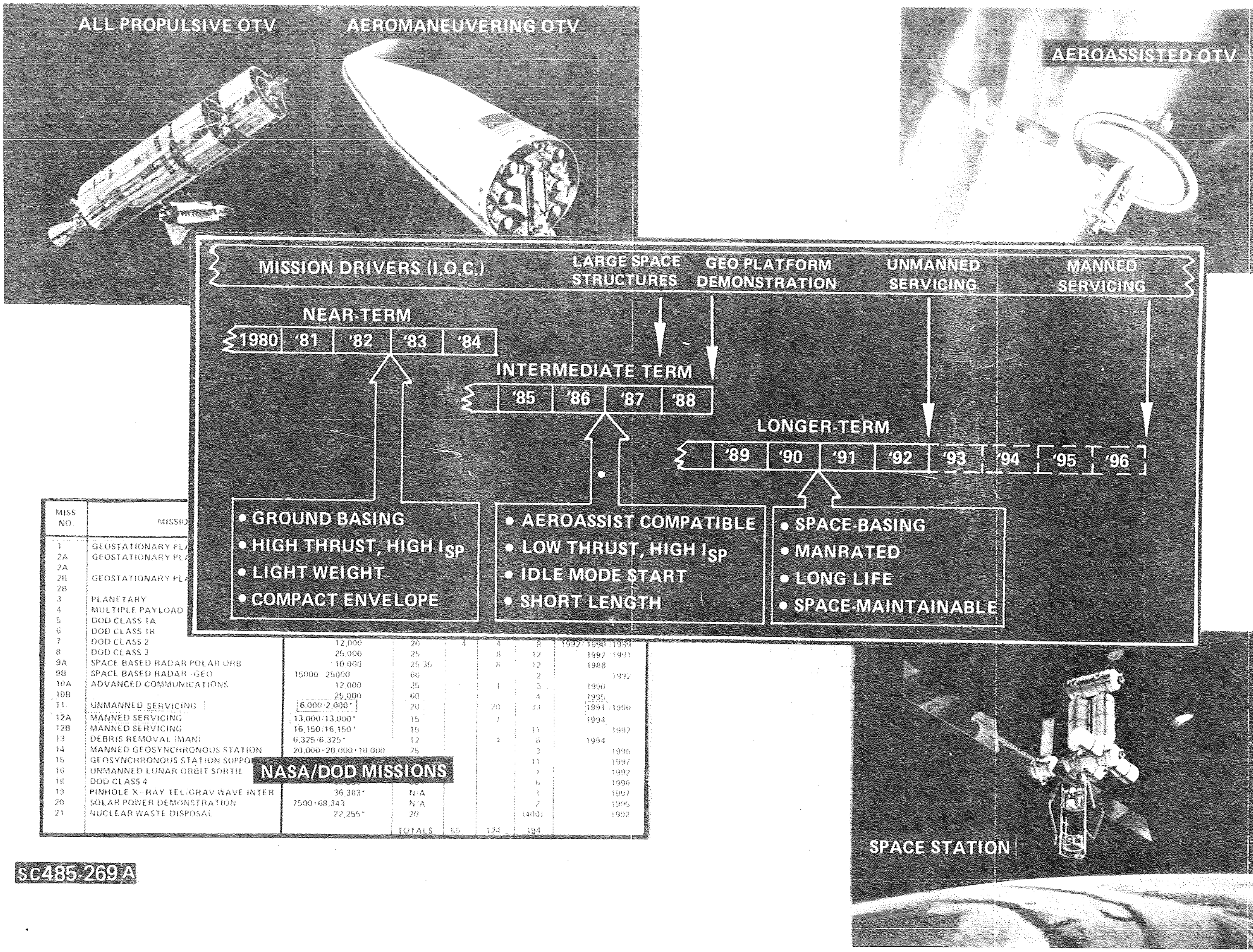


Figure 1. OTV Engine Technology Depth and Timing Drivers Background

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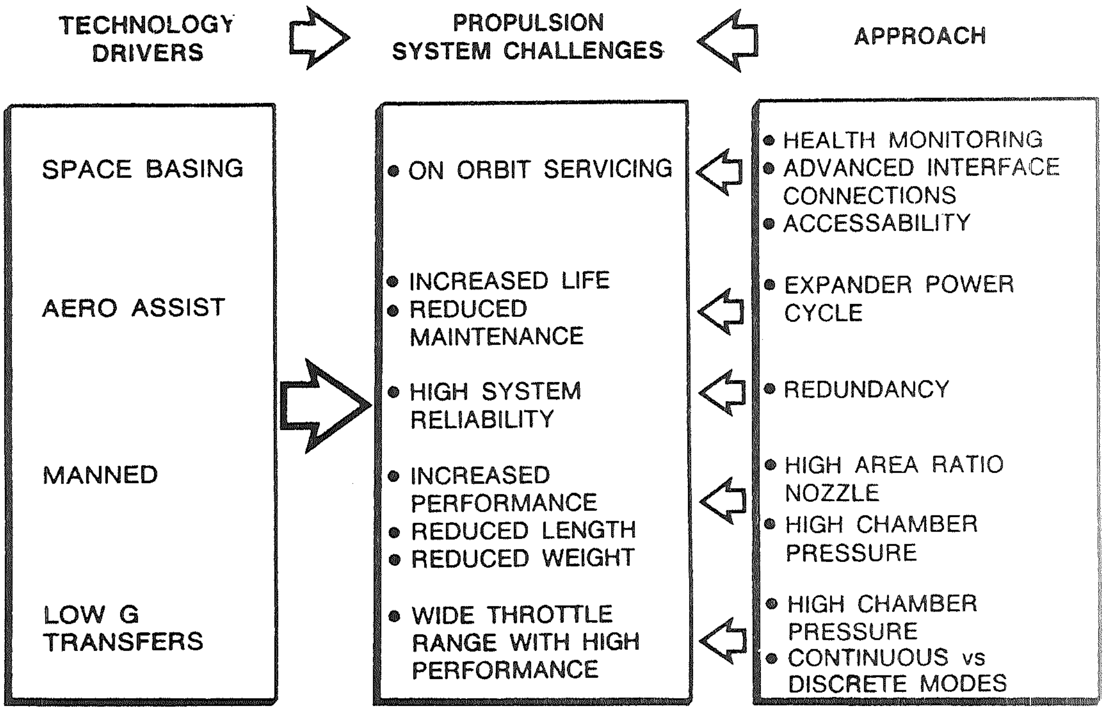


Figure 2. Engine Design Rationale

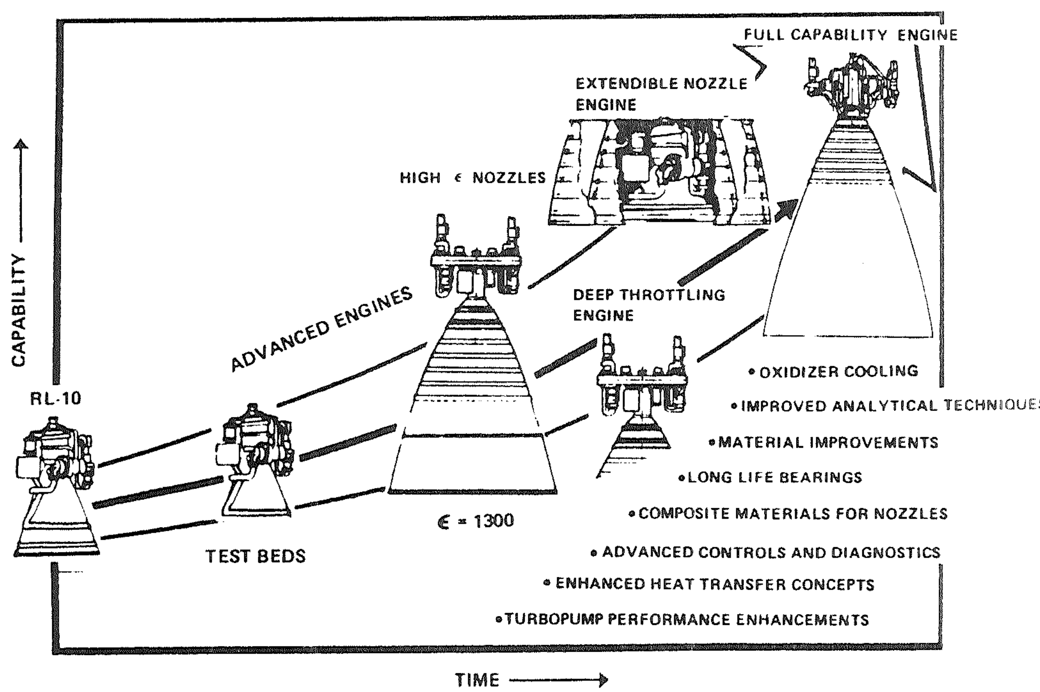


Figure 3. Cryogenic OTV Propulsion

PHASE II Exploratory Research and Technology

- Unique and generic advanced technology concepts
- Simulation testing in test rigs

PHASE III Critical Component Design and Fabrication

- Critical component design and fabrication readiness

An important first step in these plans has been taken with the completion of the conceptual design and technology definition studies entitled, "Orbit Transfer Rocket Engine Technology," with the primary objective of identification, and selection of advanced technology concepts and technology acquisition plans that will benefit the OTV engines of the 1990s.

ULTIMATE OTV ENGINE EVOLUTION

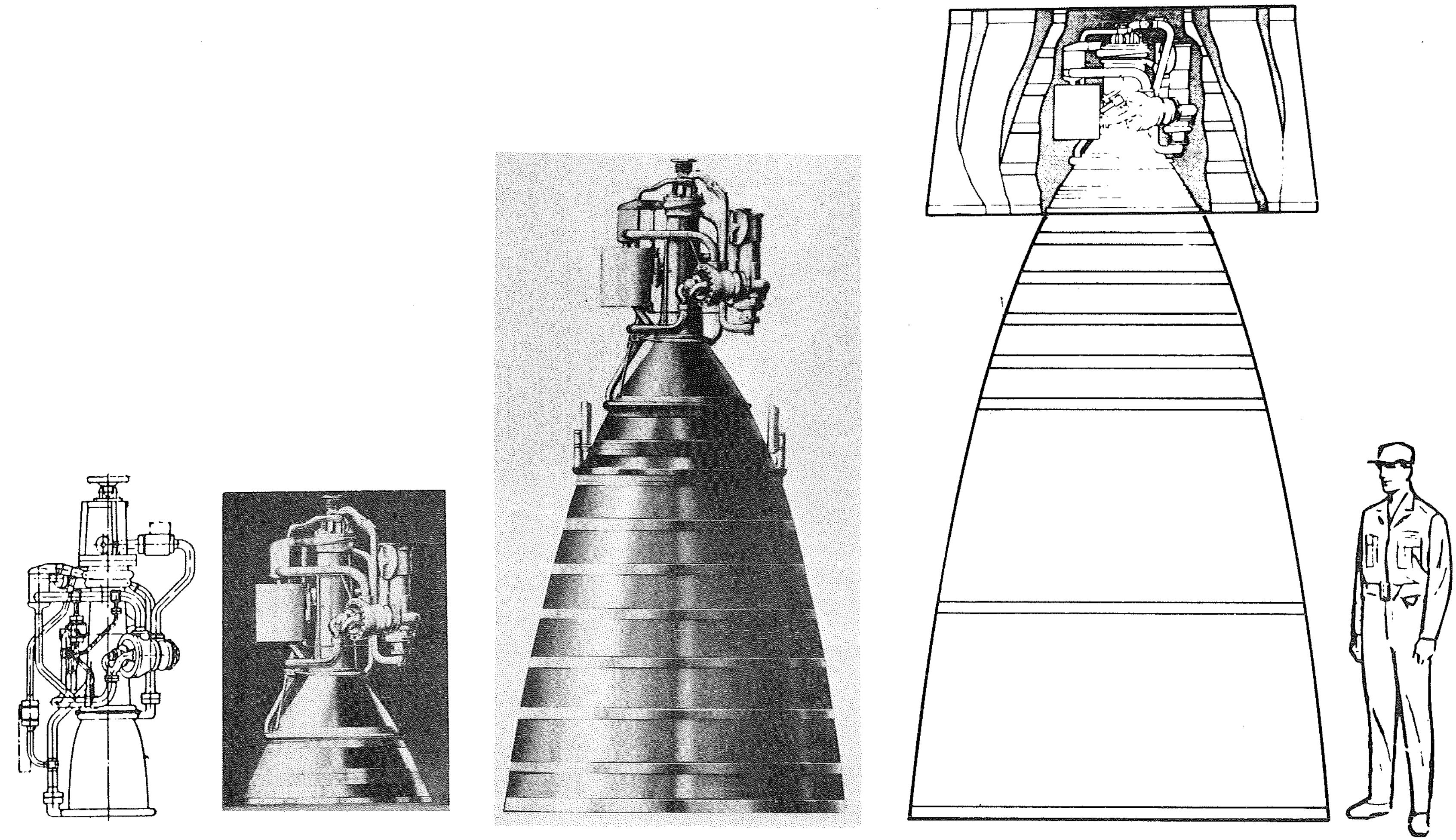
A phased approach has been selected for experimental development and verification of the technologies that will be featured in the ultimate OTV engine (fig. 4). The technologies will be evaluated in an integrated components evaluator upon which thrust chamber, turbomachinery, control system, and auxiliary system technologies will be verified in an engine system environment. The integrated components evaluator will facilitate the verification of advanced component concepts in three technology groupings: near-term, intermediate-term, and long-range categories, and their successive integration into advanced engine cores. At the completion of each technology period, an engine candidate and its technologies will have been defined that can be developed according to NASA needs. Each of these engines would provide large performance and operational benefits over the reference engine used in the studies (RL10A-3-3) and, because of the technology approach taken, could be developed as a growth version of the Advanced Core Engine. The near-term engine schematic and mockup are shown in figures 5 and 6, respectively.

The development of the ultimate engine would occur in the mid-1990s. The engine is planned as an expander cycle engine with a chamber pressure of 2000 psia, A nozzle expansion area ratio near 1300:1, and a specific impulse greater than 480 seconds. Operationally, the engine will be capable of 20 hours of service-free life, deep 30:1 throttling, and, with its health-monitoring and control system, capable of full space based maintenance and operation. For vehicles based in the Space Transportation System (STS) and for medium lift-to-drag aeromaneuvering OTVs, the engine will be fitted with a retractable nozzle to reduce stowed length to 40 inches.

The engine thrust level and number of engines will undergo final selection when the vehicle crew safety and reliability approach are definitized. Since a large degree of technology commality exists in the range of thrusts of 3000 to 15,000 pounds, Rocketdyne's interim selection of 15,000 pounds engine thrust is appropriate for technology development.

KEY ENGINE DESIGN ISSUES

Key issues of the engine system and component design reside in the combustor/injector, nozzle, turbomachinery, control system, and the auxiliary heat exchangers as outlined in figure 7. High heat extraction in the combustor, injector, and nozzle, with simultaneous efficient combustion and gas expansion, are required to provide high chamber pressure and high specific impulse. A combustor and injector with extended



●INTEGRATED COMPONENTS EVALUATOR

●ADVANCED ENGINE CORE

●NEAR-TERM ADVANCED OTV ENGINE

●ADVANCED OTV ENGINE (GROUND AND SPACE-BASED)

Figure 4. Ultimate OTV Engine Evolution

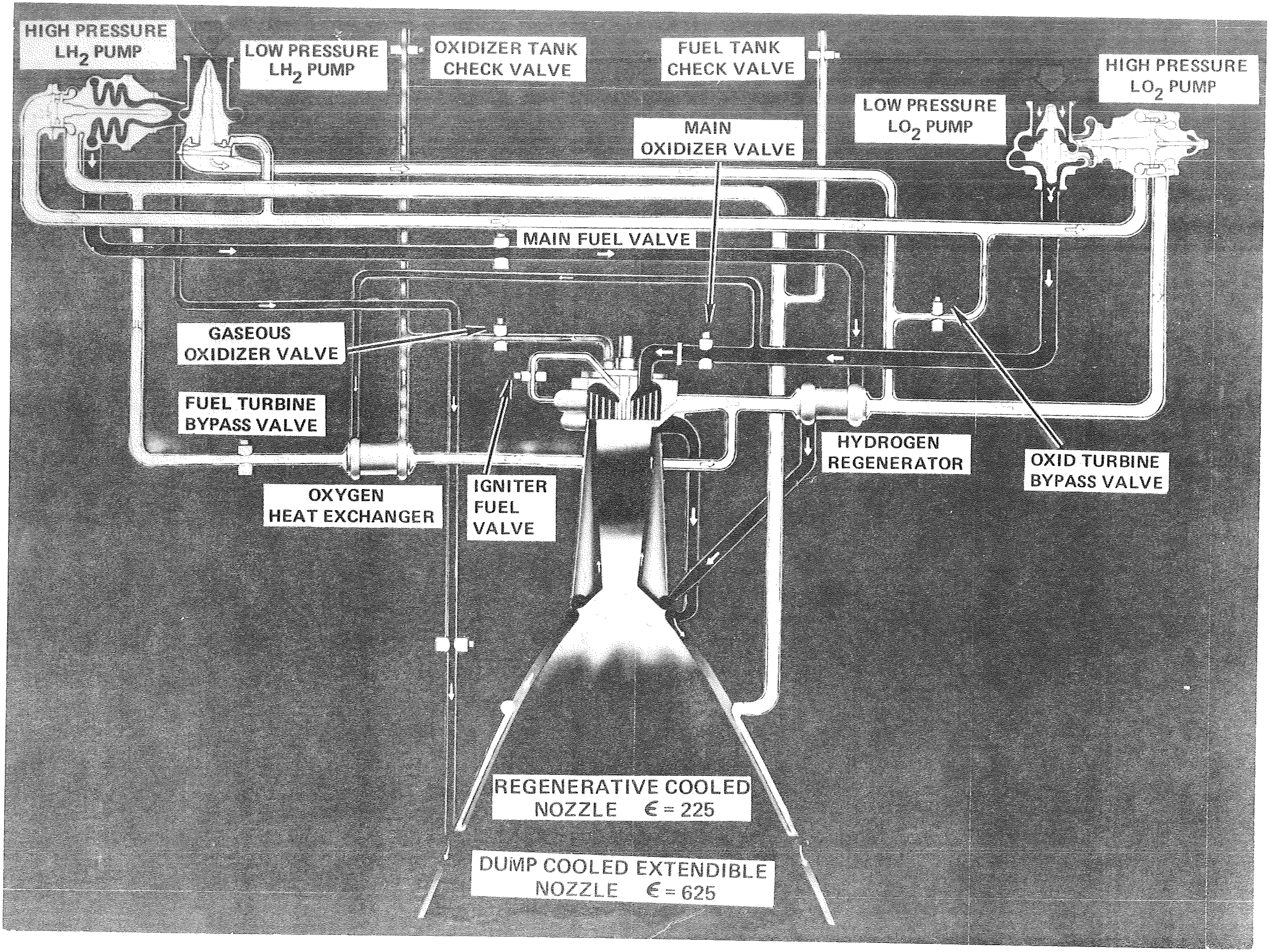


Figure 5. RS-44 Engine Schematic

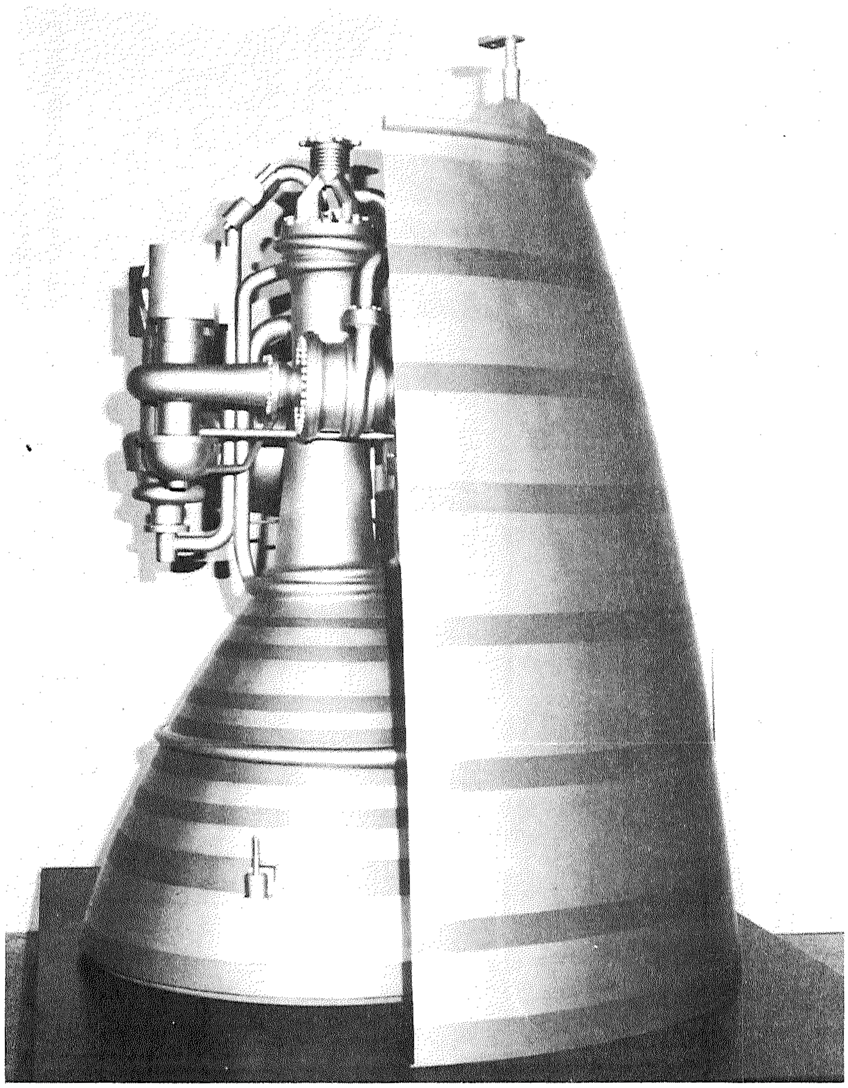


Figure 6. RS-44 Mockup

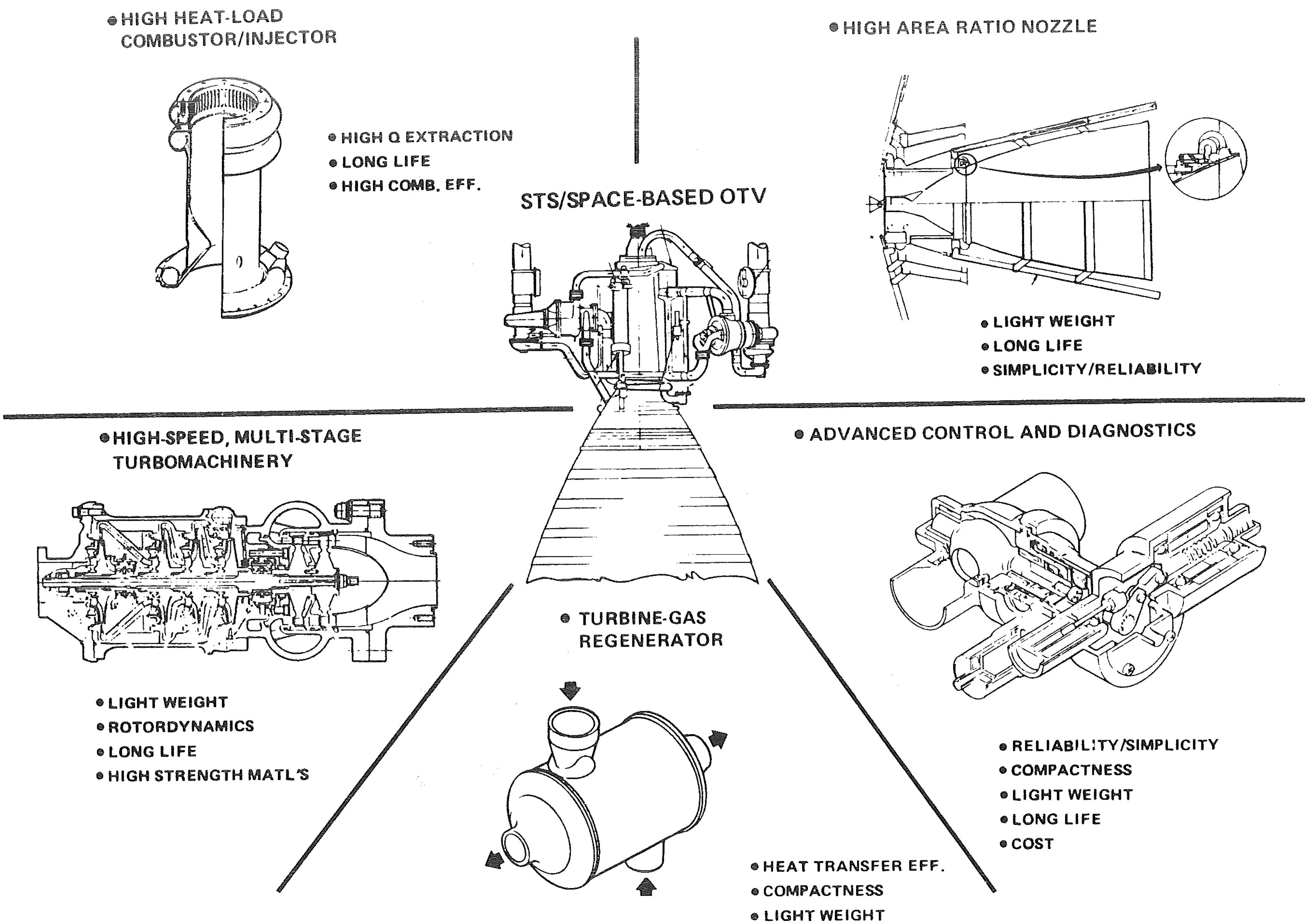


Figure 7. Ultimate Engine Design - Key Issues

heat transfer surfaces, providing high heat extraction, efficient wall cooling, and wall strain management to maintain desired component life, are the technology challenges in the combustor and injector. A large nozzle expansion area ratio with a retractable nozzle is necessary for high specific impulse and envelope compactness. Advanced material technologies and retraction mechanisms that reduce weight and yet provide adequate reliability are key technology issues of the nozzle assembly.

High speed, multiple staging, small size, and high turbine and pump efficiency are requirements of the OTV engine turbomachinery. The technologies and technology issues to be addressed in achieving the high levels of performance required in each of these areas are: bearing life; rotordynamic characteristics of multiple-staged impellers; materials for increased turbine and impeller strength, life, and reduced weight; and reduction of parasitic performance losses of small turbomachinery through use of soft seals and efficient diffuser design of impeller-to-impeller crossover networks.

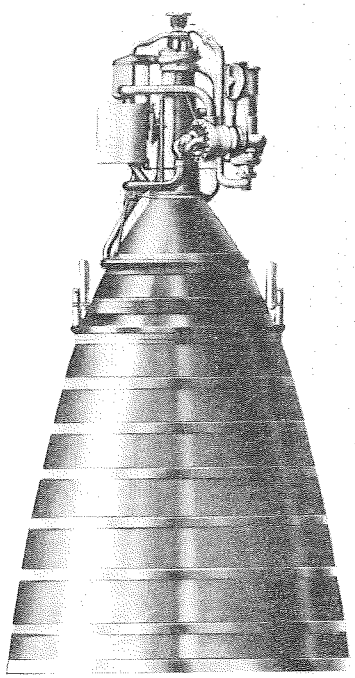
Low-torque, light-weight, electrically driven valves, and driver motors are technology issues to be developed for the advanced control system, as well as advanced sensor technology and advanced multivariable controller systems.

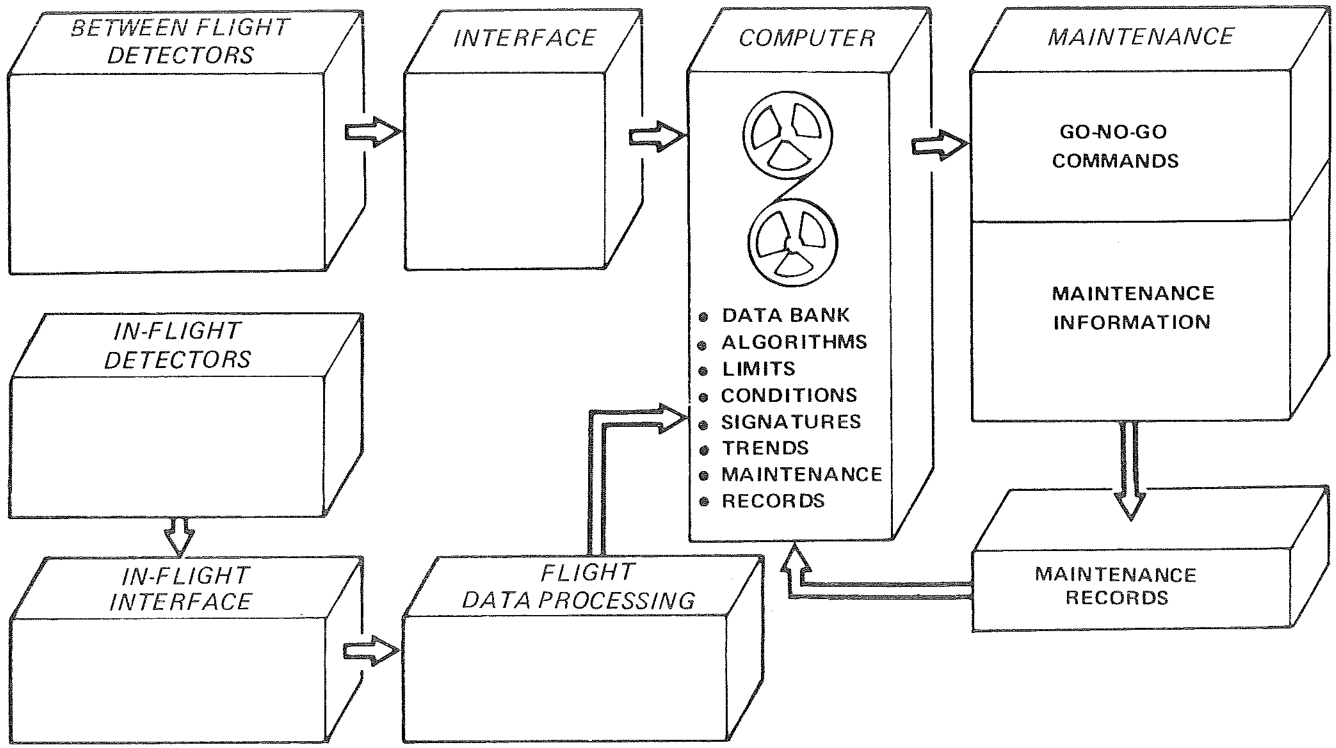
A turbine gas regenerator will provide increased power cycle performance through heat recuperation. For maximum benefits, the recuperator and the idle-mode heat exchanger will require high heat transfer efficiency in a compact envelope.

CONTROL SYSTEM TECHNOLOGY EVOLUTION

The near-term OTV engine shown in figure 4 uses a control and diagnostic system based on the current state of the art (SSME program) with one notable exemption: control valves are low-torque devices with an electric motor providing the primary means of actuation. Electrical power is desirable for upper-stage engines; however, low power requirements are necessary to keep the power supply small. The near-term OTV engine control system provides functions similar to the SSME system: control of engine operating modes, checkout and status monitoring, input/output data processing, and protection of engine and manrated capability. The controller is a full-range system providing closed-loop control of thrust and mixture ratio during mainstage, start, and shutdown transients. Control during transients is required to maintain component operating limits at levels compatible with the long life required in the near term (300 cycles, 10 hours). Redundancy in the controller, valves, and valve actuators is used to enhance the manrating capability of the system.

The longer range technology development of this system aims to improve control accuracy during transients, improve control system weight and simplicity, and improve control and diagnostic system reliability through improvements of the weakest link in the system: the sensors. Control accuracy improvement procedures will address modern multivariable control methodology and take advantage of modern miniaturization techniques for controller components. Emphasis of the long-range technology will be to provide a highly reliable control and diagnostics system specially suited for space-based OTV engine maintainability. The system will do continuous wear monitoring and fault prediction, and ideally be capable of fault compensation or avoidance. The diagnostic system is summarized in figure 8.





ACHIEVED BY USING A BETWEEN-FLIGHT AND/OR IN-FLIGHT CONDITION-MONITORING SYSTEM CONSISTING OF STATE-OF-THE-ART AND/OR NOVEL AUTOMATED DETECTION TECHNOLOGIES AND TAILORED DATA PROCESSING AND COMPUTERS.

Figure 8. Diagnostics for Maintainability Approach

ENGINE PACKAGING FOR SPACE-BASED MAINTAINABILITY

Several engine packaging concepts are shown in figure 9. Components in the near-term OTV engine are packaged to conserve space in a volume-limited shuttle orbiter. Power package components arranged around the combustor allow retraction of the extendible nozzle for stowage in the shuttle, and still provides required maintenance volumes for ground-based maintenance operations. The component interfaces of the near-term engine are designed with ground-based, line-replaceable unit philosophy.

For space-based operations, the overall maintenance, system and/or subsystem changeout philosophy will determine the engine component packaging arrangement and component interface design. Space-based maintenance costs and component changeout ease will determine the final maintenance philosophy. If, after economic analysis, engine changeout is the smallest maintenance module operation defined, then the near-term conventional engine packaging design with advanced engine/vehicle interface connections will be capable of space-based operation and maintenance.

The open-pack engine design will allow changeout of key engine components in a space environment. Increased maintenance volume will be defined for those components and a packaging design selected to facilitate component maintenance and changeout. An advanced control and diagnostics system will facilitate component changeout for cause rather than mandatory scheduled replacement.

In a space-based maintenance scenario where any or all components are subject to in-place maintenance and changeout, an advanced engine packaging configuration for efficient and speedy checkout removal and replacement will be desired. The components will be placed to facilitate access and their interface joints designed for minimum checkout and uncoupling time. A diagnostics system to facilitate judgement of components due for replacement and checkout of new components would then be required.

CONTRACT AND COMPANY-FUNDED ACTIVITIES

The current status of the NASA LeRC contract activities and Rocketdyne-funded parallel effort is outlined in Table I. With the completion of the system study to identify specific technology tasks to be studied, several of the tasks are now being funded or are in the process of approval and will be initiated shortly. The specific tasks in process are listed in Table II and encompass a number of critical technology areas in support of providing the technology base for a full capability engine in the 1990s.

In conjunction with the NASA funded effort, key technologies have been under study at Rocketdyne and are now entering the hardware stage for demonstration. The key elements are shown in figure 10, and represent an important learning process in design and fabrication of high performance engine components. It is planned to continue the evaluation of these components and thus develop a realistic perspective in the problems of developing a high performance, reliable, maintainable engine system.

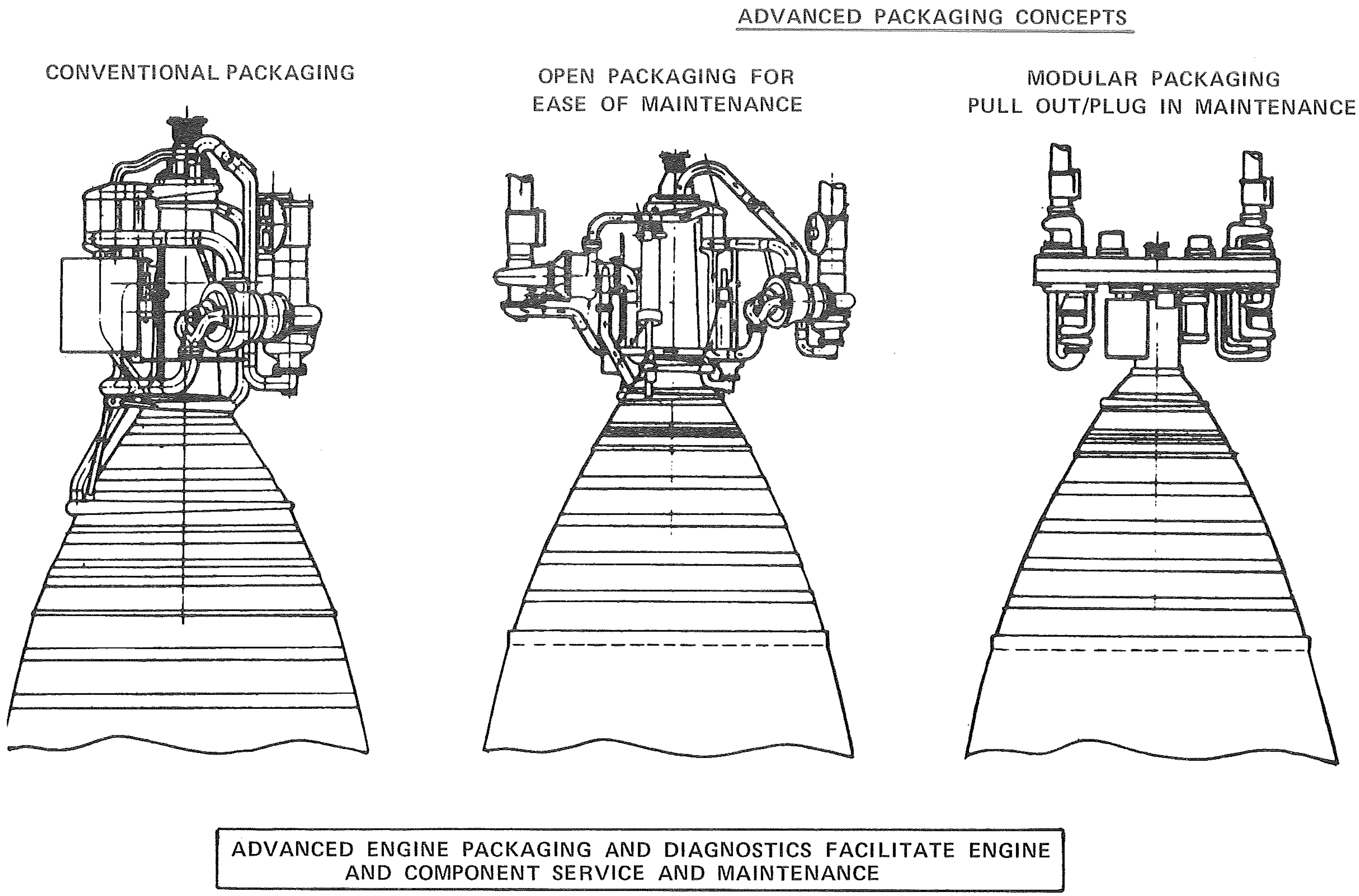


Figure 9. Engine Packaging for Space-Based Maintainability

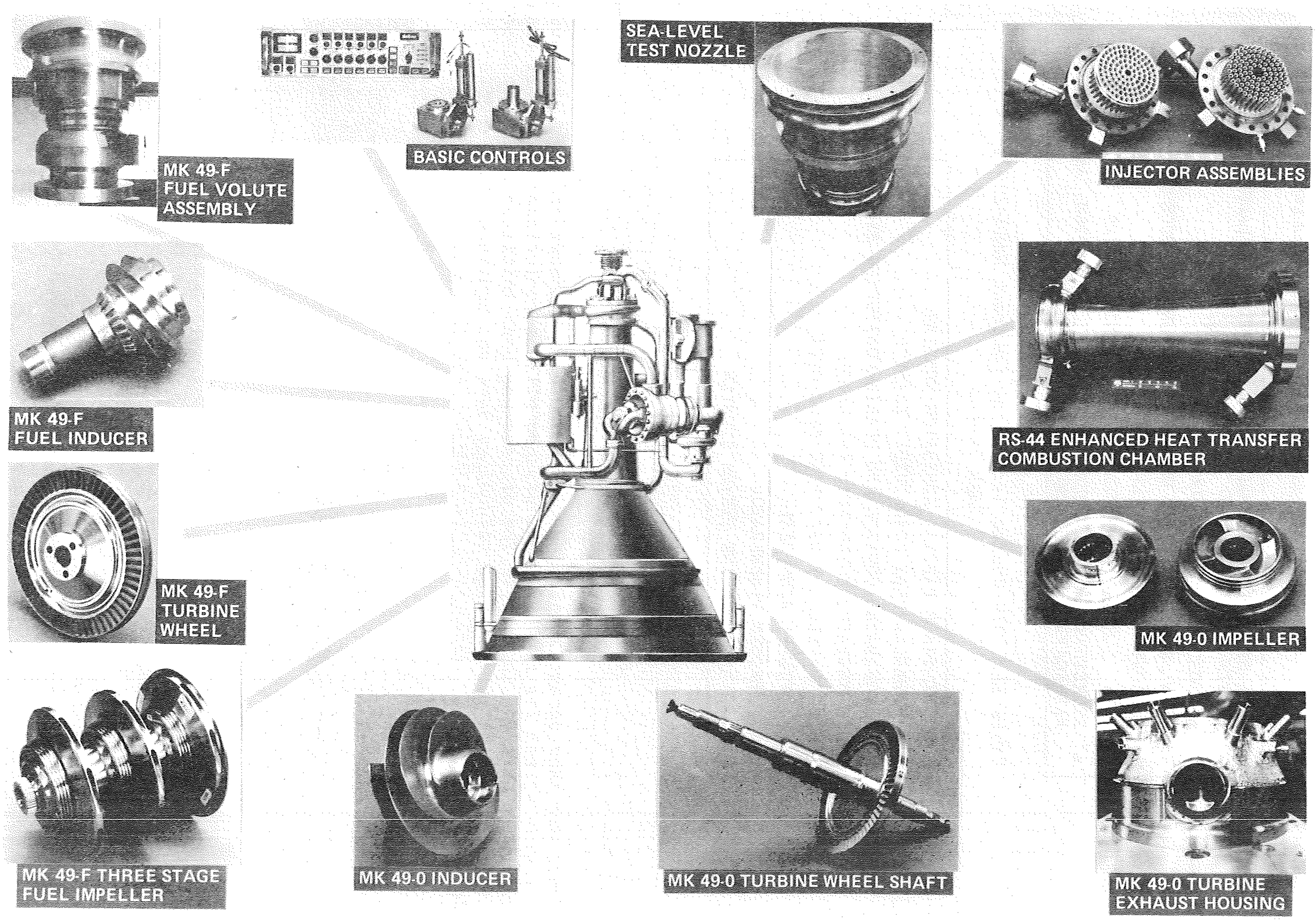


Figure 10. RS-44 Engine Test Bed Hardware

Table I. Contract and In-House OTV Engine Activities

- NASA OTV ROCKET ENGINE TECHNOLOGY PROGRAM PLANNED TO CONTINUE THROUGH 1990
  - CONTRACT NAS3-23172 COMPLETED
    - SYSTEM STUDY - DEFINITION OF NEEDED TECHNOLOGY
  - CONTRACT NAS3-23733 STARTED
    - COMPONENT TECHNOLOGY AND SYSTEM STUDIES
- ROCKETDYNE IN-HOUSE COMPONENT ANALYSIS, DESIGN, AND FABRICATION PROCEEDING ON SCHEDULE

Table II. OTV Engine Technology Contract NAS3-23773 Task Status

- TWO-STAGE PARTIAL ADMISSION TURBINE
- ENHANCED HEAT LOAD THRUST CHAMBER
- INTEGRATED CONTROL AND HEALTH-MONITORING SYSTEM
- INTEGRATED COMPONENTS EVALUATOR
- HIGH VELOCITY DIFFUSING CROSSOVER
- SOFTWARE RING SEALS
- HYDROGEN REGENERATOR
- ADVANCED ENGINE SYSTEM STUDIES

CONCLUDING REMARKS

There are many potential problems in producing a high-performance, space-viable rocket engine system for the OTV. The process will be evolutionary and will require the support of NASA and Industry. The process has been initiated with a well-ordered plan for establishing the required technology base and continued future effort should be strongly supported.

N85-17002

EARTH-TO-ORBIT PROPELLANT TRANSPORTATION OVERVIEW

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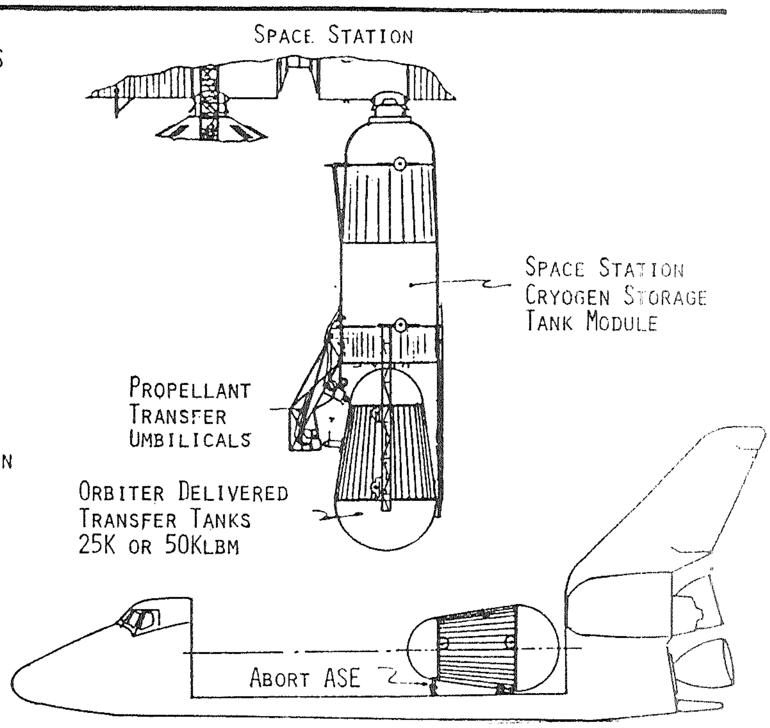
Large quantities of cryogenic propellants are needed to support Space Station/OTV operations. Two ways to get propellants into space are: transporting them in dedicated tankers or scavenging unused STS propellant (which promises significant cost savings). This discussion centers on scavenging propellant, both with and without an aft cargo carrier system. An average of two to four flights per year can be saved by scavenging and manifesting propellant as payload. Addition of an aft cargo carrier permits loading closer to maximum, reduces the required number of flights, and reduces the propellant available for scavenging. Sufficient propellant remains for OTV needs, however.

CRYOGEN PROPELLANT SUPPLY - DEDICATED TANKER

- 70,000 LBM STORAGE ON SS
- RESUPPLIED BY STS
- TWO RESUPPLY TANKS TO AID PAYLOAD SCHEDULING
- COMMON I/F FOR SS STORAGE TANKS AND STS ASE (ABORT DUMP)
- STS TANKS TO USE PRESSURE REGULATED TRANSFER
- SS TANKS TO USE AUTOGENOUS PRESSURIZATION

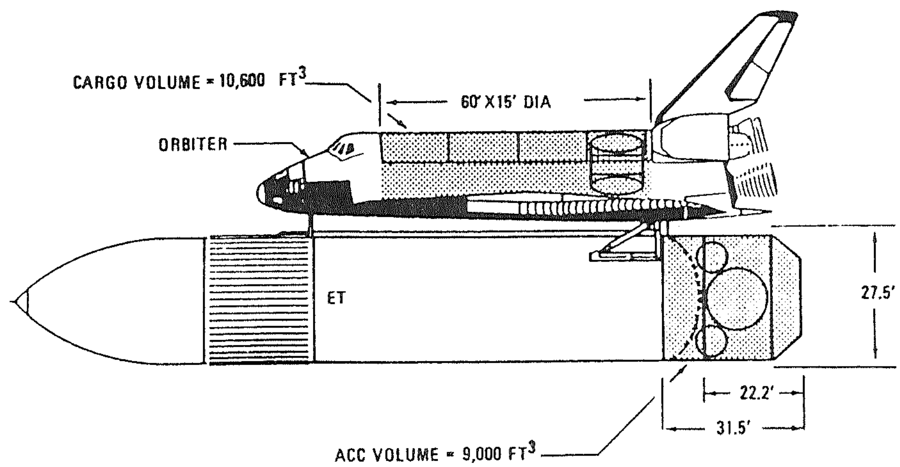


ATTACHED, FREE-FLYING, OR TETHERED DEPOTS ARE BEING CONSIDERED FOR STORAGE OF CRYOGENIC PROPELLANTS AT THE SPACE STATION. ALL HAVE SIMILAR STORAGE REQUIREMENTS. QUANTITIES OF PROPELLANT RANGE FROM 70,000 TO 100,000 LBM STORAGE.

EARLY ON, WE LOOKED AT TWO RESUPPLY TANKS OF 25,000 AND 50,000 LBM CAPACITY TO AID PAYLOAD MANIFESTING. THESE TANKS WOULD BE DELIVERED BY THE ORBITER AND WOULD HAVE A COMMON STS (ABORT DUMP) AND SPACE STATION STORAGE DEPOT INTERFACE. THE RESUPPLY TANKS WOULD USE PRESSURE REGULATED TRANSFER WHILE THE SPACE STATION TANKS WOULD USE AUTOGENOUS PRESSURIZATION TO ASSIST IN REFILLING.

Figure 1

Alternate Locations for Propellant Scavenging Tanks



TWO LOCATIONS ARE SHOWN FOR SCAVENGING TANKS: IN THE ORBITER PAYLOAD BAY AND IN THE ACC. AS WE WILL SEE LATER, VERY FEW OF THE FLIGHTS ARE WEIGHT LIMITED; MANY ARE VOLUME LIMITED. LOCATING SCAVENGING TANKS IN THE PAYLOAD BAY, THEN, DOES NOT APPEAR ATTRACTIVE. IN FACT, MORE CARGO VOLUME IS NEEDED IN MANY INSTANCES, AND THAT IS THE REAL JUSTIFICATION FOR AN ACC -- NOT SCAVENGING. IF AN ACC IS AVAILABLE, THE SCAVENGE TANKS WOULD BE LOCATED THERE.

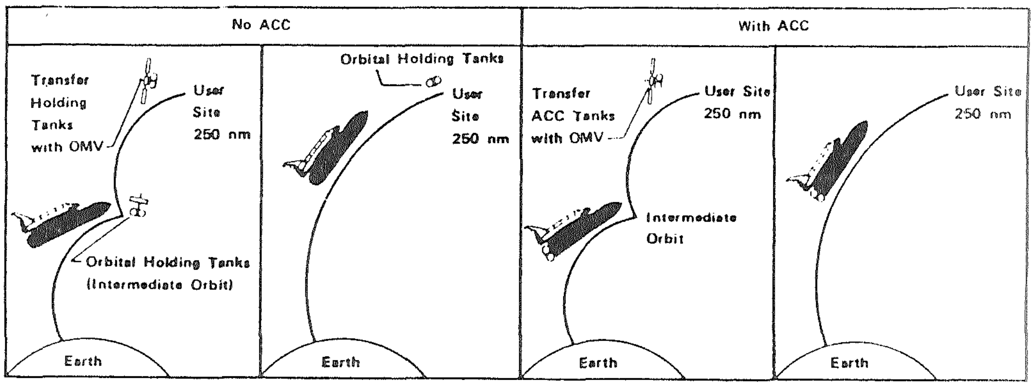
Figure 2

GROUND RULES

1. NASA LOW MISSION MODEL - REVISION G, DATED OCTOBER 1982 (1991 - 2000)
2. WTR AND DOD MISSIONS NOT USED
3. MISSION MODEL MANIFESTED ON GROSS YEARLY BASIS - LOAD FACTORS OBTAINED FROM MSFC MODEL BASED UPON WEIGHT AND/OR VOLUME
4. ALL FLIGHTS LOADED TO MAXIMUM PAYLOAD WEIGHT
5. DIFFERENCE BETWEEN AVERAGE CARGO WEIGHT AND MAXIMUM PAYLOAD CAPABILITY MADE UP BY ADDING PROPELLANT AND TANKAGE AS CARGO
6. SPACE STATION AT 28.5° INCLINATION, 250 NM ALTITUDE
7. OTV PROPELLANT REQUIREMENTS FOR GEO MISSIONS ONLY

Figure 3

Mission Concepts

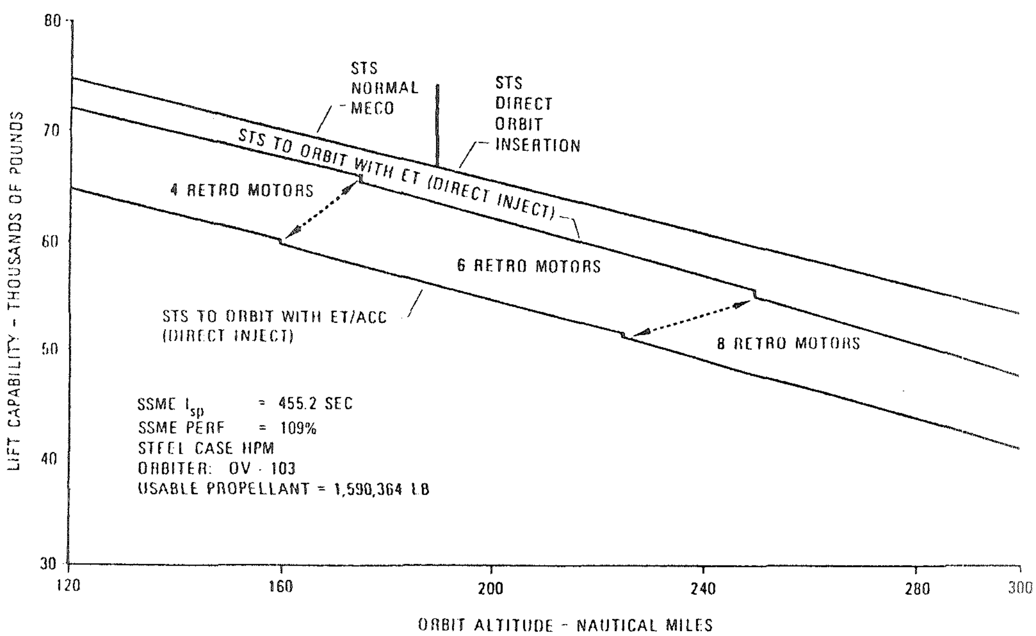


SCENARIOS WITH AND WITHOUT AN ACC ARE SHOWN. WITH NO ACC, ORBITAL HOLDING TANKS ARE FILLED AT LEO AND TRANSFERRED TO THE USER SITE WITH AN OMV. THE SAME IS TRUE OF THE SCAVENGING TANKS WITH AN ACC.

IN ALL CASES, THE ET IS TAKEN TO ORBIT, EITHER TO LEO OR TO THE SPACE STATION (DIRECT INJECTION), TO PROVIDE SUFFICIENT TIME (HOURS) FOR SCAVENGING. ON A DIRECT ORBIT INSERTION FLIGHT WITH THE ET DROPPED INTO THE OCEAN, INSUFFICIENT TIME EXISTS FOR OMV RENDEZVOUS AND DOCKING.

Figure 4

STS Lift Capability for 28.5 Degree Inclination



FOR THE STS CRITERIA LISTED, LIFT CAPABILITY IS SHOWN AS A FUNCTION OF ORBIT ALTITUDE. THE LOWER CURVES ARE FOR THE CASES OF STS AND ET TO ORBIT (DIRECT INJECT) WITH AND WITHOUT AN ACC.

WITH THIS SCENARIO, A SYSTEM IS REQUIRED TO DE-ORBIT THE ET FOLLOWING PROPELLANT SCAVENGING. SOLID MOTORS WITH HYDRAZINE ATTITUDE CONTROL WOULD BE USED. MORE RETRO MOTORS WOULD BE REQUIRED AT HIGHER ORBITS, ACCOUNTING FOR THE SMALL STEP DECREMENTS IN LIFT CAPABILITY.

Figure 5

Propellant Scavenging Definitions

RESIDUAL PROPELLANT	=	UNUSABLE + FLIGHT PERFORMANCE RESERVE + FUEL BIAS
SURPLUS PROPELLANT	=	PROPELLANT LEFT AT MECO MINUS RESIDUAL (NOT USED DUE TO LESS THAN MAXIMUM PAYLOAD)
PAYLOAD PROPELLANT	=	"SURPLUS" LOADED INTO SCAVENGING TANKS BEFORE LAUNCH

ACC ENVIRONMENT ASSUMED EQUIVALENT TO ORBITER CARGO BAY

SINCE THE ET IS ALWAYS LOADED FULL, SURPLUS PROPELLANT EXISTS WITH LESS THAN MAXIMUM PAYLOAD. SOME OF THIS SURPLUS PROPELLANT COULD BE LOADED INTO SCAVENGING TANKS BEFORE LAUNCH IF A CHANGE IN LOADING PHILOSOPHY OCCURRED.

THE ACC ENVIRONMENT IS ASSUMED EQUIVALENT TO THE ORBITER CARGO BAY. THIS IS A DESIGN REQUIREMENT FOR THE ACC. THE ACC SHROUD IS JETTISONED FOLLOWING SRB SEPARATION.

Figure 6

Cryogenic Propellant Scavenging Concepts

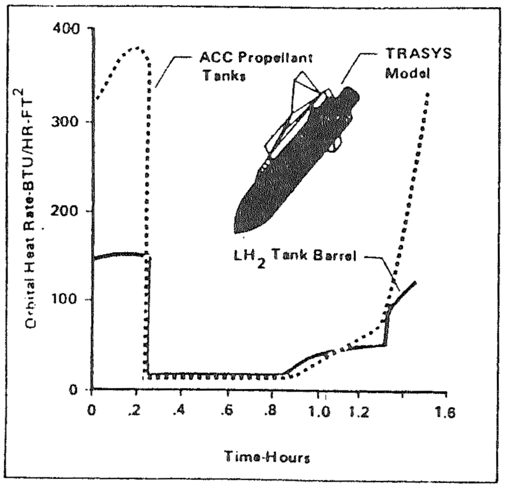
	OPERATIONS			
BASIC MISSION CONCEPTS	PROPELLANT TRANSFER	PROPELLANT SETTLING	OMV OPERATIONS	OTHER CONSIDERATIONS
TRANSFER TO STORAGE TANKS AT SPACE STATION	SPACE STATION 250 NM	• ET TRAP • THRUSTERS - RCS - ULLAGE GASES - GO2/GH2 ENGINES • TETHER	PROPELLANT SETTLING	DEORBIT ET
TRANSFER TO ORBITAL HOLDING TANKS IN LEO	LEO ≈ 160 NM		• PROPELLANT SETTLING • HOLDING TANK TRANSPORT	
TRANSFER TO HOLDING TANKS IN ACC PAYLOAD BAY	~~POST MECO SUBORBITAL~~	• THRUSTERS - RCS - ULLAGE GASES - GO2/GH2 ENGINES	HOLDING TANK TRANSPORT	
	POST MECO ORBITAL			DEORBIT ET/ACC
	PRE MECO	NOT REQUIRED		• SURPLUS AND FPR ONLY • ET QUANTITY MEASUREMENT • DEORBIT ET/ACC
	PRELAUNCH			• SURPLUS ONLY • DEORBIT ET/ACC

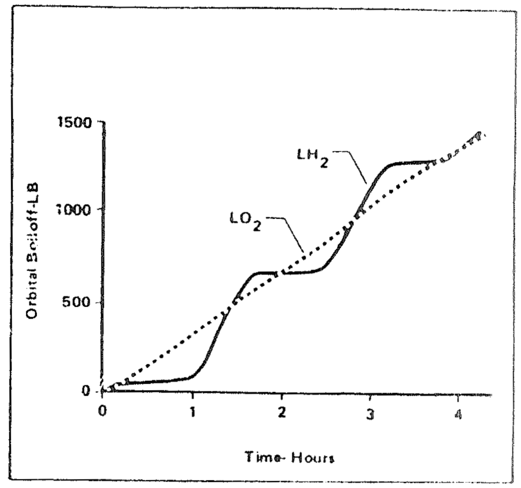
THE BASIC MISSION CONCEPTS AND ASSOCIATED OPERATIONS ARE SHOWN. PROPELLANT TRANSFER POST-MECO BUT SUBORBITAL IS NOT FEASIBLE BECAUSE INSUFFICIENT TIME IS AVAILABLE FOR OMV RENDEZVOUS (DISCUSSED EARLIER). PROPELLANT MUST BE MAINTAINED IN THE SETTLED CONDITION FOR ALL CASES WHERE PROPELLANT IS TRANSFERRED ON ORBIT. WITH AN ACC, SURPLUS PROPELLANT CAN BE LOADED IN THE SCAVENGING TANKS PRIOR TO LAUNCH OR SURPLUS AND FLIGHT PERFORMANCE RESERVE CAN BE TRANSFERRED PRE-MECO; SETTLING WOULD NOT BE REQUIRED FOR THESE CASES.

THE OMV IS ALWAYS USED TO TRANSPORT SCAVENGE OR ORBITAL HOLDING TANKS TO THE SPACE STATION. DE-ORBIT OF THE ET WITH OR WITHOUT THE ACC SUPPORT STRUCTURE IS REQUIRED FOR ALL MISSION CONCEPTS.

Figure 7

Orbital Heating Rates and Propellant Boiloff

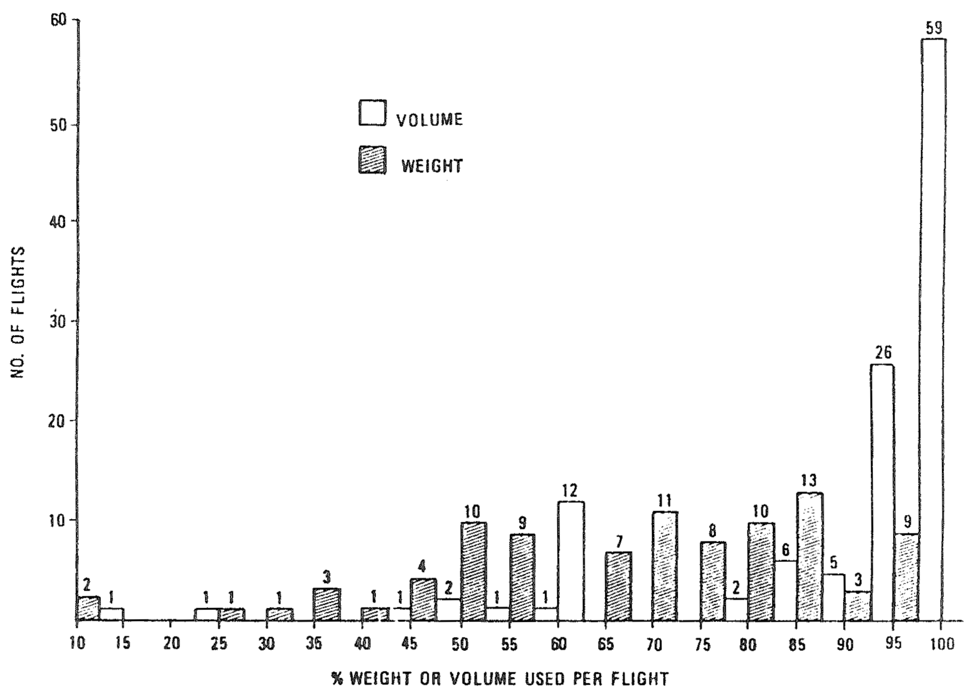




PREDICTED ORBITAL HEATING RATES AND RESULTING PROPELLANT BOILOFF ARE SHOWN AS A FUNCTION OF TIME. THESE DATA PROVIDE FURTHER VERIFICATION THAT THE SCAVENGING OPERATION MUST BE ACCOMPLISHED IN THE SHORTEST TIME POSSIBLE. PRELIMINARY ASSESSMENT HAS SHOWN THAT OMV RENDEZVOUS IS THE TIME-CONSUMING OPERATION, NOT SCAVENGING.

Figure 8

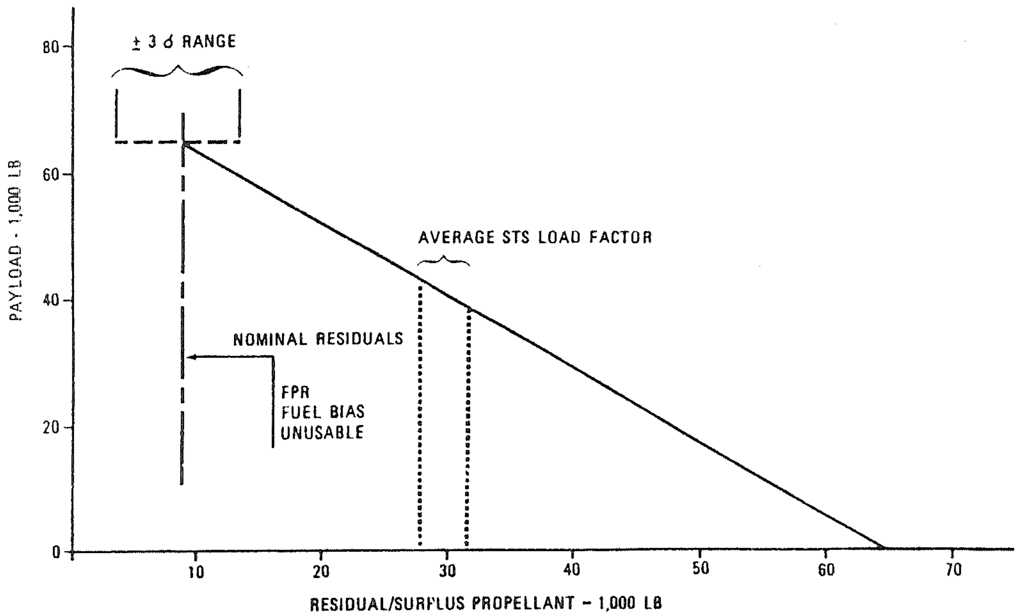
Weight Volume Utilization for STS Non-DOD ETR Flights



OF THE OVER 200 FLIGHTS SHOWN FOR THE NEXT 10 YEARS, NEARLY HALF ARE VOLUME LIMITED, WHILE ONLY A FEW (LESS THAN 10 PERCENT) ARE WEIGHT LIMITED, INDICATING AN ADVANTAGE OF USING AN ACC.

Figure 9

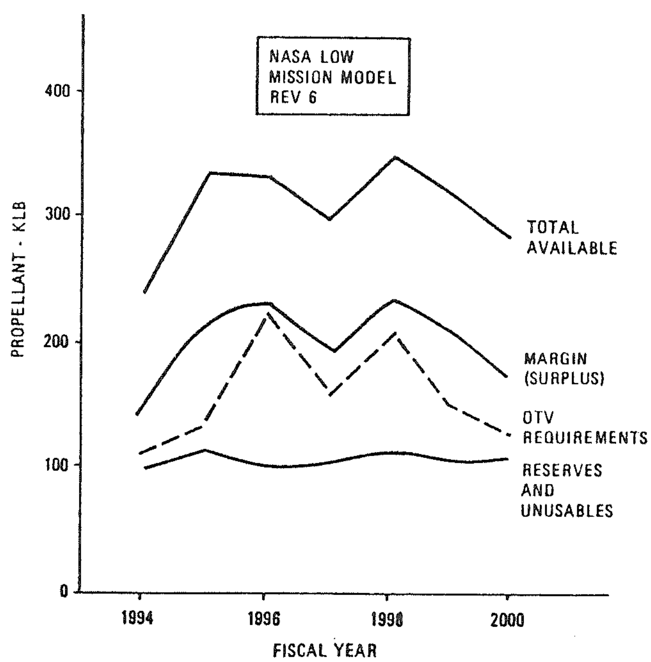
Payload vs Residual/Surplus Propellant



NOMINAL RESIDUALS OF 9000 LBM ARE SHOWN FOR THE FULLY WEIGHT LOADED SITUATION. HOWEVER, THE AVERAGE LOAD FACTOR IS ABOUT 40,000 LBM WHICH RESULTS IN ABOUT 21,000 LBM OF SURPLUS PROPELLANT IN ADDITION TO THE 9000 LBM OF RESIDUALS.

Figure 10

OTV Requirements vs Scavenged Propellant Available

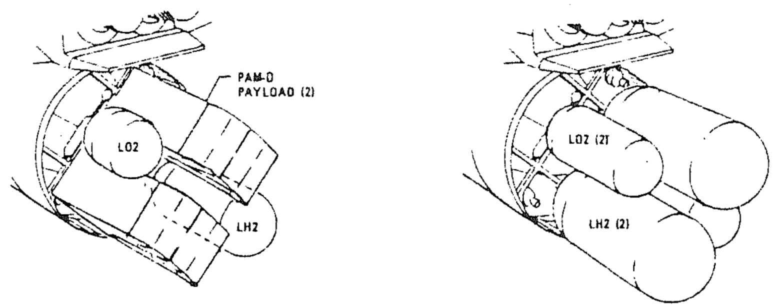


QUANTITIES OF PROPELLANTS AVAILABLE FOR SCAVENGING ARE SHOWN FOR THE YEARS 1994 TO 2000. OTV REQUIREMENTS WERE ESTABLISHED BY ROUTING ALL OTV MISSIONS TO GEO, ASSUMING APPROXIMATELY 45,000 LBM OF PROPELLANTS. RESERVES AND UNUSABLES AMOUNT TO ABOUT 100,000 LBM PER YEAR AND SURPLUS IS ABOUT 200,000 LBM PER YEAR, RESULTING IN A TOTAL OF ABOUT 300,000 LBM PER YEAR.

NO LOSSES HAVE BEEN INCLUDED IN THIS ASSESSMENT. HOWEVER, THESE CURSORY RESULTS SHOW THAT THE SURPLUS ALONE COULD MEET THE OTV NEEDS.

Figure 11

Typical Propellant Tank Installations in ACC



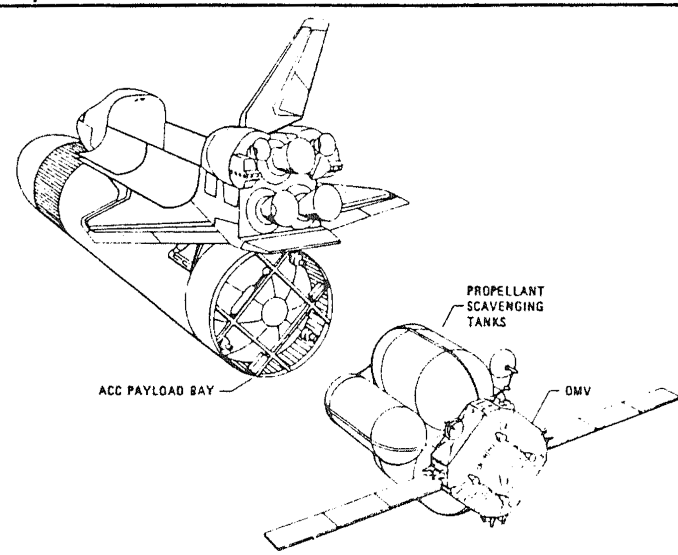
SCAVENGE TANKS PLUS PAYLOAD
9,000 LB PROPELLANT

SCAVENGE TANKS FOR TANKER MISSION
30,000 LB PROPELLANT

TWO POSSIBLE INSTALLATIONS ARE SHOWN. ONE INCLUDES TWO PAM-D PAYLOADS PLUS TANKS FOR 9000 LBM PROPELLANT AND THE OTHER HAS SCAVENGE TANKS ONLY TO ACCOMODATE 30,000 LBM PROPELLANT.

Figure 12

Propellant Tank Transfer with OMV



THE OMV HAS PICKED UP THE SCAVENGE TANKS FOR TRANSFER TO THE SPACE STATION DEPOT. THE ET DEORBIT MOTORS CAN BE SEEN ATTACHED TO THE ACC STRUCTURE.

Figure 13

SUMMARY

- AN AVERAGE OF 2 TO 4 STS FLIGHTS PER YEAR CAN BE SAVED BY SCAVENGING AND MANIFESTING PROPELLANT AS PAYLOAD TO SUPPORT SPACE-BASED OTV OPERATIONS.
- STS FLIGHTS SHOULD BE MANIFESTED TO CARRY MAXIMUM PAYLOAD WEIGHT.
- ADDITION OF AN ACC PERMITS LOADING CLOSER TO MAXIMUM AND REDUCES PROPELLANTS AVAILABLE FOR SCAVENGING (162 VS. 172 FLIGHTS). OTV NEED STILL MET.
- ET RESIDUAL PROPELLANTS HAVE AN OXIDIZER/FUEL RATIO FROM 2 TO 4.

Figure 14

PASSIVE STORAGE TECHNOLOGIES

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Passive storage of cryogens in space has been used for sometime in scientific instruments. This paper will describe some recent advances in storage technology and how passive techniques could be applied to the storage of propellants at the Space Station. The devices considered here are passive orbital disconnect struts, cooled shield optimization, liftweight shields and catalytic converters.

Cooled Shields

Cooled shields can greatly reduce the tank heat load in cryogen storage systems. This is the case for both passive and refrigerated systems. In passive systems, the enthalpy of the effluent gas is used to cool shields embedded in the insulation and thus intercept the heat before it gets to the tank. In refrigerated systems, the shields and tank are cooled by an external refrigerator. The use of cooled shields can reduce the overall refrigeration power.

The analysis presented in this section are given in a parametric form. This has the advantage of being insensitive to the exact model used for the thermal conductivity of the insulation. The analysis also applies to insulations that are penetrated by struts, plumbing, and wires. All of these penetrating devices are assumed to be attached to the cooled shields and the thermal conductivity function adjusted accordingly. We have also assumed that the insulation is multilayered insulation (such as double aluminized Mylar with silk net spacers). The analysis uses a tank temperature of 20 K and an outer shell temperature of 256 K (460°R). It should be noted that the heat loads are extremely sensitive to the outer shell temperature.

The use of cooled shields has the further advantage of allowing thicker insulation. Practical limitations limit MLI to blankets of 5 cm (2 inches) thick. Thus, an unshielded system can only have 5 cm of insulation. In a shielded system, each shield can support a blanket allowing more insulation.

The optimization analyses presented here are not the only ones that can be done, and they may not be the appropriate ones in all cases. In doing the optimization, I have not considered the mass or volume constraints on the system.

PASSIVE ORBITAL DISCONNECT STRUTS (PODS-III)

- 0 VARIABLE CONDUCTANCE-VARIABLE STRENGTH TANK SUPPORTS
  - 0 HIGH STRENGTH DURING LAUNCH
  - 0 LOW CONDUCTANCE ON ORBIT
- 0 LAUNCH AND ORBITAL CHARACTERISTICS INDEPENDENTLY SPECIFIED
  - 0 EXAMPLE
    - 0 FOR LAUNCH RESONANCE FREQUENCY 35 Hz
    - 0 FOR ORBITAL RESONANCE FREQUENCY 20 Hz
      - 0 RESULTS IN x10 LOWER ORBITAL HEAT LEAK
    - 0 WOULD IMPROVE IF ORBITAL REQUIREMENT WERE LOWER
- 0 STATUS
  - 0 A SYSTEM OF 6 STRUTS TO SUPPORT A 430 KG TANK IS BEING LAB TESTED
  - 0 THESE STRUTS ARE LIMITED BY A MINIMUM GAUGE REQUIREMENT
  - 0 STRUTS FOR A BIGGER SYSTEM WILL PERFORM BETTER

Figure 1

COOLED SHIELDS

- 0 THE USE OF COOLED SHIELDS CAN GREATLY REDUCE THE TANK HEAT LOAD
  - 0 SHIELDS ARE USEFUL IN BOTH PASSIVE AND REFRIGERATED SYSTEMS
  - 0 ANALYSIS APPLIES TO COOLED STRUTS, PLUMBING, WIRES, ETC., ALSO
- 0 THE PERFORMANCE OF BOTH SYSTEMS IS SENSITIVE TO THE OUTER SHELL TEMPERATURE

Figure 2

The analysis of the passive system is based on the method of minimum mass flow.[5] This type of analysis is appropriate for storage systems where there is no internally generated (from a scientific instrument, for example) heat load. The heat load on the tank is converted to a mass flow which then cools the shields. The mass flow is minimized with respect to position and temperature of the shields within the constraints of thermodynamics. This can be reduced to a system of 2N+1 simultaneous equations, where N is the number of shields. These equations can be easily solved by an iterative procedure. This gives the optimum locations, temperatures, and heat loads of the shields. These are given in the attached Table. The shield locations have been normalized by dividing by the total insulation thickness (not including the shield thicknesses). The temperatures are given in Kelvin. The resulting heat loads have been normalized by dividing by the heat load of an unshielded tank with 2 inches (5 cm) of insulation, this being the thickest blanket that can be easily manufactured. There are two columns of heat loads shown. The first is for a total insulation thickness of 2 inches. The second column assumes that the thickest insulation blanket (the outermost one) is 2 inches and the others are increased proportionally.

An interesting result of this analysis is that the shields are not evenly spaced in position or temperature. Rather, they are closer to the tank. This is a result of the thermal conductivity decreasing with temperature. One can also see that the heat load decreases as the number of shields is increased. The first shield results in a large reduction. There is still a significant gain by using 2 shields, but not for more than 2 shields. The relative heat load for the case of an infinite number of cooled shields is given for reference.

The analysis for refrigerated systems uses the same assumptions as above but takes a different approach. It uses the method of minimum entropy production. This method is appropriate for active refrigerated systems and for passive systems with sufficient internal heat generation. The analysis presented here is for the later case where the efficiency of the refrigeration can be ignored (Carnot efficiency is assumed). For an active refrigerated system the entropy produced by the refrigerator inefficiency would have to be included. The entropy produced at the shields and at the tank is $S_i = Q_i/T_i$ where Q_i is the heat flux absorbed at the ith shield (or at the tank) and T_i

COOLED SHIELDS
PASSIVE SYSTEMS

- 0 BASED ON METHOD OF MINIMUM BOIL-OFF
  - 0 OPTIMIZED LOCATION AND TEMPERATURE OF SHIELDS
- 0 ASSUMPTIONS: TANK AT 20 K, OUTER SHELL AT 256 K
  DOUBLE ALUMINIZED MYLAR/SILK NET INSULATION

NUMBER OF SHIELDS	NORMALIZED[1] LOCATION	SHIELD TEMP., K	RELATIVE HEAT LOAD 2" TOTAL[2]	2" THICKEST[3]
0			1.0	1.0
1	.35	114	.34	.22
2	.19, .50	77, 162	.26	.13
3	.13, .32, .60	61, 120, 188	.24	.097
∞			.18	

1) TANK = 0, OUTER SHELL = 1, THICKNESS OF SHIELDS NOT INCLUDED
2) TOTAL THICKNESS OF INSULATION (NOT COUNTING SHIELD THICKNESS) IS TWO INCHES
3) THICKEST INSULATION BLANKET IS TWO INCHES THICK

Figure 3

is the respective temperature. The total entropy produced is $S = -Q_0/T_0 + Q_i/T_i$ where T_0 is the outer shell temperature and Q_0 is the sum of the Q_i's. The most thermodynamically efficient system occurs when S is minimized with respect to the location and temperature of the shields. (This will give the system that requires the least refrigeration.) This method involves a simple iterative process similar to the one used in the passive case.

The parametric results are shown in the attached Table. The values in the last columns refer to the heat load refrigeration required on the tank due to the insulation. The heat that must be removed from each of the shields is also simple to calculate but has been left off the chart for clarity. More heat must be extracted at the shields than in the storage case. In a passive system this additional cooling must be supplied by an increased mass flux caused by the instrument dissipation. The principal result of this analysis is that the optimum location of the shields is slightly farther out in the insulation and their optimum temperatures are slightly colder.

COOLED SHIELDS
PASSIVE INSTRUMENT OR REFRIGERATED SYSTEMS

- 0 BASED ON METHOD OF MINIMUM ENTROPY PRODUCTION
  - 0 OPTIMIZED LOCATION AND TEMPERATURE OF SHIELDS
- 0 ASSUMPTIONS: TANK AT 20 K, OUTER SHELL AT 256 K
  DOUBLE ALUMINIZED MYLAR/SILK NET INSULATION

NUMBER OF SHIELDS	NORMALIZED[1] LOCATION	SHIELD TEMP., K	RELATIVE HEAT LOAD 2" TOTAL[2]	RELATIVE HEAT LOAD 2" THICKEST[3]
0			1.0	1.0
1	.42	95	.19	.11
2	.26, .59	60, 141	.11	.044
3	.19, .41, .68	46, 95, 168	.082	.026

1) TANK = 0, OUTER SHELL = 1, THICKNESS OF SHIELDS NOT INCLUDED
2) TOTAL THICKNESS OF INSULATION (NOT COUNTING SHIELD THICKNESS) IS 2 INCHES
3) THICKEST INSULATION BLANKET IS 2 INCHES THICK

Figure 4

Both the passive and the refrigerated systems require good heat exchangers. These are particularly difficult to model in the passive case because the effluent gas passes through several flow regimes. It starts as laminar flow at the tank and ends as sonic flow at the exhaust nozzle. The flow also spans a large temperature range with the concomitant variation in the properties of the gas. Fortunately, there are models available to handle this problem.[8]

If shields are to be used, it is important that their mass be kept small to keep the system mass down. However, the shields must be stiff enough to meet the resonance requirements. A possible choice is to use thin aluminum sheet bonded to aluminum honeycomb.[1] We have done an analysis for a 2 m^3 (70 ft^3) instrument system. This analysis showed that a 0.13 mm (0.005") Al sheet bonded to 0.64 cm (0.25") thick Al honeycomb (1.3 cells/cm of 0.033 mm gauge) should have sufficient thermal conductivity and strength. The density of such a structure is 0.57 Kg/m^2.

COOLED SHIELDS

- 0 HEAT EXCHANGERS FOR PASSIVE SYSTEMS
  - 0 DIFFICULT TO MODEL
    - 0 LARGE TEMPERATURE DEPENDENCE OF GAS PARAMETERS
    - 0 SPANS DIFFERENT FLOW REGIMES - SONIC AT EXIT
- 0 MASS OF SHIELDS ARE IMPORTANT IN EITHER PASSIVE OR ACTIVE SYSTEMS
  - 0 USE LIGHTWEIGHT CONSTRUCTION
    - 0 THIN AL SHEET BONDED TO AL HONEYCOMB
    - 0 $\sim$0.6 KG/M^2 POSSIBLE

Figure 5

Para-Ortho Conversion

The heat load in passive hydrogen systems can be reduced by using a catalyst on the shields.[9] The converter speeds up the conversion of para-hydrogen to ortho-hydrogen. The equilibrium mixture of para (anti-parallel proton spins) and ortho (parallel proton spins) is temperature dependent. At 20 K it is >99% para and at 300 K it is 25% para. The para-ortho conversion is an endothermic reaction that is usually too slow to be of use. However, it can be speeded up by using appropriate catalysts. The heat of conversion has a maximum of 400 J/g at 100 K. This compares with the enthalpy of the gas of 900 J/g for a change in temperature from 20 K to 100 K. Thus, the reaction can be used to supply additional refrigeration to the shields. Conversion efficiencies of $\sim$100% are possible with commercially available catalysts. For example, the use of APACHI-1 would require about 100 g of catalyst for each g/s of hydrogen flow.

One way of using a catalyst would be to increase the performance of a passive system. This is most effective if the catalyst can be attached to a shield running near the 100 K peak in the heat of conversion. From the previous Tables it is seen that the 1 shield and 3 shield cases are ideal for this. An analysis of the one shield case is given in the attached Table. A 15% reduction in heat load can be achieved.

PARA-ORTHO CONVERSION

- 0 TWO FORMS OF H_2
  - 0 PARA-ANTI-PARALLEL PROTON SPINS
  - 0 ORTHO-PARALLEL PROTON SPINS
  - 0 AT 300 K - 25% PARA
  - 0 AT 20 K - >99% PARA
  - 0 PARA TO ORTHO CONVERSION IS ENDOTHERMIC
  - 0 PEAK HEAT OF CONVERSION ~400 J/G, @ ~100 K (ENTHALPY 20 K - 100 K ~900 J/G)
  - 0 CONVERSION REQUIRES CATALYST
    - 0 100% CONVERSION POSSIBLE
    - 0 ~100 G CAT/GH_2S^{-1} FOR APACHI-1
- 0 IMPROVED PASSIVE SYSTEM

			Relative Heat Load	
Shields	Location	Temperature	2" Total	2" Thickest
1	.35	106	.29	.19

Figure 6

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PODS:

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N85-17004

ACTIVE COOLING REQUIREMENTS FOR PROPELLANT STORAGE

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Recent NASA and DOD mission models have indicated future needs for orbital cryogenic storage and supply systems. Cryogens required will include hydrogen and oxygen. Tank sizes will vary from 32 ft^3 to 1800 ft^3 for applications ranging from Space Station on board propulsion to Space Station Orbital Transfer Vehicle (OTV) propellant storage. The storage durations may vary from a few hours for such missions as OTV Low Earth Orbit (LEO) to Geosynchronous Equatorial Orbit (GEO) transfer and resupply, to several years for mission such as Space Station station keeping and space-based laser systems. There is strong economic incentive for reducing the boiloff losses for long duration missions. It has been proposed that refrigeration be investigated to reduce the heat load to the tanks and thereby minimize boiloff.

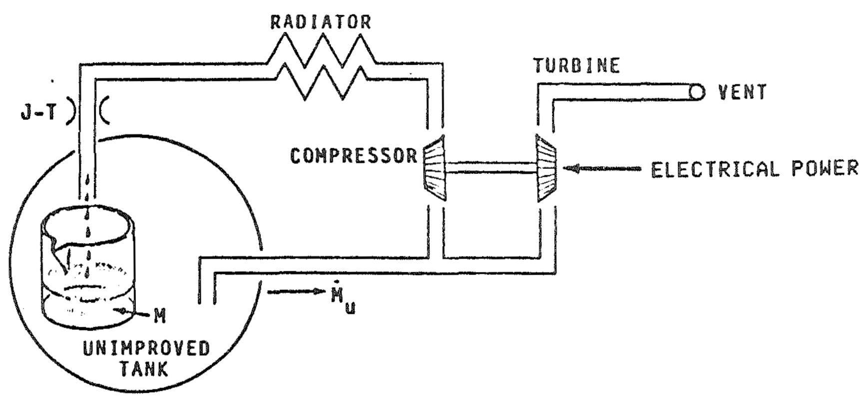
Two thermal control systems were evaluated in this analysis. These systems showed the greatest promise for improving storage life and include:

- An open cycle thermodynamic vent system with:
  - a refrigeration system for partial hydrogen reliquefaction located at the LH_2 tank
  - refrigeration at the LH_2 tank - vapor cooled shield for integrated and non-integrated tank designs to reduce boiloff
- A closed system with direct refrigeration at the LH_2 tank vapor-cooled shield to eliminate boiloff

For storage tank designs utilizing active coolers, careful design of the passive thermal control system is necessary to achieve the optimal refrigeration system performance and minimal overall thermal control system mass.

Individual subsystems must be integrated functionally and structurally to form an operable propellant reliquefier. The liquefaction equipment in this schematic includes the refrigerators (ex., reversed Brayton), their drive motors, and large space radiators. Boiloff from the liquid hydrogen and liquid oxygen storage vessels is recycled through the refrigeration equipment where reliquefaction occurs. However, the boiloff reliquefaction process requires refrigerator operation at cryogenic temperatures.

OPEN CYCLE-RELIQUEFACTION



- LIQUID HYDROGEN REQUIRES LOW TEMPERATURE REFRIGERATOR OPERATION

Figure 1

In the present study, a reversed Brayton cycle unit was baselined for the propellant processor. The Brayton cycle refrigerator was selected over Sterling, Vuilleumier and other cycles because it has the lowest weight and volume at the higher power refrigeration requirements. It uses gas bearing turbomachinery, resulting in high cycle efficiency, long life and high reliability. Two refrigeration stages are required for hydrogen liquefaction.

A summary of the estimated liquefaction performance capability used for final processor sizing is shown. Current refrigeration systems could practially reliquefy only a percentage of the total boiloff from an OTV propellant storage depot tank.

Unless modifications are made to the tank design (with addition of refrigerated or non-refrigerated vapor cooled shields) it would appear that reliquefaction systems may not be as attractive for minimizing propellant boiloff, as alternative thermal control system designs.

ELECTRIAL POWER REQUIREMENT FOR HYDROGEN RELIQUEFACTION

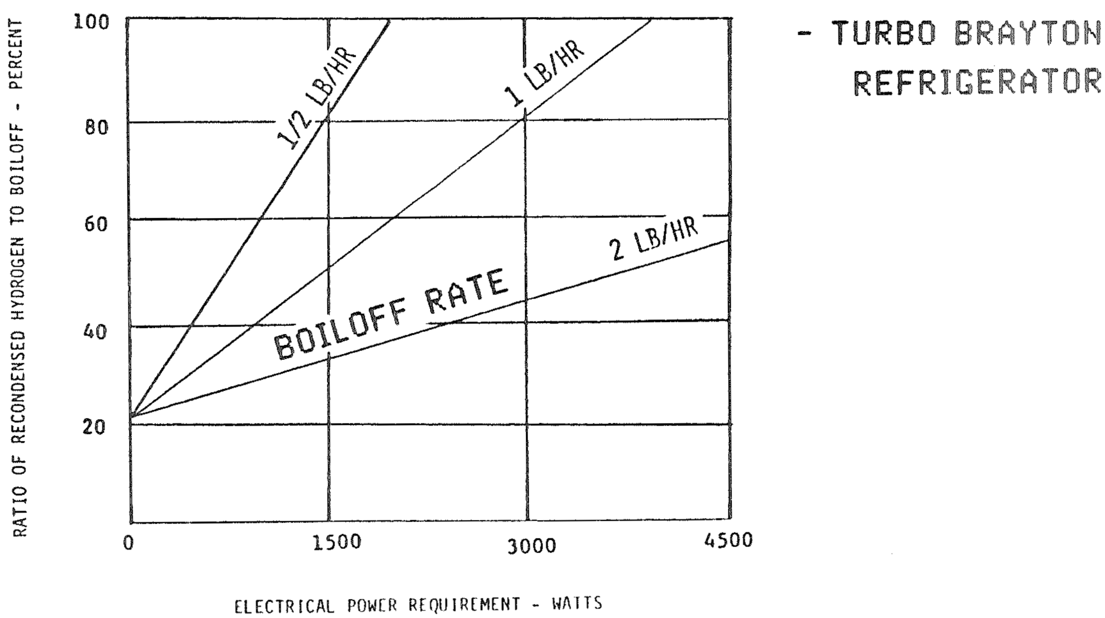


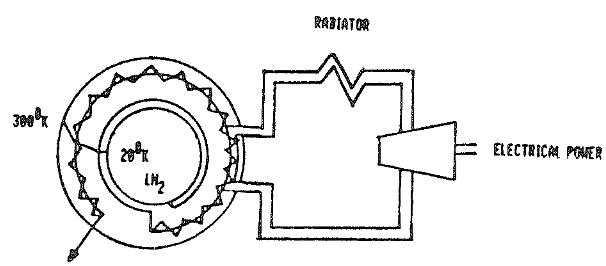
Figure 2

The cryogenic storage system, described in the present study, is for orbital long-term storage of subcritical liquid cryogens. The system consists of a pressure vessel containing the saturated liquid cryogen, a structural support system, multilayer insulation (MLI), and a vapor-cooled shield (VCS) with a heat exchanger.

Use of a vapor-cooled shield integrated with a refrigerator permits operation of the refrigerator at temperatures higher than 20°K, thereby obtaining a marked improvement in cooler efficiency.

Two thermal control open cycle systems were analyzed. These systems were evaluated for their ability to reduce boiloff losses while minimizing their environmental impact. The systems include integrated and independent LO_2/LH_2 thermal control systems.

OPEN CYCLE - COOLED SHIELD



- PERMITS REFRIGERATOR OPERATION AT HIGHER TEMPERATURE
- TWO APPROACHES
  - INTEGRATED LO_2-LH_2 THERMAL CONTROL
  - INDEPENDENT LO_2-LH_2 THERMAL CONTROL

Figure 3

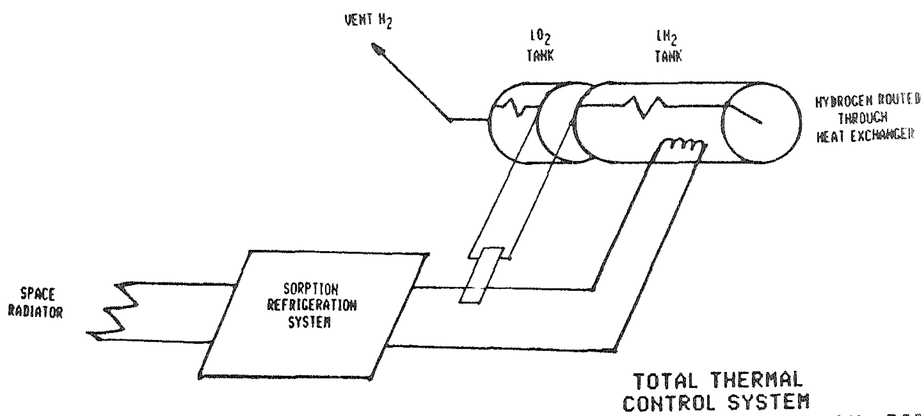
For a given location, the shield temperature can be optimized to:

- o minimize the combination of thermal control system, total propellant and tankage mass.

In the accompanying figure, a sorption refrigeration system which has been coupled to the LH_2 and LO_2 propellant tanks is shown and is representative of an integrated thermal control system design for the LH_2/LO_2 tanks. The vented liquid hydrogen is passed over the surface of the fuel tank where it evaporates and maintains the tank temperature at 20°K, before entering an intermediate heat exchanger. Here, the refrigerator working fluid is precooled to 28°K, thereby increasing the cooler performance. The hydrogen leaving the intermediate heat exchanger would then be routed through the heat exchanger placed around the oxygen tank, before being vented to space. The amount of boiloff is governed by a requirement to remove all of the liquid oxygen tank heat load, with no LO_2 venting.

The table gives a comparison of the thermal control system mass and LH_2 boiloff for three thermal control options for a 15700 Kg (7800 ft^3) LH_2 OTV storage tank. The amount of boiloff which directly evolves from the LH_2 tank, when

OPEN CYCLE-INTEGRATED LO_2-LH_2 TANK THERMAL CONTROL



COOLING METHOD	TOTAL THERMAL CONTROL SYSTEM MASS (kg)	LH_2 BOILOFF (kg/hr)
MLI only	less than 20	.728
TVS+VCS	1436.3	.314
TVS+ REFRIGERATOR	111.0	.292

*SORPTION COOLER WITH 28°K PRECOOLING

Figure 4

configured for passive cooling, will serve as a baseline. Using a TVS attached to a 55°K hydrogen tank shield, all of the excess heat from the LH_2 and LO_2 tank can be removed without the need for any additional refrigeration or oxygen venting. With this design, the boiloff is reduced to 53 percent of its original value. Because the system is constrained by the requirement that the boiloff be large enough to intercept all of the heat leak into the LO_2 tank, the amount of LH_2 boiloff is not reduced substantially by lowering the shield temperature to 20°K and coupling a refrigerator to it. However, by letting the tank wall serve as a 20°K vapor cooled shield (VCS), the mass of the VCS can be eliminated. Thus, the total thermal control system mass is greatly reduced. In spite of the above efforts to reduce structural weight, the amount of fluid vented during a long mission can be large. Vent losses can be greatly reduced by providing an independent LH_2 tank thermal control system design.

In order to properly design a propellant tank thermal control system, it is important to assess the impact of various parameters upon the vapor generation of the cryogenic propellants stored within the tank. The accompanying figure illustrates the effect of varying from its nominal value the magnitude of a given parameter (i.e., MLI thickness, environment temperature, strut heat leak, pipes and penetrations and para/orthos conversion efficiency) upon the calculated tank heat input relative to the tank's heat input using the parameter's nominal value. The nominal values represented in this figure were obtained from data representing the SOA technology as defined by Martin Marietta for an OMV.* The tank heat inputs appear to be most sensitive to changes from nominal values in the MLI thickness and environment temperature. Consequently, atten-

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* J. Robinson, Point Design and Technology Assessment, Long Term Cryo Storage Study, Final Program Review, Sept. 20, 1983.

tion was focused on the effects of these two parameters in developing a tank thermal design which minimized boiloff within the system constraints.

LH_2 TANK PARAMETRIC HEAT INPUT SENSITIVITY

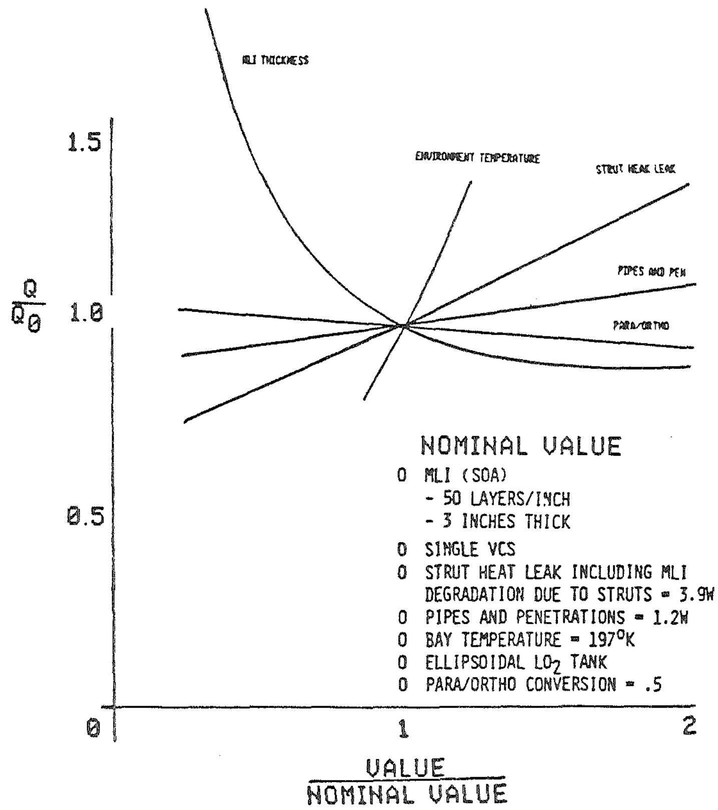

Figure 5

Mass and energy conservation equations have been applied to the system in order to minimize the propellant boiloff. The accompanying figure shows the reduction in boiloff for refrigerated and non-refrigerated tank designs versus the boundary temperature ratio T_H/T_C. The optimization study was performed for an OMV3, LH_2 tank designed for a ten year mission duration. The boiloff from this tank, configured for passive cooling (utilizing 2 inches of MLI and no shield), served as a reference against which the boiloff from improved tank designs were judged. The curves are represented successively from the top as:

i) a non-refrigerated single shield tank which is temperature and position optimized at T = 87°K nd x/t* = 0.44

ii) a multiple shielded tank (i.e., an infinite number of cooling shields)

iii) a refrigerated single shield tank which is optimized at T = 45°K and x/t = 0.5.

Here, x/t represents the non-dimensional distance from the tank for a given shield thickness, t. An optimized single shield refrigerated tank design substantially reduces the boiloff as compared to that generated from single and multi-shielded nonrefrigerated tank systems.

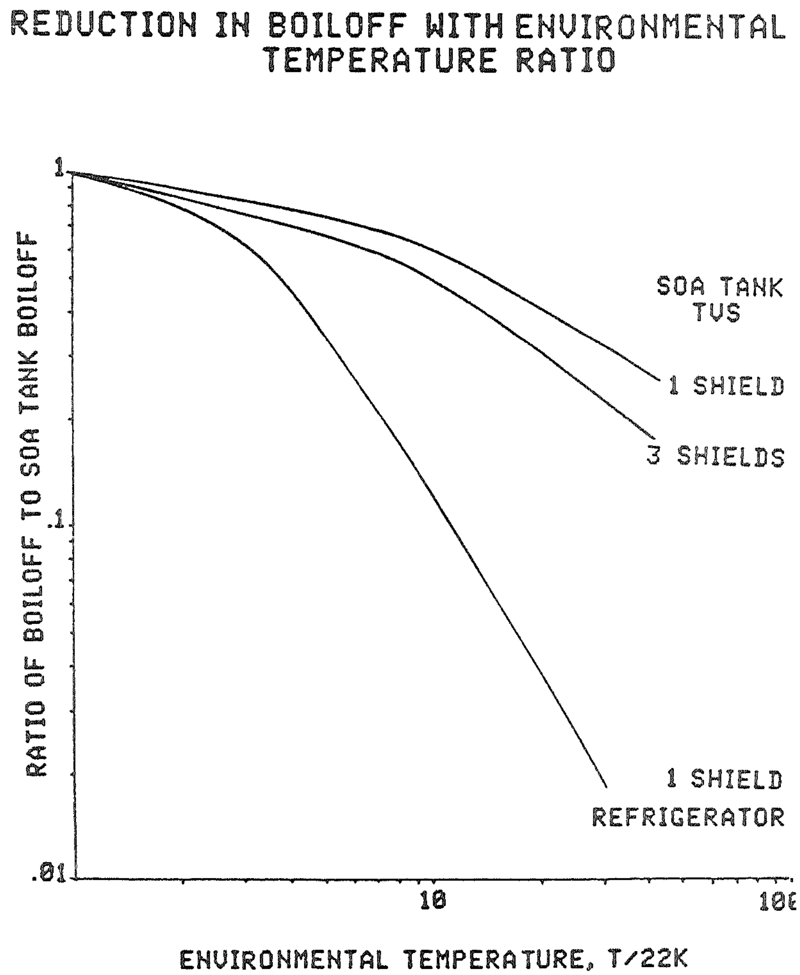

Figure 6

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* x/t represents the non-dimensional distance from the tank wall for a given insulation thickness, t.

Material technology advances and vessel design ingenuity can reduce propellant boiloff and overall system structural weight. In spite of these efforts, the weight of fluid vented during a long mission can be large. Consequently, a comparison of the relative performance of open and closed cycle thermal control schemes for a typical propulsion vehicle (OMV 3) is presented. It was assumed that the mass of propellant required at the end of any given mission was held constant. In addition, the shield temperature and position were optimized as a function of mission duration.

The launch weight of a tank configured for open cycle passive cooling utilizing two inches of MLI is shown to exceed the launch weight of the closed cycle system with active cooling at the 20°K tank wall (reference system) for mission durations greater than 1/2 year. Furthermore, an open cycle thermal control system with an actively cooled shield is shown to be preferable to the reference system for missions less than 1.6 years. Beyond this time, there

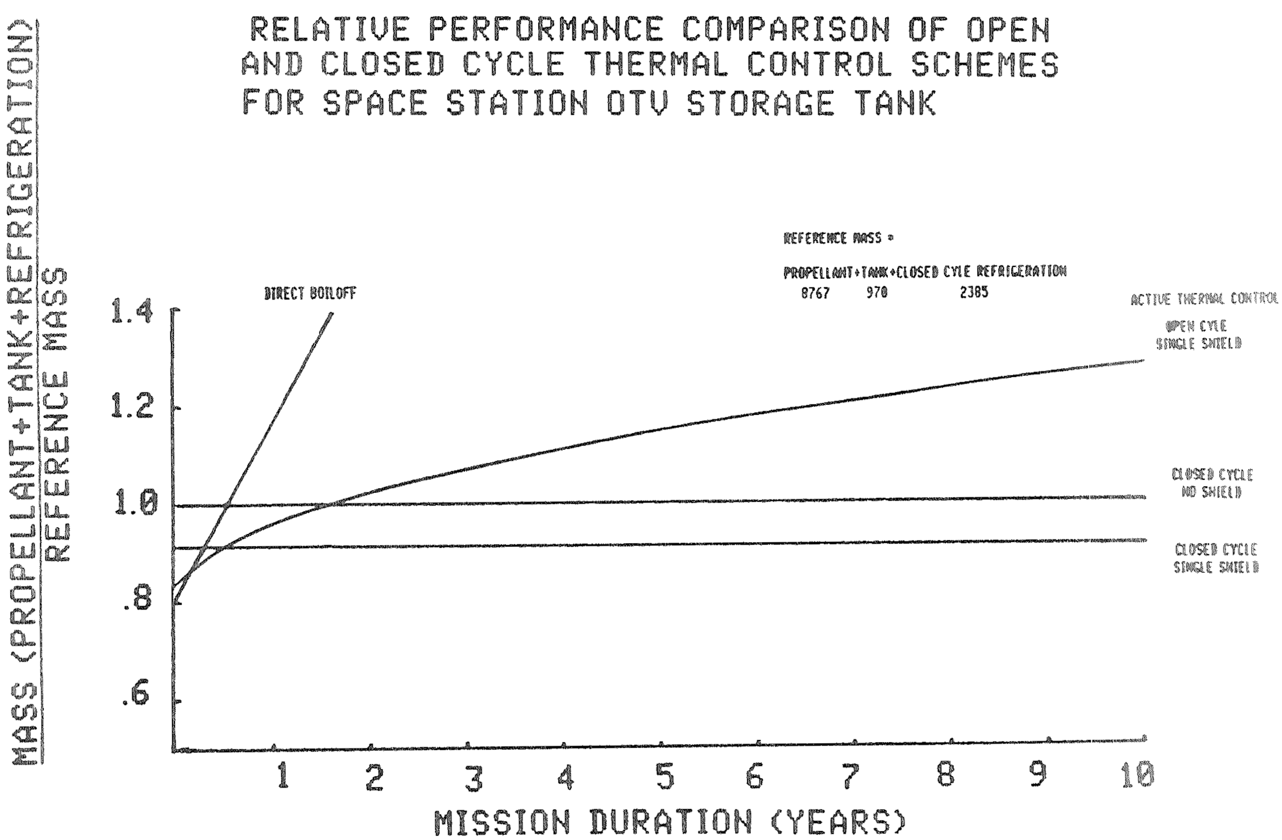

Figure 7

is a substantial mass savings to be gained by employing a closed cycle system with direct refrigeration at the tank wall. Finally, the use of actively cooled shields will enhance the overall thermal control system performance.

As an alternative to actively cooled, open cycle systems, a refrigeration system can be employed that provides direct cooling of both cryogenic tanks. Figure 1 shows a hybrid $LaNi_2$ charcoal nitrogen (C/N_2) propellant tank direct cooling refrigeration scheme. Alternative designs could utilize Stirling, Brayton and Veuillemier refrigeration systems.

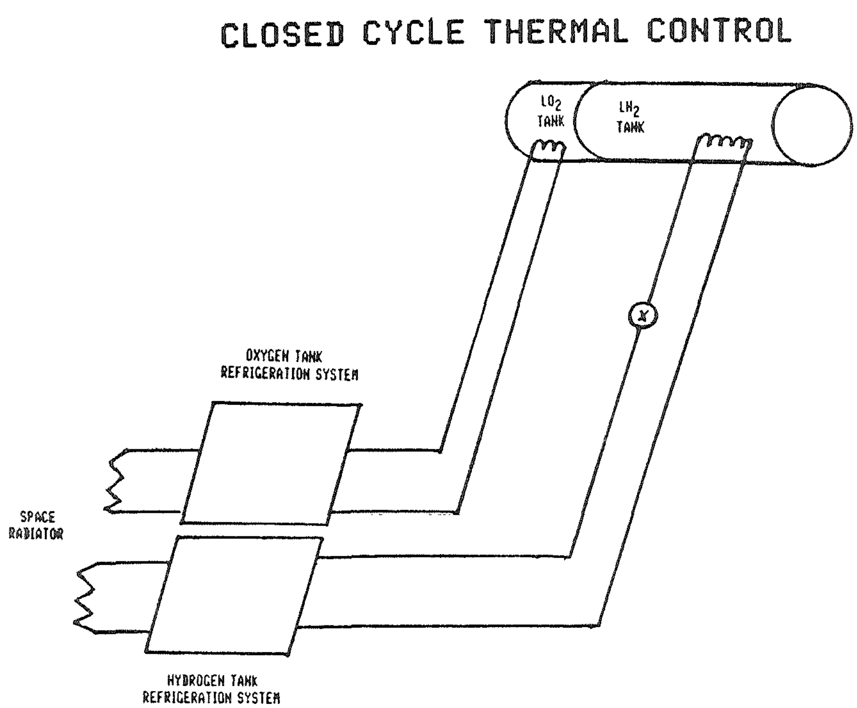

Figure 8

A preliminary investigation was made of the above refrigeration system design to determine the overall closed cycle refrigeration system mass. An optimized shield temperature and location were found which minimized the overall refrigeration system mass. For mechanical coolers, the optimized shield temperature and position is 94°K and x/t = 0.5.* By adding a refrigerated shield to an insulated tank (which utilizes a 20°K cooler to intercept heat at the tank wall), the overall refrigeration mass was reduced by approximately 55% for the mechanical coolers.

An assessment of the space station propellant thermal control system mass and heat loads has been made, corresponding to the minimum and maximum size propellant tanks which could be maintained on space station. The Turbo Brayton system was used to represent a typical mechanical refrigeration system, which was attached to a vapor-cooled shield. The refrigeration system mass included power supply, energy storage and radiator. The mass of the vapor-cooled shield was not included in the analysis and the shield weight could become quite substantial, particularly if the mass of the tubing and supports are accounted

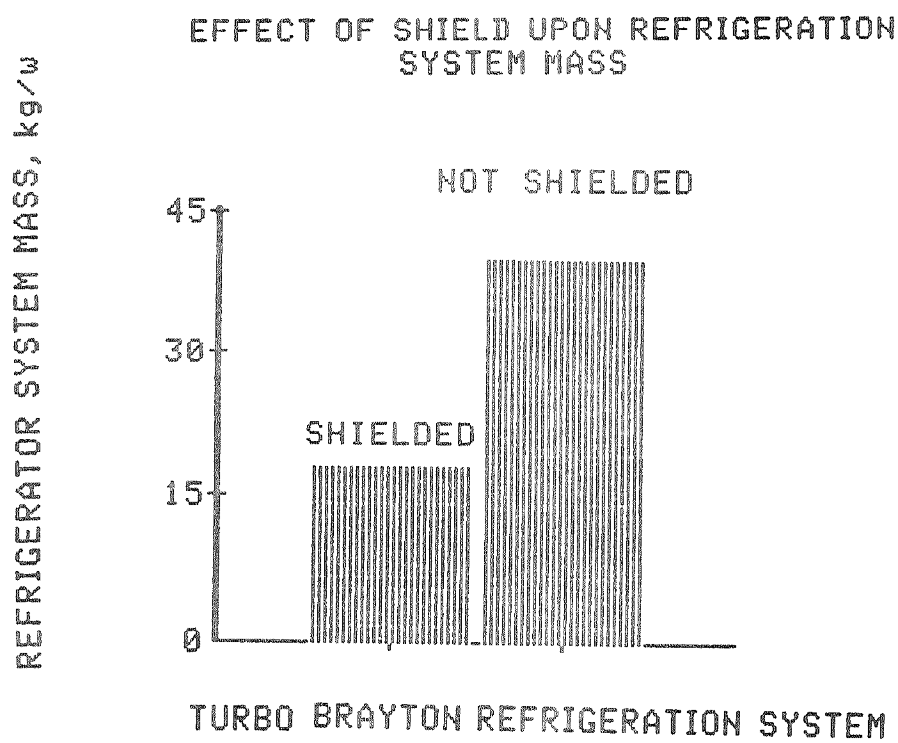

Figure 9

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* x/t represents the non-dimensional distance from the tank wall for a given insulation thickness, t.

for. The shield masses for the onboard propulsion and OTV tank farm storage tanks could conservatively reach a maximum of 24.5 kg and 1340 kg, respectively.

Although these refrigeration systems represent a non-trivial mass penalty, their employment can substantially reduce the mass of accumulated boiloff expended over the life of a long duration mission. This translates into a substantial savings in initial wet system mass transportation costs.

SPACE STATION PROPELLANT THERMAL
CONTROL SYSTEM MASS AND HEAT LOADS

System	Propellant	Volume (ft^3)	Propellant Mass (Kg)	Heat Input (watts)	Refrigeration System Mass	
					Shielded (Kg)*	Unshielded (Kg)
On Board Propulsion System	H_2	32	61	2.22	40.0	89.9
OTV Tank Farm	H_2	7800	15,700	116.5	2097.0	4718.0

* Assumptions

- Shield mass not inlcluded
- Sized for Turbo Brayton System

Figure 10

Increasing the storage life of state of the art, passive vented and non-vented propellant tanks is essential in order to satisfy the requirements of long duration missions within economic constraints. For a vented propellant storage tank design, active coolers may be employed for propellant reliquefaction, or for intercepting heat along a vapor-cooled shield in order to reduce the heat load to the tank. Reliquefaction systems are shown to not be attractive for minimizing propellant boiloff in an unshielded tank design. Careful thermal design is necessary to achieve the minimum possible boiloff within the system constraints. Independent storage tank thermal control, utilizing actively refrigerated vapor-cooled shields for vented propellant storage, results in a significant reduction in boiloff loss over alternative vented storage tank system designs. However, open cycle systems may not be economically attractive for long-term storage. The maximum fluid and vessel weight savings occurs if the refrigeration capacity is chosen to match the vessel heat leak, thereby allowing storage without venting. Use of refrigerated shields has been shown to significantly improve the performance of mechanical coolers in non-vented, as well as vented, storage tank designs. This type of storage tank, thermal control system design, results in a significant reduction in refrigeration system mass.

CONCLUSIONS

RELIQUEFACTION SYSTEMS ARE NOT ATTRACTIVE FOR MINIMIZING PROPELLANT BOILOFF

OPEN CYCLE SYSTEMS MAY NOT BE ECONOMICALLY ATTRACTIVE FOR LONG TERM STORAGE

A NUMBER OF REFRIGERATION SYSTEMS ARE AVAILABLE TO ASSIST IN THE LONG TERM STORAGE OF CRYOGENIC PROPELLANTS

SHIELDS CAN SIGNIFICANTLY IMPROVE THE PERFORMANCE OF MECHANICAL COOLERS

Figure 11

PROPELLANT TRANSFER: ATTACHED DEPOT

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Propellant transfer at an attached depot involves: 1) resupply tankers (dedicated launch from the ground or scavenging from the External Tank) to resupply the depot: 2) depot storage and supply tanks (attached, free-flyer or tethered) from which liquid hydrogen and liquid oxygen are transferred to fill the space-based OTV; and 3) the space-based OTV which is resupplied with cryogens from the depot. Liquid storage and supply, thermal control and transfer/resupply requirements for an attached depot are listed, and technologies defined. The specific fluid management elements and approaches for an attached depot are defined. The Cryogenic Fluid Management Facility (CFMF) Shuttle attached-payload test bed, scheduled for a mid-1988 first launch, will provide much of the needed technology.

Three elements are involved in the propellant transfer operation associated with an attached depot — a resupply tanker, a space station depot and a user system such as a space-based OTV. The technologies that are involved for each are shown on this chart. Liquid storage and supply is an element of each one of these systems as is thermal control. For the resupply tanker the thermal control period is relatively short, on the order of several days. For the space station depot the thermal control is relatively long, on the order of several months, perhaps 90 days or 180 days. For the space-based OTV the thermal control requirement is of intermediate length, perhaps on the order of several weeks. The space station depot not only must be resupplied by the resupply tanker, but in turn is the supply source for transferring propellant to the space-based OTV. The space station depot must be resupplied and also must be a supply source. For the OTV, resupply to initiate the next mission is accomplished prior to the mission. Fluid transfer is key to the space basing of OTVs.

ON-ORBIT CRYOGENIC FLUID MANAGEMENT

RESUPPLY TANKER (RT) (DEDICATED OR STS SCAVENGING)	SPACE STATION DEPOT (SSD) (ATTACHED, FREE FLYER OR TETHER)	SPACE-BASED OTV (SB OTV)
• LIQUID STORAGE AND SUPPLY	• LIQUID STORAGE AND SUPPLY	• LIQUID STORAGE AND SUPPLY DURING MISSION
• THERMAL CONTROL (RELATIVELY SHORT TERM ~ SEVERAL DAYS)	• THERMAL CONTROL (RELATIVELY LONG TERM ~ SEVERAL MONTHS)	• THERMAL CONTROL (INTERMEDIATE TERM ~ SEVERAL WEEKS)
	• RESUPPLY/TRANSFER CAPABILITY INCORPORATED IN DESIGN	• RESUPPLY TO INITIATE NEXT MISSION - FLUID TRANSFER CAPABILITY KEY TO SPACE-BASING

Figure 1

C-3

The functions that make up cryogenic fluid management are liquid storage and supply, thermal control and fluid transfer and resupply. Liquid storage and supply involves liquid acquisition devices that acquire the liquid in low-g, and retain it in a position to be transferred as single-phase liquid using capillary or fine-mesh screen acquisition devices. Thermal control can either be passive or active, and fluid transfer involves receiver tanks as well as a transfer line.

DEFINITION - IN-SPACE CRYOGENIC FLUID MANAGEMENT (CFM)

- LIQUID STORAGE/SUPPLY (LS/S)
  - ACQUISITION/RETENTION
  - SINGLE PHASE LIQUID EXPULSION
- THERMAL CONTROL (TC)
  - PASSIVE SYSTEMS
  - ACTIVE SYSTEMS
- FLUID TRANSFER/RESUPPLY (FT/R)
  - RECEIVER TANK
  - TRANSFER LINE

Figure 2

This chart lists some of the fluid management requirements for an attached or free-flier depot. Fluid acquisition devices are designed to feed single-phase liquid down to low residuals, on the order of several percent of the loaded volume. Relatively high volumetric flow rates may be required to transfer propellant to an OTV in a several-hour transfer period. We would like not to have an imposed gravity or settling as part of the transfer operation because some systems may be limited by having special low-gravity requirements. The depot must incorporate adequate meteoroid protection. A key technology is the ability to gauge the mass of liquid in the supply and receiver systems so we can determine when it is time to stop the resupply operation. Two particularly important requirements involve contaminant or particle buildup in the tank over time when we are running basically a filling-station type operation, and the impact that slosh forces generated by liquid moving within the tank systems may have upon attitude control.

The purpose of the thermal control system is to minimize boil-off losses. Some studies have indicated a 90-day resupply time period is a reasonable operational criterion, in which case the thermal control system should be designed to provide up to 180 days for a contingency storage period assuming the resupply launch does not occur as planned. One attractive thermal control approach is to integrate the entire hydrogen-oxygen system by using a coupled heat exchanger which allows the boiloff from the hydrogen system to thermally condition the oxygen system to prevent or minimize the boiloff of oxygen.

From a fluid transfer and resupply technology standpoint, the capacity is needed to top partially full tanks. This is consistent with the concept of making the resupply operation somewhat similar to a filling station operation, where we would not empty the tanks every time we were ready to refill, but merely top a tank that had already been filled and partially used. Mass gaging is again a very key technology when we consider how we are going to control the operations of transferring fluid from one tank to another, knowing when we have completed the transfer process. It is important that we minimize rapid venting in the vicinity of the space station or other payloads. It may be that separate catch tanks are required for reliquefaction if venting of vapors, in particular non-condensibles, is damaging to payload elements. Resupply on 90-day intervals has already been discussed. Diagnostics for efficient operational control and safety will also be a part of any transfer system.

ATTACHED OR FREE-FLYER DEPOT FLUID MANAGEMENT REQUIREMENTS

- LIQUID STORAGE AND SUPPLY
  - SINGLE-PHASE LIQUID FEED TO LOW RESIDUALS (~ 2 PERCENT)
  - RELATIVELY HIGH VOLUMETRIC FLOW RATES (FILL OTV IN SEVERAL HOUR TRANSFER OPERATION)
  - NO IMPOSED GRAVITY/SETTLING REQUIREMENTS
  - INCORPORATE ADEQUATE METEOROID PROTECTION
  - GAUGE MASS TO DETERMINE TIME FOR RESUPPLY
  - DESIGNED FOR ADEQUATE SUPPLY PRESSURE TO TRANSFER FLUID
  - MINIMIZE CONTAMINANT AND PARTICLE BUILD-UP IN TANK OVER TIME
  - MINIMIZE SLOSH AND ITS IMPACT ON SPACE STATION ATTITUDE CONTROL
- THERMAL CONTROL
  - PROVIDE 180 DAYS OF STORAGE, MAINTAINING DESIRED SATURATED CONDITIONS (QUALITY OF LIQUID WITH TIME)
  - MINIMIZE BOILOFF LOSSES
  - INTEGRATE THERMAL CONTROL OF ENTIRE HYDROGEN/OXYGEN SYSTEM (COUPLED HEAT EXCHANGER APPROACH)
- FLUID TRANSFER/RESUPPLY
  - PROVIDE CAPABILITY TO TOP PARTIALLY FULL TANKS
  - GAUGE MASS/METER MASS FLOW TO CONTROL OPERATIONS
  - MINIMIZE NEED FOR RAPID VENTING INTO SPACE OF SIGNIFICANT QUANTITIES OF VAPOR (INCLUDING NON-CONDENSIBLES). MAY REQUIRE SEPARATE 'CATCH TANKS' OR RELIQUEFACTION.
  - RESUPPLY ON 90-DAY INTERVALS
  - INCORPORATE DIAGNOSTICS FOR LEAK DETECTION, OPERATIONAL CONTROLS, ETC.

Figure 3

This chart shows a depot concept with a coupled tank thermodynamic vent system. In this concept we refrigerate at the liquid hydrogen tank vapor-cooled shield. Circulators and a radiator panel are included as part of the heat rejection system. Hydrogen is fed as single-phase fluid from the total communication acquisition device, through an expander where it becomes two-phase fluid, and into a thermodynamic vent heat exchanger. The heat exchanger is attached to the vapor-cooled shield. Once the fluid reaches the end of the heat exchanger on the vapor-cooled shield, it is routed to a heat exchanger that could either be on the liquid oxygen supply tank or on the shield around the tank. The hydrogen is then vented overboard or reliquified. By coupling the heat exchanger of the hydrogen tank to the oxygen tank we can minimize or prevent the boiloff of oxygen.

DEPOT CONCEPT-COUPLED TANK WITH REFRIGERATION AT LH_2 TANK VCS

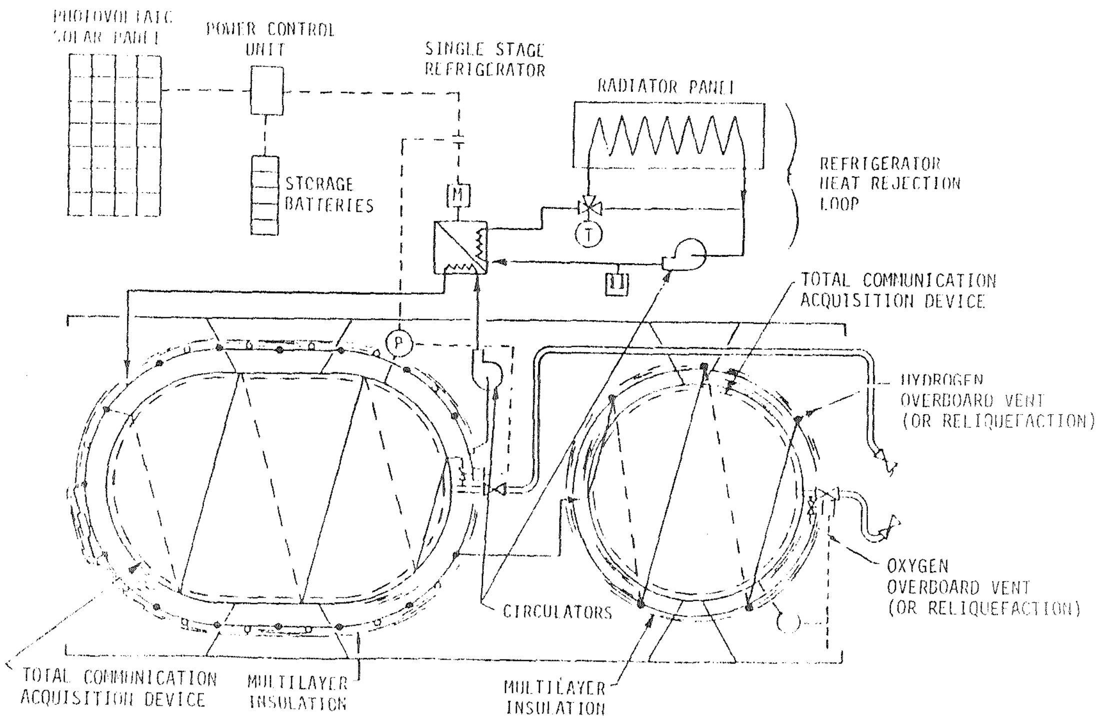

Figure 4

A prioritization of the cryogenic fluid management technologies relative to the resupply tanker, the space station depot and the space-based OTV was performed. This chart shows the various categories that were considered and a description of the prioritization criterion for each. Category one includes technologies that must be addressed as an enabling technology. Category two contains technology items which must be addressed for efficient design, such as minimizing weight, minimizing losses of fluid such as boiloff losses, or maximizing performance. Category three includes technologies which provide an intermediate performance gain. Categories four and five represent technology categories which can either be designed around with minimum impact or are not required for the application.

REQUIREMENTS PRIORITIZATION CATEGORIES

CATEGORY	DESCRIPTION
1	ITEM MUST BE ADDRESSED - ENABLING TECHNOLOGY
2	TECHNOLOGY MUST BE ADDRESSED FOR EFFICIENT DESIGN (MIN WEIGHT, MIN LOSSES, MAX PERFORMANCE, ETC.)
3	TECHNOLOGY WHICH PROVIDES INTERMEDIATE PERFORMANCE GAINS
4	TECHNOLOGY REQUIREMENTS THAT CAN BE DESIGNED AROUND WITH MINIMUM ADVERSE IMPACT
5	TECHNOLOGY NOT REQUIRED FOR APPLICATION

Figure 5

The liquid storage and supply technology priority assessment is shown for the resupply tanker, the space station depot, and the space-based OTV. For fluid management systems direct outflow with settling for the space-based OTV is an enabling technology. A total communication device which allows contact of the liquid in all locations of the tank is an enabling technology for the space station depot because settling would be disruptive to the stabilization of the space station. Autogenous pressurization is an enabling technology for the depot because the interjection of a non-condensible pressurant, such as helium, to assist in transferring the cryogen from the space station to user tanks, is disruptive to resupplying the depot as a partially full tank. The filling of a partially full tank is discussed in more detail on a later chart that addresses the transfer/resupply priority assessment. Mass gaging and instrumentation are key technologies for the resupply tanker and the depot because of the control required for the transfer process.

LIQUID STORAGE/SUPPLY TECHNOLOGY REQUIREMENTS PRIORITY ASSESSMENT

	APPLICATIONS		
	RT	SS DEPOT	SB OTV
• FLUID MANAGEMENT SYSTEMS			
ACQUISITION/EXPULSION SYSTEMS			
DIRECT OUTFLOW WITH SETTLING	5	5	1
TOTAL COMMUNICATION DEVICE	2	1	5
PARTIAL COMMUNICATION DEVICE	5	5	2
PRESSURIZATION SYSTEMS			
AMBIENT HELIUM	2	2	2
CRYO-COOLED HELIUM	2	2	2
AUTOGENOUS	3	1	3
SLOSH CONTROL SYSTEMS	4	2	3
- - - - - - - - - - - - - - - - - -			
• ADDITIONAL TECHNOLOGY ISSUES			
START TRANSIENTS	4	4	3
OUTAGE/PULLTHROUGH	3	3	3
MASS GAGING/INSTRUMENTATION	1	1	2
NON-CONVENTIONAL TANKAGE	3	5	2

Figure 6

A total communication acquisition device is shown schematically in this chart. Fine-mesh screen channels form the total communication device, allowing communication with all regions of the tank. In low-g the liquid tends to fill in between the tank wall and the channel, and therefore allows expulsion to a very small residual. Channel devices similar to this have been considered for applications to tanks as large as 14-foot in diameter. Even with the low surface tension of liquid hydrogen good expulsion efficiencies are obtainable.

TOTAL COMMUNICATION LIQUID ACQUISITION DEVICE

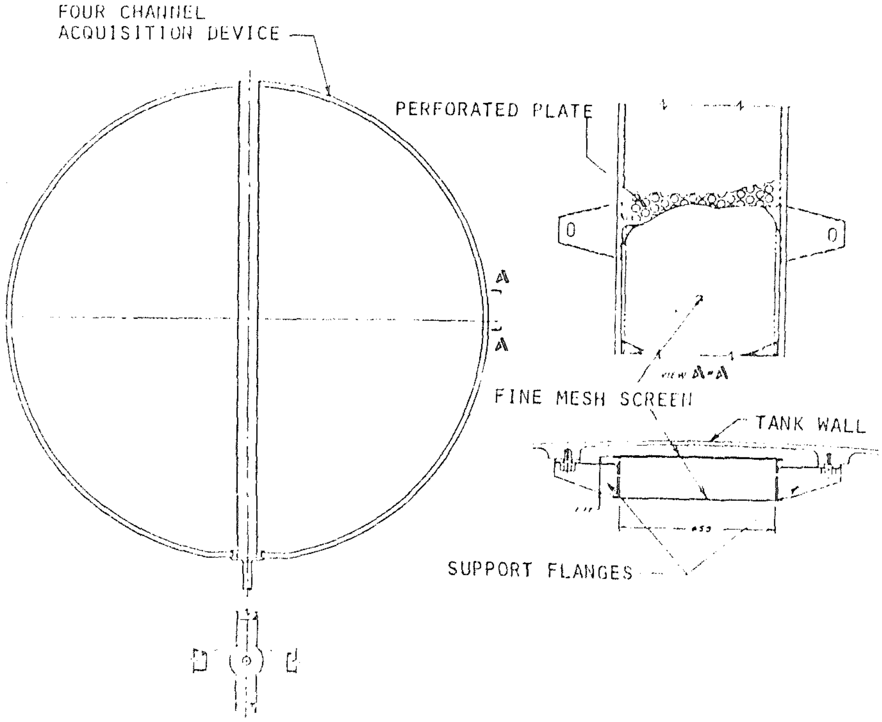

Figure 7

The technology priority assessment for thermal control is shown here. For the resupply tanker, thermal protection systems that will allow efficient ground operation and lightweight tankage for launch operations would include purged MLI and foam underneath the purged MLI. The foam underneath the purged MLI allows gaseous nitrogen rather than gaseous helium to be used as the purge gas and this decreases the heat flux by about a factor of six while the tanks are loaded and still on the ground. Internal and external heat exchangers as part of thermodynamic vent systems are key technologies in terms of effective thermal control and minimizing boiloff losses.

Some additional thermal control technology issues are listed here. One important issue is the degradation that may occur over time with the insulation system. Insulation can be designed to a prescribed requirement and if that insulation degrades significantly over time severe thermal performance impacts will result.. It is important to pay attention to contamination, meteoroid impacts, atomic oxygen degradation on the insulation performance. Thermal conditioning of the outflow is also important to preserving the quality or condition of the fluid that is being transferred from the resupply tanker into the depot or from the depot into the space-based OTV.

THERMAL CONTROL TECHNOLOGY REQUIREMENTS PRIORITY ASSESSMENT

	APPLICATIONS RT	SS DEPOT	SB OTV
• THERMAL PROTECTION SYSTEMS			
VACUUM JACKET/INSULATION (DEWAR)	4	3	5
PURGED - MLI	1	3	3
FOAM - MLI	1	3	3
• THERMAL MANAGEMENT SYSTEMS			
THERMODYNAMIC VENT SYSTEMS			
- INTERNAL HEAT EXCHANGER	3	1	1
- EXTERNAL HEAT EXCHANGER (INCLUDING VAPOR-COOLED SHIELD)	3	1	1
- COUPLED HEAT EXCHANGER (VENT FREE STORAGE)	3	3	2
- PARA-TO-ORTHO CONVERSION	3	2	2
DIRECT TANK VENTING WITH SETTLING	5	5	2
REFRIGERATION SYSTEMS	5	3	3
• ADDITIONAL TECHNOLOGY ISSUES			
INSULATION REUSABILITY (NON-DEWAR)	1	4	2
INSULATION DEGRADATION (WITH TIME)	5	1	1
SUPPORTS/LINES/PENETRATION HEAT LEAKS	2	2	2
THERMAL ACOUSTIC OSCILLATIONS	3	2	2
CONVECTION CONTROL	4	2	2
THERMAL CONDITIONING OUTFLOW	1	3	1

Figure 8

This chart illustrates schematically the tanker concept with a total communication liquid acquisition device in the supply tank, and a depot which would also contain a total screen acquisition device. For the depot we have shown the hydrogen autogenous pressurant system which would be preferred considering the topping of a partially full tank by a resupply tanker. We would then preclude the problem of having to vent noncondensibles to precondition the tank and lower the pressure to allow the transfer process to occur. The resupply tanker could use a helium pressurant system to expel liquid in the tanks since it will be taken back to the ground, expelled and reconditioned for filling for the next resupply operation.

FLUID TRANSFER/RESUPPLY - TANKER TO DEPOT

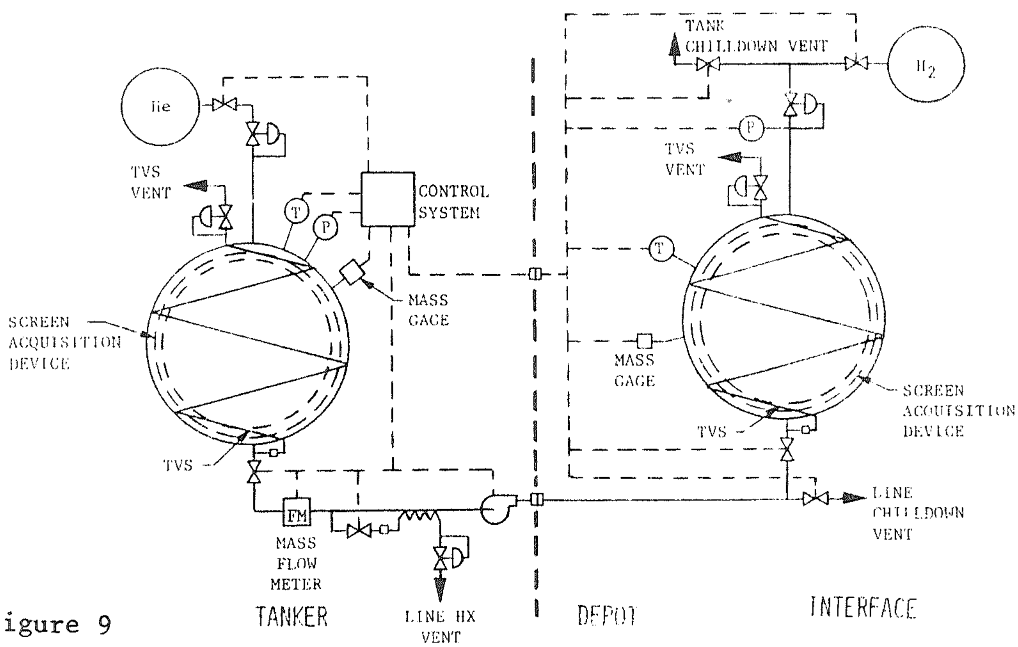

Figure 9

This chart depicts the depot and the space-based OTV. Again, the depot has a spherical tank with a total communication device and a gaseous hydrogen autogenous pressurant system. The space-based OTV would likely be non-spherical tank similar to the cylindrical tank shown, possibly having a start basket or partial acquisition device for fluid management. Elements of the system would again consist of mass gages and mass metering devices to allow us to control the operation and status when we had filled the system. The space-based OTV would likely have some kind of chill and fill system that would allow chilldown of the initially dry and empty tank prior to the filling of the tank. The filling of the tank would likely be accomplished by what's called a no-vent fill operation.

FLUID TRANSFER/RESUPPLY - DEPOT TO SPACE-BASED OTV

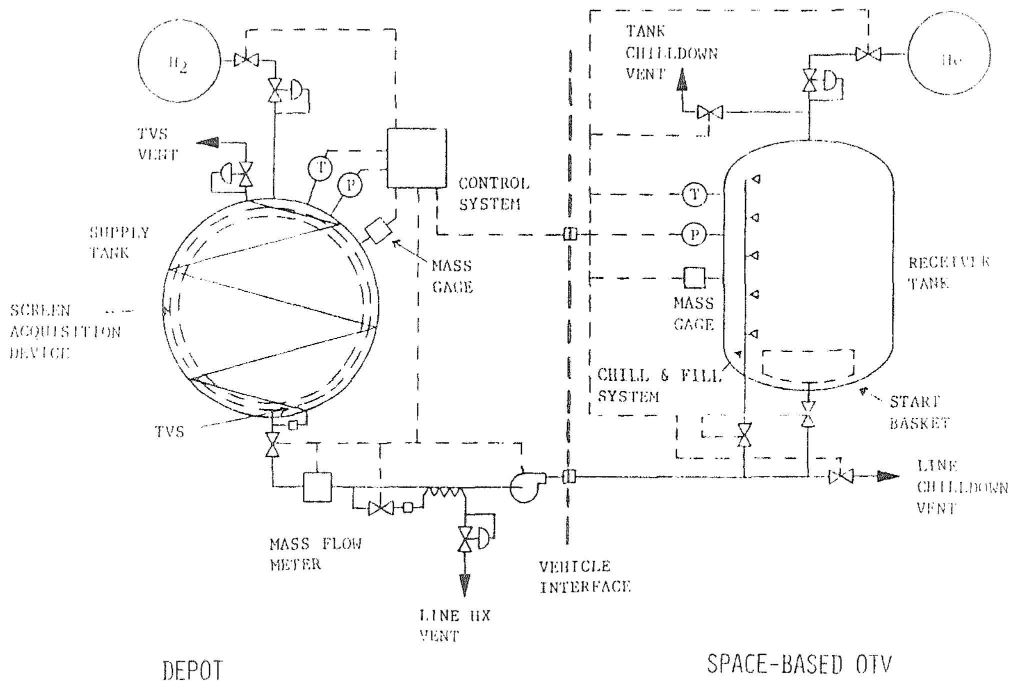

Figure 10

This chart shows fluid transfer/resupply technology requirements. If we have initially empty tanks, as we might have with a space-based OTV, then we would have to go through a childown prior to accomplishing a no-vent fill. For the space station depot we would likely be topping a partially full tank. No-vent fill is a preferred resupply technology and venting of noncondensibles is undesirable. Transfer line chilldown and quick disconnects represent technologies that are enabling for this kind of operation. As can be seen from this chart compared to the charts for liquid storage and supply and thermal control, there is much more enabling technology that is required.

The mass gaging and quality metering technology issues associated with fluid transfer and resupply are again identified as enabling. Long-term effects are important, but the storage periods for the space-based operations have not been clearly defined and the repeated cycling determination aspects have not been delineated. These could become enabling based on the particular operational scenarios proposed and implemented.

FLUID TRANSFER/RESUPPLY TECHNOLOGY REQUIREMENTS PRIORITY ASSESSMENT

	APPLICATIONS		
	RT	SS DEPOT	SB OTV
• RECEIVER TANK			
EMPTY			1
CHILLDOWN			1
ACQUISITION DEVICE FILL			1
VAPOR COLLAPSE			1
PURGE, NON-CONDENSIBLES			2
NO-VENT FILL			1
PARTIALLY FULL		1	
VENTING NON-CONDENSIBLES		2	
NO-VENT FILL		1	
VENTED FILL		2	
• TRANSFER LINE			
CHILLDOWN	1	1	1
QUICK DISCONNECT	1	1	1
• ADDITIONAL TECHNOLOGY ISSUES			
MASS GAGING	1	1	1
MASS/QUALITY METERING	1	1	1
PUMP VS. PRESSURIZED TRANSFER	2	2	2
LONG TERM EFFECTS			
REPEATED CYCLING DEGRADATION	3	3	3
CONTAMINATION	2	2	2

Figure 11

This chart summarizes the fluid management technology requirements based upon the previous prioritization assessment. The last item addresses an issue of scavenging propellant from the Shuttle External Tank (ET) following boost. Since topping a nearly full depot tank with propellant scavenged from the ET at different saturation conditions may be difficult, it may be required to handle scavenged propellants in separate tanks.

ATTACHED OR FREE-FLYER DEPOT FLUID MANAGEMENT TECHNOLOGIES

- TOTAL COMMUNICATION LIQUID ACQUISITION DEVICE
- AUTOGENOUS PRESSURIZATION
- INTERNAL AND COUPLED THERMODYNAMIC VENTS FOR THERMAL (PRESSURE) CONTROL AND PROPELLANT CONDITIONING
- THERMAL CONTROL SYSTEM PROTECTION FROM ENVIRONMENT (CONTAMINATION, ATOMIC OXYGEN, ETC) TO PREVENT DEGRADATION WITH TIME
- MASS GAUGING, INSTRUMENTATION, CONTROL SYSTEM AND DIAGNOSTICS
- MUST BE CAPABLE OF BEING RESUPPLIED AS A PARTIALLY FULL TANK AS WELL AS A DRY, WARM TANK
- DIFFERENT SIZE TANKS MAY BE REQUIRED TO HANDLE SCAVENGED PROPELLANT SINCE TOPPING NEARLY FULL DEPOT TANKS WITH PROPELLANT AT A DIFFERENT SATURATION CONDITION MAY BE DIFFICULT.

Figure 12

A Cryogenic Fluid Management Facility (CFMF) has been planned to obtain much of the data that has been discussed for attached depot operations. The purpose of the facility is to carry a reusable test bed into space attached to the Orbiter to obtain basic data on cryogenic fluid management. The facility uses liquid hydrogen as the test fluid and is designed for seven Shuttle flights. The detailed design of the facility is nearly complete, and mission planning is now proceeding for three flights. The facility will provide data to allow low-g verification of fluid and thermal models that encompass methods of integrating pressure control, liquid acquisition device and liquid transfer concepts. The experimental data will provide the data base for design criteria applicable to subcritical cryogenic systems in space and will provide the technology required to efficiently and effectively manage those cryogens.

CRYOGENIC FLUID MANAGEMENT FACILITY (CFMF)

DESIGN, FABRICATE, AND CARRY INTO SPACE A REUSABLE TEST BED WHICH WILL BE UTILIZED TO PROVIDE THE TECHNOLOGY REQUIRED TO EFFICIENTLY AND EFFECTIVELY MANAGE CRYOGENS IN SPACE

- LIQUID HYDROGEN TEST FLUID
- DESIGNED FOR SEVEN SHUTTLE FLIGHTS
  CURRENT MISSION PLANNING FOR THREE FLIGHTS
- LOW-G VERIFICATION OF FLUID AND THERMAL MODELS - METHODS OF INTEGRATING PRESSURE CONTROL, LIQUID ACQUISITION AND LIQUID TRANSFER CONCEPTS.
- ESTABLISHMENT OF DESIGN CRITERIA FOR SUBCRITICAL CRYOGENIC SYSTEMS IN SPACE

Figure 13

The Cryogenic Fluid Management Facility will provide enabling technology for the space station cryogenic fluid elements, and associated cryogenic users such as the space-based OTV. The emphasis of the facility is on liquid acquisition devices and thermodynamic vent systems and how they can be integrated together for effective storage and thermal control, and on liquid transfer and resupply operations. The seven-day Shuttle operation with an attached payload does not permit thorough testing of long-term storage effects. Current planning is for three missions with the first launch about mid-1988, and subsequent launches on six to nine month intervals.

CFMF APPLICABILITY TO CRYO FLUID MANAGEMENT TECHNOLOGY NEEDS

- CRYOGENIC FLUID MANAGEMENT FACILITY WILL PROVIDE ENABLING TECHNOLOGY FOR SPACE STATION CRYO FLUID ELEMENTS AND ASSOCIATED CRYO USERS SUCH AS THE OTV.
- EMPHASIS IS ON LIQUID ACQUISITION DEVICES, THERMODYNAMIC VENT SYSTEMS, AND LIQUID TRANSFER/RESUPPLY
- MAXIMUM SEVEN DAY SHUTTLE FLIGHT DOES NOT PERMIT THOROUGH TESTING OF LONG TERM STORAGE CONCEPTS.
- CURRENT PLANNING IS FOR THREE MISSIONS; FIRST LAUNCH ABOUT MID-1988

Figure 14

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PROPELLANT TRANSFER - TETHERED DEPOT

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Spacebasing of orbital transfer vehicles at a space station will require a depot that will safely and efficiently store and transfer the resupply propellants. In order to transfer propellants, a method to effectively acquire only liquid and vent only gas must exist. Unfortunately, the current methods of transferring propellant, under the zero 'G' condition of random liquid orientation, have several weaknesses. A method that produces a low gravity to settle propellants would bypass these weaknesses, while allowing ground-like operations. This low gravity can be passively produced using gravity gradient techniques. A satellite with a large length to diameter ratio, such as a depot attached to a space station with a tether, will stabilize along an earth radial because of an outward acceleration proportional to the distance from the satellite's center of gravity. Analysis indicates that liquid can be settled with relatively short tether lengths. However, longer tether lengths may be required to prevent excessive residuals due to suction dip, to allow transfer using gravity feed, and to allow slosh control.

Currently the tethered refueling depot concept is being studied by Martin Marietta Aerospace under contract to NASA, Johnson Space Center. The objectives of this contract are to determine the feasibility, design requirements, and operational limitations of a tethered refueling depot with special emphasis on slosh control.

The purpose of this presentation is to introduce the tethered refueling depot concept to the orbital transfer vehicle community. This should allow the concept's effects on propellant resupply, space station, and orbital transfer vehicle to be given some consideration during preliminary design.

DEPOT COMPARISON

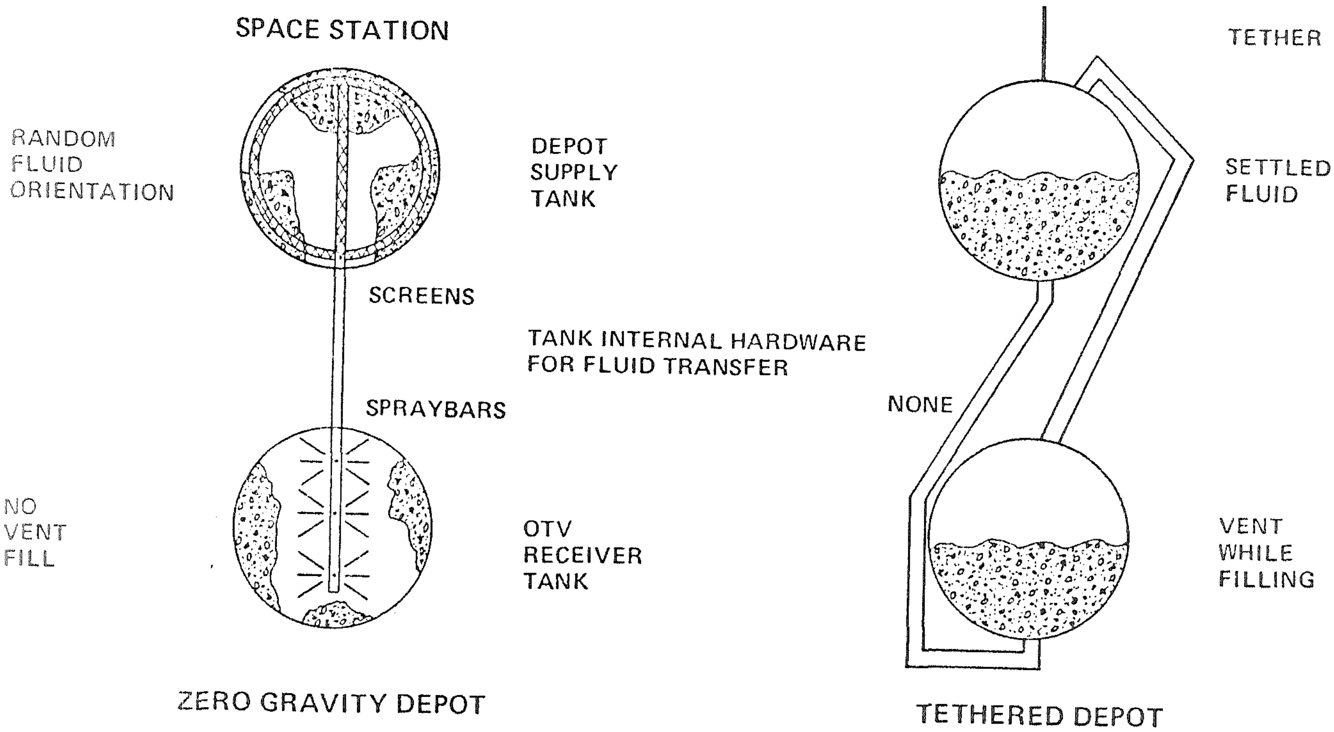

Figure 1

A zero gravity depot is attached directly to the space station. Because of the random orientation of fluid phases this depot needs hardware internal to the tanks for acquiring liquid from the supply tank and filling the receiver tank without venting. The tethered depot is connected by a tether to the space station. Because of the settled condition of the fluid, the tethered depot can acquire liquid from the supply tank and vent gas during fill of the receiver tank without internal hardware. The supply and receiver tank ullages can be interconnected to equalize the pressure in both tanks or autogenously pressurize the receiver tank, which will also reduce the venting of propellant into space.

CURRENT TECHNIQUES TO SEPARATE FLUID PHASES ON-ORBIT

- TYPES
  - PHYSICAL BARRIERS (BLADDERS AND DIAPHRAGMS)
  - SURFACE TENSION DEVICES (VANES AND SCREENS)
  - ACTIVE PROPELLANT SETTLING (PROPULSIVE AND ROTATIONAL)
- PROBLEMS
  - COMPLEX HARDWARE
  - SUBJECT TO FATIGUE WITH REUSE
  - COMPATIBILITY PROBLEMS WITH PROPELLANT
  - SENSITIVE TO GAS FORMATION WITH CRYOGENS
  - COMPLEX VENTING AND VAPOR COLLAPSE TECHNIQUES
  - ACTIVE SETTLING INCOMPATIBLE WITH SPACE STATION CONTROL
- BASIC PROBLEM: RANDOM ORIENTATION OF LIQUID IN ON-ORBIT 'ZERO-GRAVITY' ENVIRONMENT

Figure 2

Currently two types of techniques, passive and active, are used to separate fluid phases on-orbit. Passive techniques include physical barriers, such as bladders and diaphragms, and surface tension devices, such as vanes and screens. These devices are internal to the propellant tanks which can present problems for long term reusability on a space station depot. They are complex to design, fabricate, and install; subject to fatigue with reuse; and difficult to repair and replace. Hardware material selection will be limited because of incompatibility with oxidizers and cryogens. Hardware will increase sensitivity to gas formation in cryogens which will increase the problem of separating the fluid phases. Surface tension devices are meant to insure that liquid is acquired, but they can't insure the gas position for venting; therefore, complex venting and vapor collapse techniques are required. Alternatively, active propellant settling, propulsive thrusting or rotation of a vehicle with offset propellant tanks, can force the denser liquid in the direction of the acceleration. Active propellant settling is incompatible with the space station requirement for control of attitude and orbital position.

ARTIFICIAL GRAVITY AT TETHERED SATELLITES

- BODY FORCES CANCEL AT CENTER OF GRAVITY (C.G.)
  - "ZERO GRAVITY"
- NET BODY FORCE OFF THE C.G.
  - GRAVITY FORCE INCREASES TOWARD EARTH
  - CENTRIFUGAL FORCE INCREASES AWAY FROM EARTH
  - CENTRIFUGAL AND GRAVITY FORCES IN OPPOSITE DIRECTION
  - BODY FORCE POINTS AWAY FROM C.G.
- TENSION IN STRUCTURE REACTS AGAINST BODY FORCE
  - "ARTIFICIAL GRAVITY"

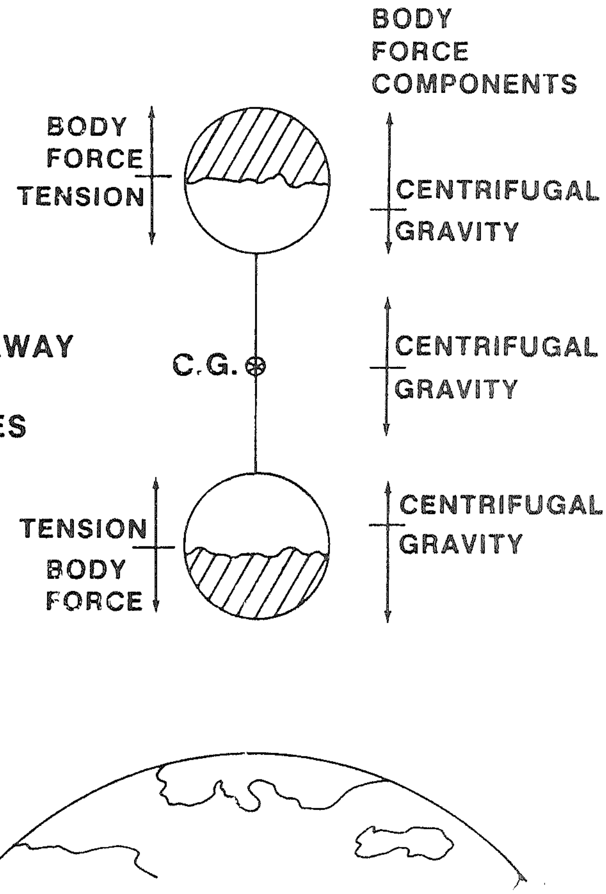

Figure 3

The cancelling of the centrifugal and gravity body forces for a tethered satellite in a circular orbit, "zero gravity", occurs only at the center of gravity. The gravity force is stronger toward the earth, while the centrifugal force is stronger away from the earth. The net body force, when not at the center of gravity, points away from the satellite's center of gravity along an earth radial to be reacted against by tension in the depot structure and tether resulting in "artificial gravity". This artificial gravity will stabilize a large length to width object, such as a tethered satellite, pointing at the earth. This stabilization is called "gravity gradient stabilization".

ARTIFICIAL GRAVITY FOR STATIC TETHER

ARTIFICIAL GRAVITY AT TETHER END

$$\frac{A}{G_O} = \frac{3^* \; TL \; ^*R_O{}^{**}2}{K^* \; R_C{}^{**}3}$$

- **APPARENT ACCELERATION**

  $A = C\text{-}G$

  - **CENTRIFUGAL ACCELERATION**

    $C = V^{**}2/R$

  - **GRAVITATIONAL ACCELERATION**

    $G = G_O \; (R_O/R)^{**}2$

  - **VELOCITY OF ANY POINT ON TETHER**

    $V = V_C{}^*R/R_C$

  - **SATELLITE VELOCITY**

    $V_C = \sqrt{G_C{}^*R_C}$

SUBSCRIPTS

O = EARTH'S SURFACE
C = SATELLITE CENTER OF GRAVITY

PARAMETERS

R = RADIUS FROM EARTH'S CENTER
TL = TETHER LENGTH FROM C.G.
K = 6076 FT/NM

Figure 4

The apparent acceleration that a point in orbit sees is the differance between the centrifugal acceleration, which is a function of velocity and radius from the earth's center, and the gravitational acceleration, which is a function of the distance from the earth's center. For a static tether, the velocity for any point on the tether can be found as a function of its distance from the earth's center,which can be related to the tether length from the satellite's center of gravity which is a known distance from the earth's center. The artificial gravity, the ratio of apparent acceleration to the earth's surface gravity, is found to be a direct function of the tether length.

TETHER LENGTH VS. ARTIFICIAL GRAVITY

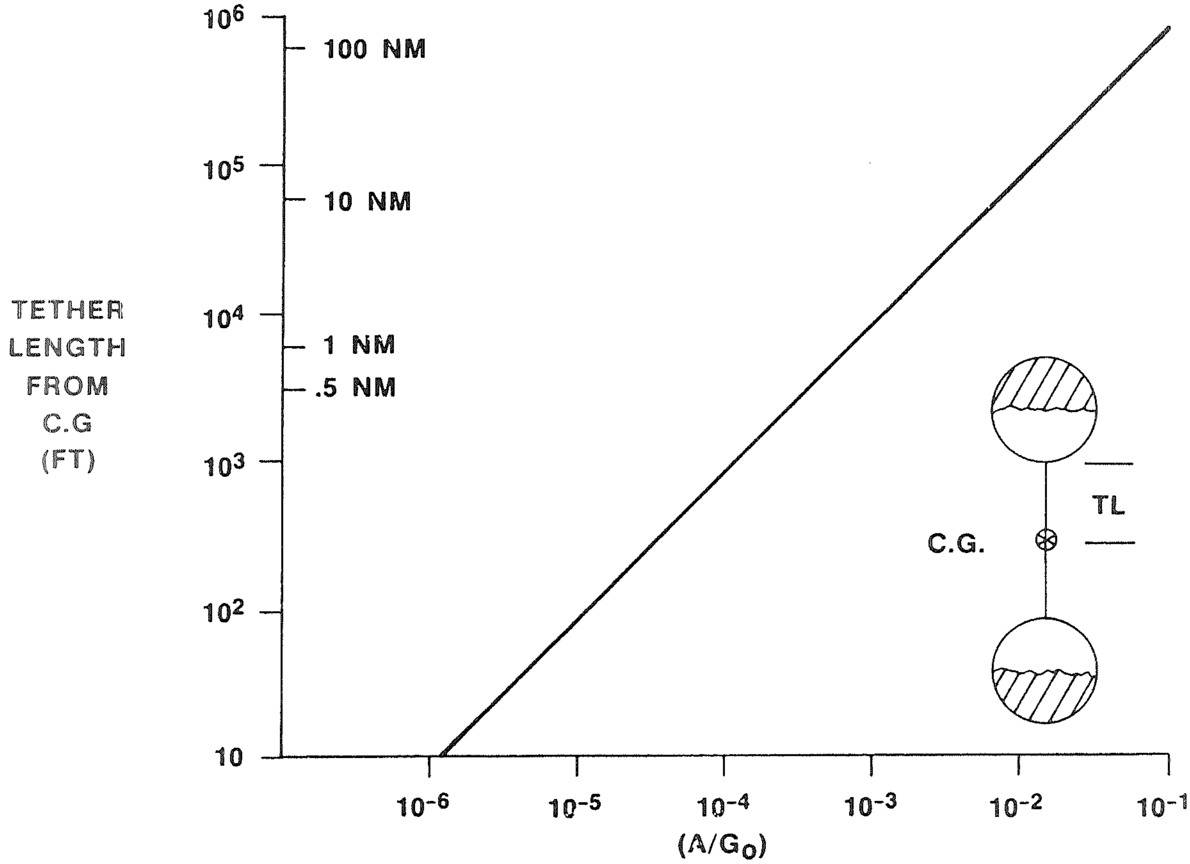

Figure 5

For a static vertical tether the artificial gravity is $7.06*10^{-4}$ g/nm or $1.16*10^{-7}$ g/ft of tether length from the satellite center of gravity.

A/Go	TL
10^{-6}	8.6 ft
10^{-5}	86 ft
10^{-4}	860 ft
10^{-3}	1.4 nm
10^{-2}	14 nm
10^{-1}	140 nm

CRITERIA FOR FLUID SETTLING

- DESCRIPTION OF FLUID SETTLING PARAMETER

$$\text{BOND NUMBER (Bo)} = \frac{\text{ACCELERATION FORCE}}{\text{SURFACE TENSION FORCE}} = \frac{\rho * A * D**2}{4 * G_C * \sigma}$$

ρ = FLUID DENSITY (LBM/FT**3)
A = ACCELERATION (FT/SEC**2)
D = EFFECTIVE TANK DIAMETER (FT)
G_C = 32.174 LBM *FT/LBF/SEC**2
σ = SURFACE TENSION (LBF/FT)

- BOND NUMBER CRITERIA

Bo $<$ 1	SURFACE TENSION DOMINATES ACCELERATION, THEREFORE NO FLUID SETTLING
1 $<$ Bo $<$ 10	TRANSITION ZONE
Bo $>$ 10	ACCELERATION DOMINATES SURFACE TENSION, THEREFORE FLUID SETTLES
Bo = 50	ALLOWS RELATIVELY FLAT FLUID PHASE INTERFACE (CHOSEN AS MINIMUM BOND NUMBER FOR ANALYSIS)

Figure 6

Fluid settling is the basic requirment on a tethered depot to position liquid over the outlet so only liquid will be transfered and only gas vented. The fluid settling parameter is the Bond number which is the ratio of the acceleration force to the surface tension force. The Bond number is primarily a function of the fluid properties, the effective tank diameter, and the acceleration. The Bond number can be used to divide the fluid behavior into a number of zones with a value greater than ten required to settle fluid. A value of fifty was chosen as the minimum for analysis for conservatism while allowing a relatively flat interface.

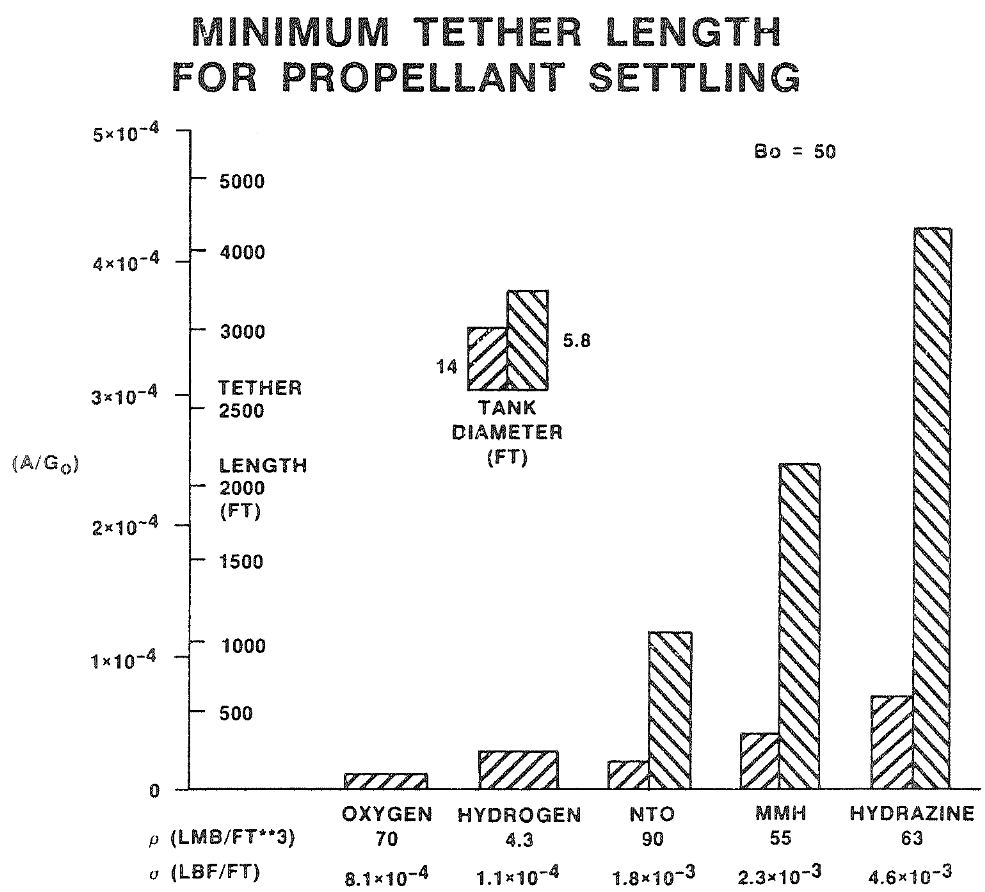

Figure 7

The tether length required for fluid settling is affected by fluid properties and effective tank diameter. Longer tethers are required for having higher surface tension or smaller liquid density. Decreasing the effective tank diameter also requires a longer tether. This is a consideration when looking at baffles for slosh control because because they can change the effective tank diameter. With no baffles the cryogenics would require 230 feet of tether length with 14 foot diameter tanks, the bipropellant storables would require 2180 feet with 5.8 foot tanks, and hydrazine would require 3750 feet with 5.8 foot tanks to settle propellant.

PROPELLANT SLOSHING

- ISSUE
  - SLOSHING SHOULD NOT INTERRUPT FLUID TRANSFER
    - DO NOT UNCOVER SUPPLY TANK OUTLET
    - DO NOT COVER RECEIVER TANK VENT
- POTENTIAL SOLUTIONS
  - INCREASE ACCELERATION LEVEL TO REDUCE SLOSH HEIGHT
  - INTERNALLY DAMP SLOSHING WITH BAFFLES
    - DECREASE IN EFFECTIVE DIAMETER WILL INCREASE REQUIRED ACCELERATION FOR SETTLING
  - EXTERNALLY DAMP SLOSHING
    - MAY NOT BE EFFICIENT
- SLOSHING WILL BE STUDIED BY MARTIN MARIETTA UNDER CONTRACT TO NASA, JOHNSON SPACE CENTER

Figure 8

Even though a liquid will eventually settle it can still slosh when disturbed, changing the position of the liquid relative to the tank outlet and vent. This sloshing should not cause the fluid transfer to be interrupted by uncovering the the supply tank outlet or covering the receiver tank vent. Sloshing can be reduced by increasing the acceleration to limit the slosh height, internally dampening the sloshing with baffles, or externally dampening the sloshing with devices such as reaction wheels, dashpots, etc. Baffles may have a problem because they can reduce the effective diameter of the tank, thus requiring greater tether length to insure settled propellant. The external dampening methods may not be efficient. This problem will be further studied by Martin Marietta under contract to NASA, Johnson Space Center.

RESIDUAL DUE TO SUCTION DIP

- RESIDUAL IS REMAINING LIQUID WHEN SUCTION DIP REACHES OUTLET
- SUCTION DIP HEIGHT (H)

$$H = .51 * D_L \left[\left(\frac{D_T}{D_L} \right)^2 \left(\frac{V^2}{A^* D_L} \right) \right]^{.143}$$

  V = FLUID VELOCITY IN LINE
  A = APPARENT ACCELERATION
  D_L = LINE DIAMETER
  D_T = TANK DIAMETER

- PRIMARY VARIABLE AFFECTING RESIDUALS
  - MASS FLOW
  - LINE DIAMETER
  - ACCELERATION LEVEL

DRAINING TANK

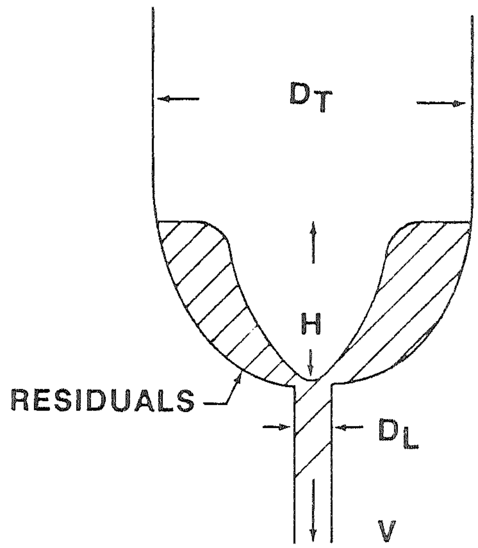

Figure 9

A problem that a tethered depot does have that a zero gravity depot does not have is preventing the vapor from dipping into the outlet due to suction from propellant outflow. This is called "suction dip". For the tethered tank, outflow must be stopped when vapor reaches the outlet, which can result in substantial residual propellant in the tank. The suction dip is primarily a function of mass flow, line diameter, and the acceleration level; therefore, if the steady state mass flowrate during a transfer is decreased to reduce residuals longer transfer times will result. Special outlet provisions such as outlet contouring and screens can limit the affect of the suction dip. However, if no special outlet provisions are used, varying the flowrate, increasing line diameter, and increasing the acceleration levels will be required to minimize transfer times and residuals.

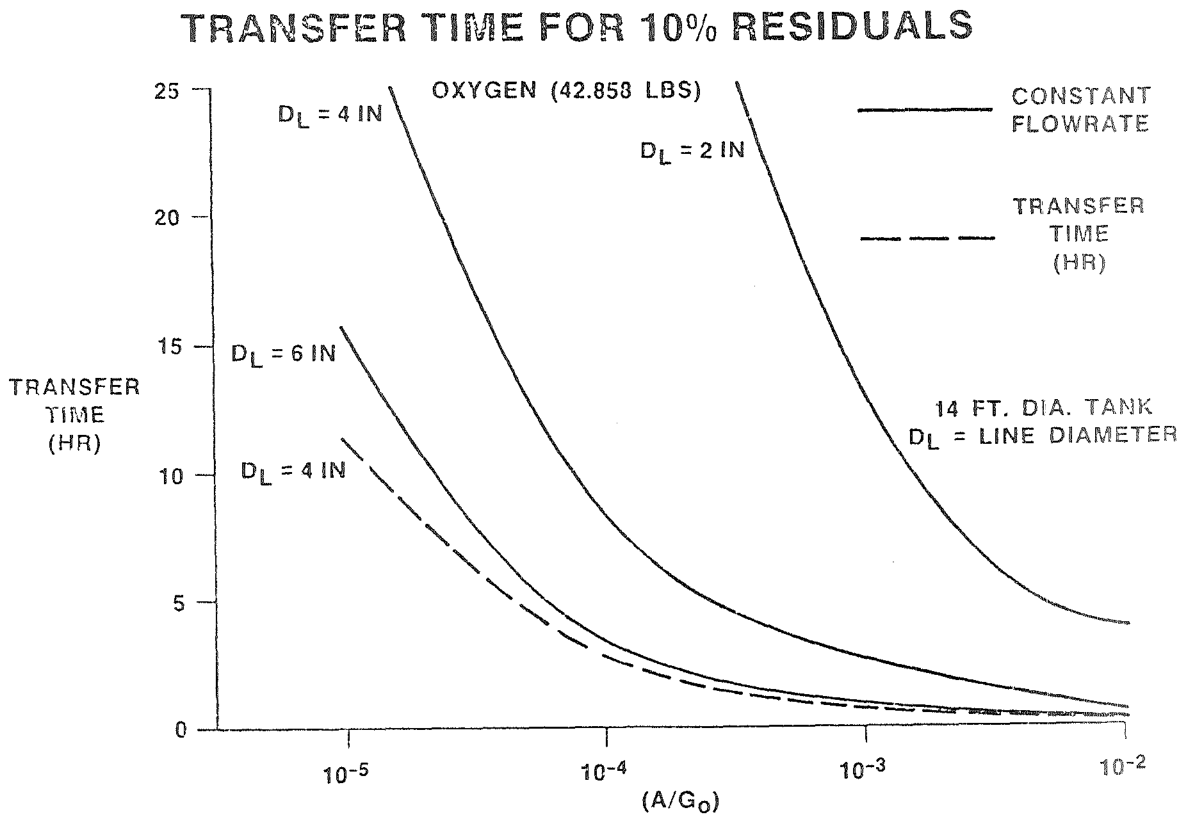

Figure 10

Each plotted point is transfer time assuming a constant flowrate during the transfer; therefore, along the lines of constant diameter shown, the mass flowrate varies continuously. The exception is the two flowrate case shown where high initial flowrate was assumed until the suction dip from that flowrate reaches the outlet. A step change to a lower flowrate was then assumed to occur. Along this line the flowrate also varies in a continuous manner. Assuming a tank with hemispherical ends, no special outlet provisions, and a gas/liquid interface near the wall that has a curvature corresponding to the local bond number, oxygen has longer transfer times than hydrogen for 10% residuals. Larger propellant line diameters reduce the transfer time by increasing the mass flow that will create a given suction dip height. A line diameter of two inches as currently used in the Centaur would probably be unacceptable due to the long transfer times or high acceleration requirements. A four inch line diameter would appear to be acceptable at about an 8 hours transfer time with an acceleration of 10**-4 g; furthermore, if a two step flow is used to limit the flow near the end, the transfer time is reduced to 3 hours. Therefore, a reasonable acceleration level can be used for reasonable residuals.

GRAVITY FEED

- GRAVITY FEED USES HYDROSTATIC HEAD FROM ARTIFICIAL GRAVITY TO COUNTERACT PRESSURE DROP IN FEEDLINE
  - FLOWRATE IS DETERMINED BY BALANCE BETWEEN HYDROSTATIC HEAD AND PRESSURE DROPS
- HYDROSTATIC HEAD SOURCES
  - VERTICAL LINES
  - PROPELLANT IN TANK
- PRESSURE DROP SOURCES
  - LINE FRICTION
  - COMPONENT LOSSES
  - INLET/OUTLET AND ELBOW LOSSES
- COMPLETE ANALYSIS NEEDS DETAILED FLUID SYSTEM CONFIGURATION
- SIMPLE ANALYSIS LOOKS AT ONLY FLOWRATE FOR VERTICAL PIPE SECTION
  - HYDROSTATIC PRESSURE HEAD FROM VERTICAL LINE ONLY
  - PRESSURE DROP FROM VERTICAL LINE FRICTION ONLY
  - INDEPENDENT OF LINE LENGTH

Figure 11

A tethered depot can possibly use gravity feed as a passive fluid transfer technique. Gravity feed uses hydrostatic head to provide the driving force to counteract the pressure drop associated with a certain mass flow. The hydrostatic head is determined by the vertical separation between the gas/liquid interfaces of the supply and receiver tank and the density of the liquid. The pressure drop results from line friction, component losses, and changes in the direction of the flow. A complete analysis would require a detailed fluid system configuration. However, an idea of the minimum requirement for tether length can be determined by looking at a simple case where all the pressure drop is in vertical lines.

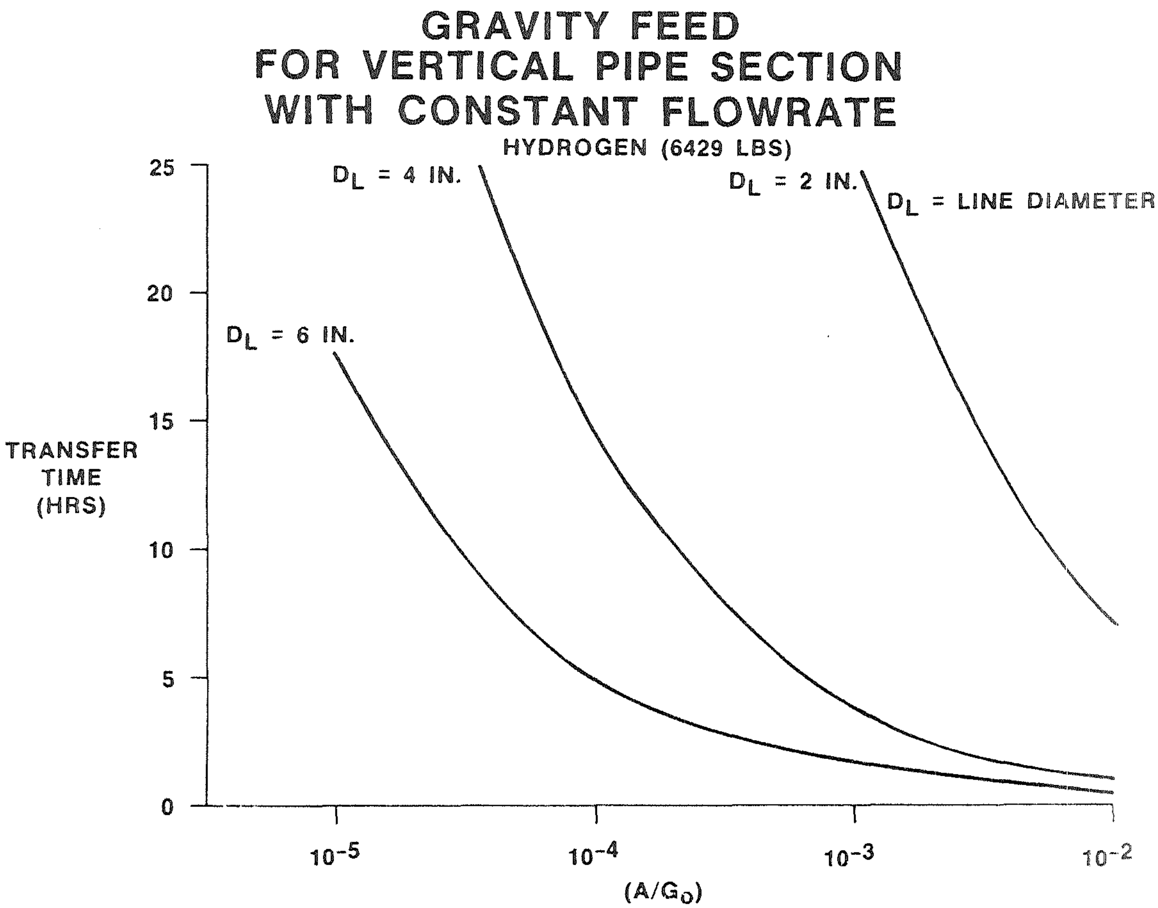

Figure 12

Each plotted point is transfer time assuming a constant flowrate during the transfer; therefore, flowrate varies continuously along each line of constant diameter. Hydrogen is the limiting propellant for determining gravity feed transfer time because of its low density. If 8 hours is assumed to be the maximum acceptable transfer time, the tether length with a line diameter of four inches would be about one half a nautical mile. This simple analysis would say that gravity feed can be used with a reasonable tether length; however, because of the low density of hydrogen and relatively low acceleration levels a tether can produce, a tether will require much more tether length to produce an increased hydrostatic head to compensate for a larger realistic pressure drops. This may cause transfer time to become excessive, although this still may be acceptable if gravity feed is used as a backup mode of operation.

OTHER DEPOT REQUIREMENTS

- DEPOT SHOULD PROVIDE HAZARD CLEARANCE FROM EXPLOSIONS AND CONTAMINATION
- BOTTOM END MASS SHOULD NOT DEORBIT WITH TETHER BREAKAGE
- OPERATIONS BETWEEN THE END MASSES SHOULD NOT BE EXCESSIVELY DIFFICULT
- DEPOT SHOULD NOT ADVERSELY AFFECT TETHER DYNAMICS
- SPACE STATION MAY REQUIRE ZERO GRAVITY FOR MICROGRAVITY LABORATORY

Figure 13

Besides fluid transfer the tethered depot has a number of other requirements. The depot should provide hazard clearance from other space station hardware to prevent catastrophic or long term damage from explosion or contamination. If the tether breaks the bottom end mass should not deorbit to prevent the loss of the bottom end mass and damage or injury on the ground. The operations between the end masses, such as transfer of the OTV, men, etc., should not be excessively difficult so that the depot can be fully utilized. The depot should not adversely affect tether motion to insure safety and space station control. A space station/depot configuration may be required that allows a low acceleration level for a microgravity laboratory at the manned part of the space station. Of these requirements , operations involving transfer of men and materials between the end masses appears to be the hardest to meet.

DEPOT/SPACE STATION CONFIGURATIONS

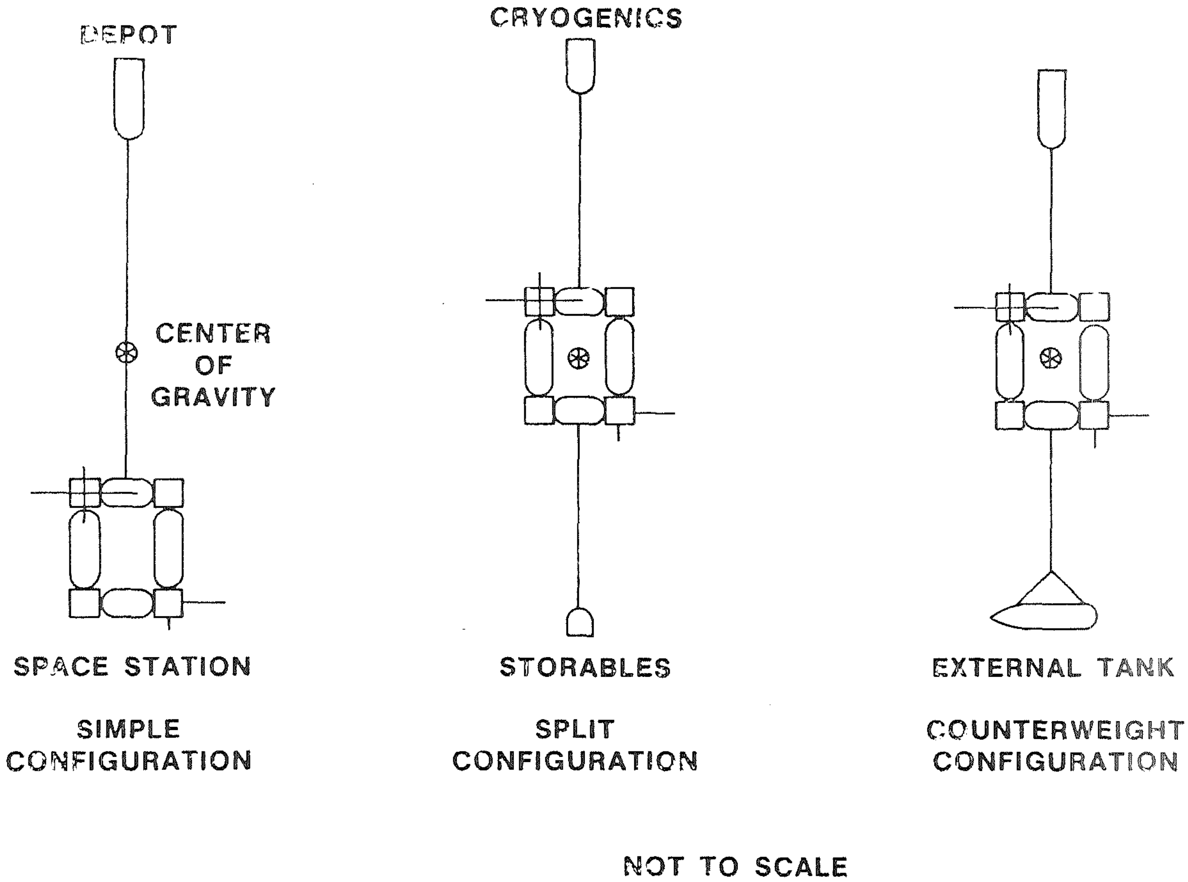

Figure 14

A simple configuration would be a depot,where an OTV would be fueled, attached with a single tether to a space station; however the resulting center of gravity of this system is not at the space station. Options to provide zero gravity at the space station include spliting the depot into cryogenic and storable facilities and tethering in opposite directions from the space station or using a counterweight, such as an external tank or other tethered system. If at least one piece of the depot is the upper mass with the space station kept at the center of gravity of the system, the bottom mass must have sufficient mass so its tether length limitation to prevent deorbit does not result in too little upper mass tether length for depot requirements.

OTV CONDISERATIONS

- FLUIDS INTERFACE WILL BE REQUIRED
  - BOTH FEEDLINE AND VENT DISCONNECTS
  - FORWARD POSITION IF NO PAYLOAD ATTACHED DURING TRANSFER
  - AFT POSITION IF PAYLOAD ATTACHED DURING TRANSFER
    - — CONSIDERATION SHOULD BE GIVEN TO DUAL USE OF VENT AND FEED LINES
- BAFFLES MAY BE REQUIRED TO DAMP SLOSHING
- MINIMUM TANK DIAMETER WILL BE LIMITED TO ENSURE SETTLING
- TANK OUTLET OR VENT MAY BE REQUIRED TO BE OFFSET FROM CENTERLINE
- LARGER FEEDLINES MAY BE REQUIRED FOR GRAVITY FEED
- A LOW GRAVITY FLUID QUANTITY GAGE MAY BE REQUIRED FOR LOADING ACCURACY

Figure 15

SUMMARY

- A TETHER CAN PRODUCE SUFFICIENT ARTIFICIAL GRAVITY TO SIMPLIFY PROPELLANT TRANSFER TO AN OTV FROM AN ON-ORBIT DEPOT
- MARTIN MARIETTA UNDER CONTRACT TO NASA, JOHNSON SPACE CENTER IS STUDYING THE FEASIBILITY, DESIGN REQUIREMENTS, AND OPERATIONAL LIMITATIONS OF USING A TETHER FOR PROPELLANT TRANSFER
  - PRIMARY CONCERN IS SLOSHING
- OPERATIONS TO TRANSFER MEN AND MATERIAL BETWEEN END MASSES WILL REQUIRE EXAMINATION
  - POSSIBLE TRANSFER ALONG TETHER
- TETHERED REFUELING DEPOT APPEARS TO HAVE MINIMAL EFFECT ON OTV

Figure 16

N85-17007

OTV FLUID MANAGEMENT SYSTEMS

L. Hastings
NASA Marshall Space Flight Center

Design, performance, and technology issues associated with reduced gravity propellant management for Orbital Transfer Vehicles (OTV's) have been reviewed. The inspace cryogenic management state-of-technology will significantly affect the overall confidence level associated with a resupply mission and propulsion performance. Thus, although mission requirements are frequently used to determine technology requirements, it is also apparent that technology availability drives mission requirements. Cryogen resupply sequences, timelines, controls, and associated crew involvement are all affected by the technology state. Additionally, OTV propellant tankage configurations, tankage thermodynamic conditions, acceleration environment, propulsion interfaces, and instrumentation are significant factors. Basic propellant transfer phases examined that drive orbital servicing requirements include: (1) tankage preconditioning (purging, venting, etc.), (2) tankage chilldown, and (3) propellant fill. Propellant management support of the OTV propulsion phases includes engine restart requirements (pressurization, chilldown, burn duration, etc.) and orbital coast between engine burns. Technology activities in support of identified technology issues are reviewed.

SPACE-BASED OTV PROPELLANT REQUIREMENTS

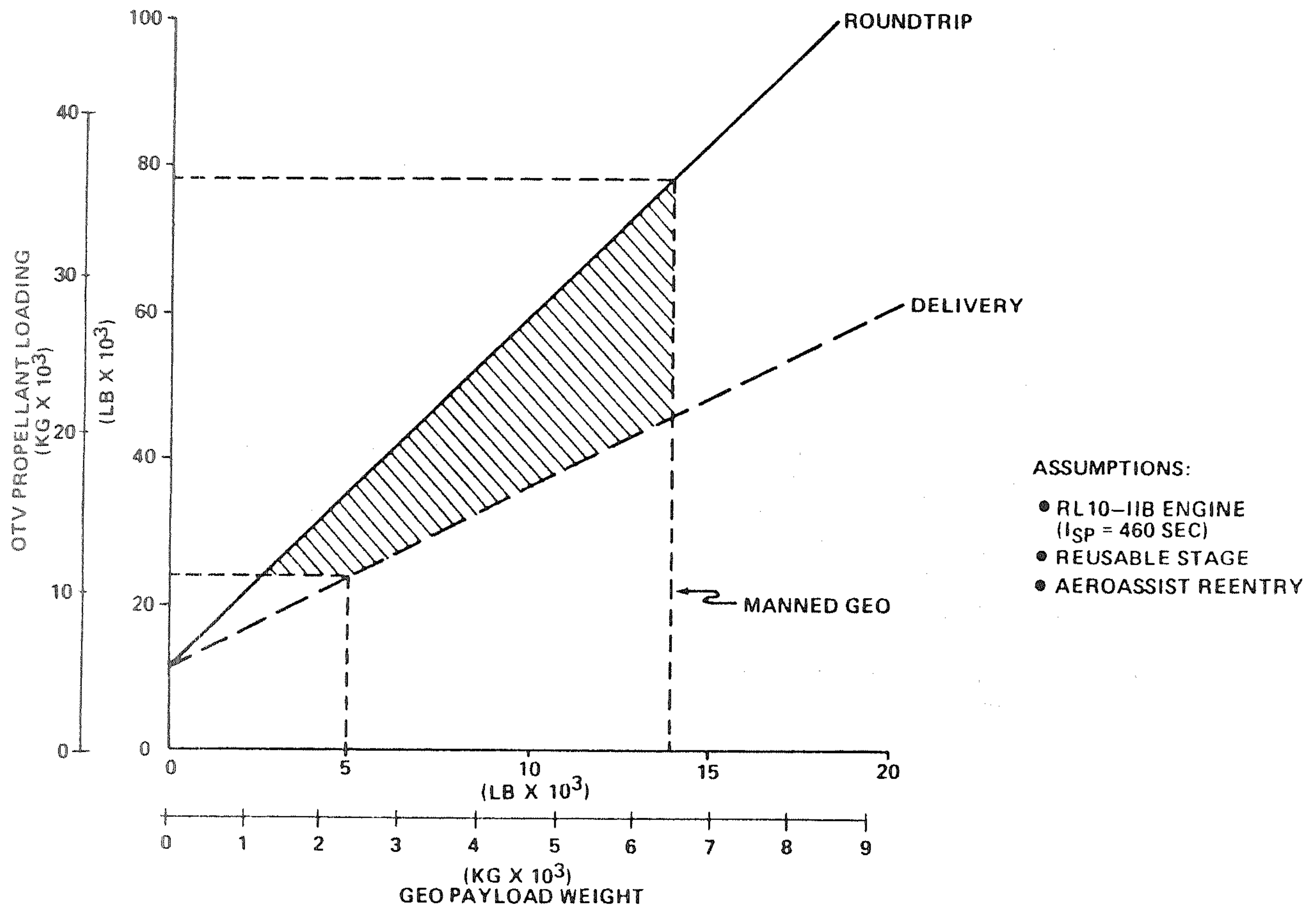

Figure 1

The average payload weight required to be transported from LEO to GEO will be in the range of 5,000 to 14,000 pounds. The upper range of payloads is normally associated with manned GEO roundtrip missions. The resultant propellant requirements, based on these payload weights, ranged from approximately 24,000 to 78,000 pounds. The chart on the opposite page graphically portrays these requirements.

OTV CRYOGENIC MANAGEMENT CONSIDERATIONS

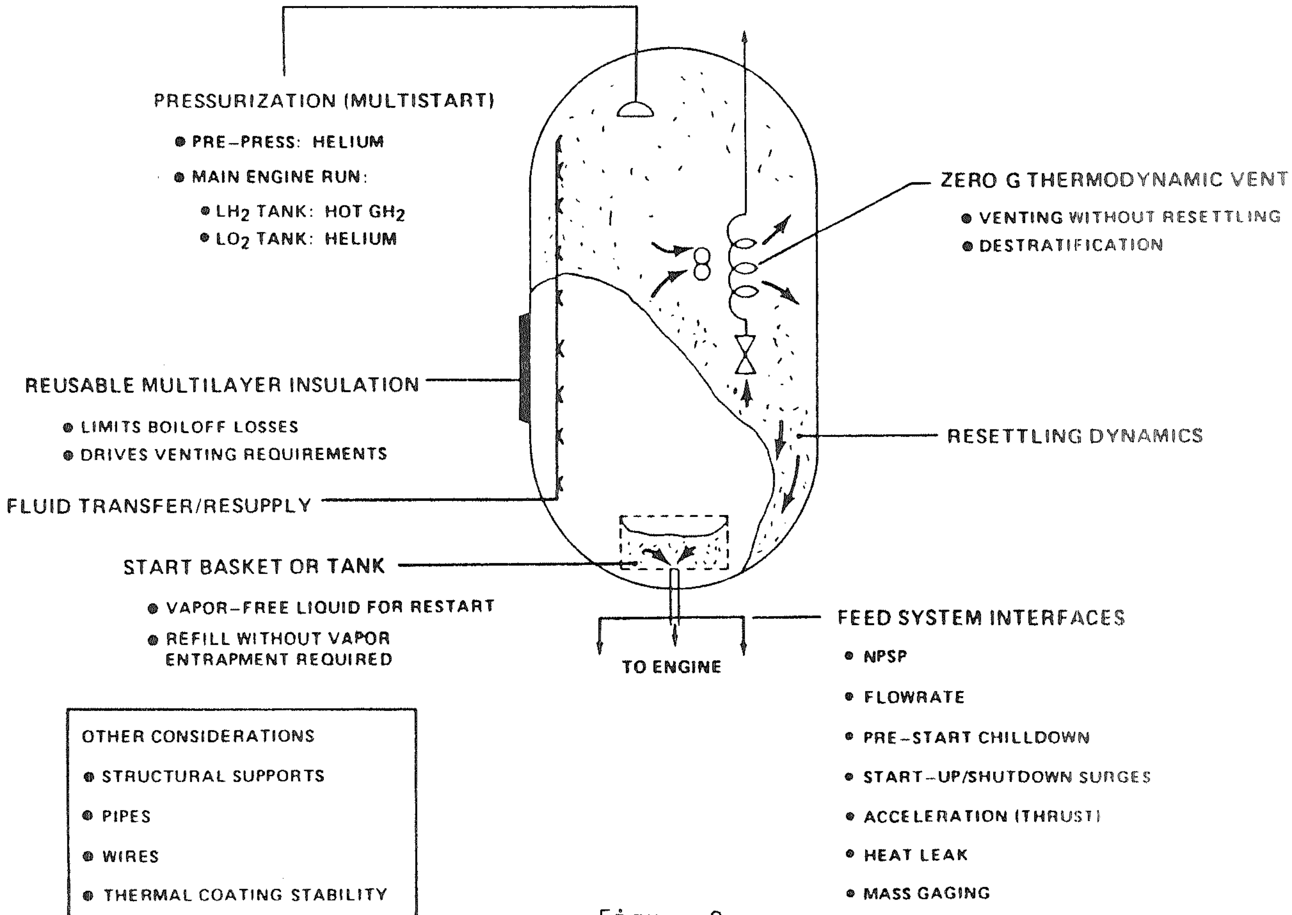

Figure 2

Fluid management of an OTV will require component, subsystem, and system development emphasis. The chart on the opposite page pictorially shows the major areas that must be addressed in the design of a cryogenic OTV. Some of the major issues involved in the design are no-liquid venting, stratification, vapor entrapment in the start basket, engine feed system requirements and reusability. Several items will require orbital testing for verification of their performance (e.g. thermodynamic vent, fluid dynamics, start basket, fluid transfer, etc.). Also, it is important to note that the thermodynamic, fluid mechanic and heat transfer interactions between components and subsystems must be addressed/understood to assure proper system integration. For example, the zero G vent system design is driven by heat leak control/distribution. Similiarly, the start basket liquid retention capability is degraded by increases in feed system heat leak, pressurization gas temperature, and propellant temperature. Engine system re-start/run requirements on propellant conditions significantly affect thermodynamics within the tank and start basket design.

OTV CRYOGENIC MANAGEMENT CONSIDERATIONS

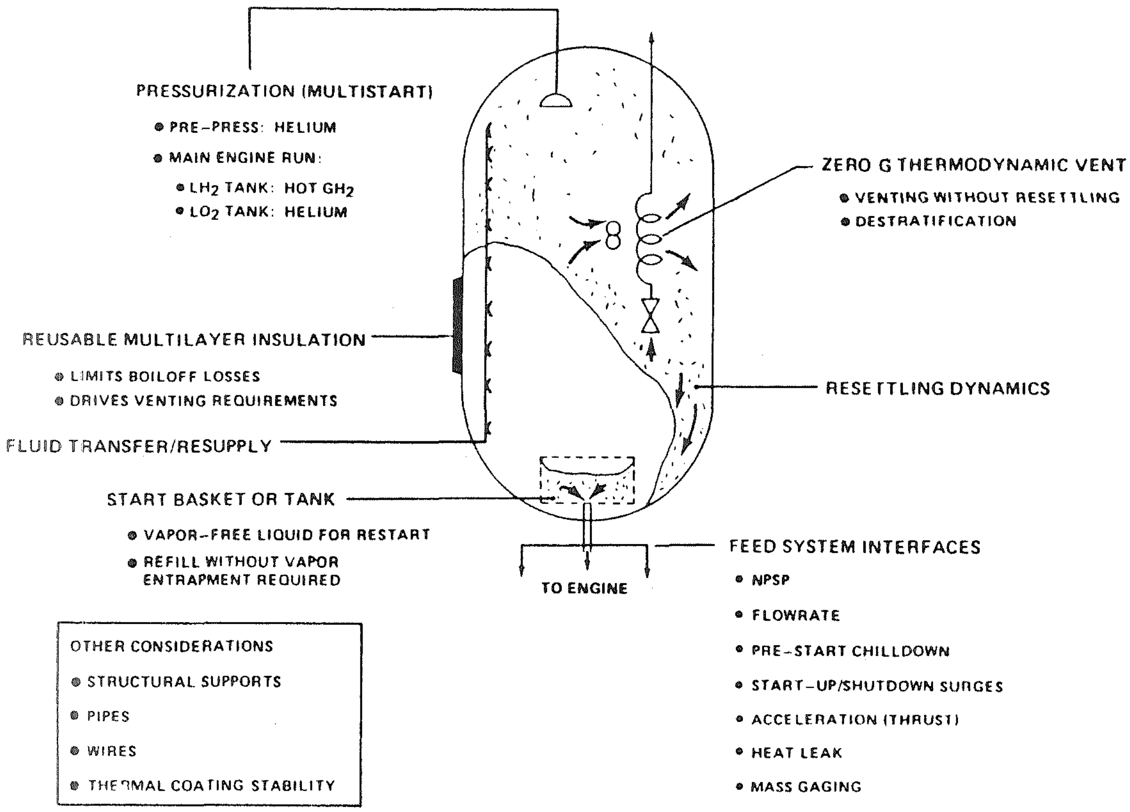

Figure 3

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ORBITAL CRYOGEN TRANSFER CONSIDERATIONS

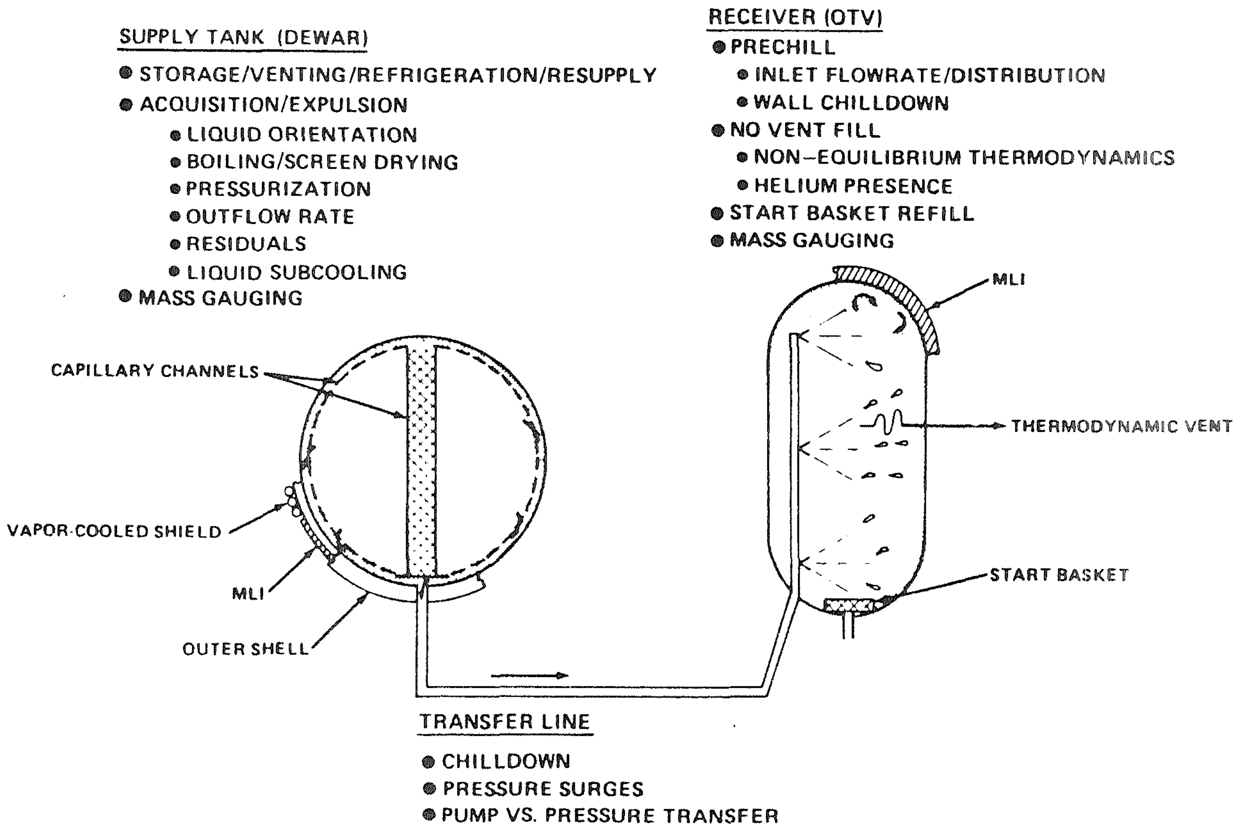

Figure 4

One of the primary fluid management requirements will be the transfer of both LOX and LH_2 in a zero-g environment. Both filling of the OTV tanks and delivery of the propellant to the engine must be considered. The chart on the opposite page describes the major areas that must be investigated. The supply tank could be an orbital storage facility located at the Space Station. The receiver tank would be the OTV LOX and LH_2 tank. Primary issues to be addressed on the OTV are tank prechill, vent vs. no-vent fill, start basket refill and mass gauging. Other areas requiring study are transfer line pressure and temperature transients and pump versus pressure fed fluid transfer.

OTV TANK INSULATION EFFECTS ON VEHICLE PERFORMANCE

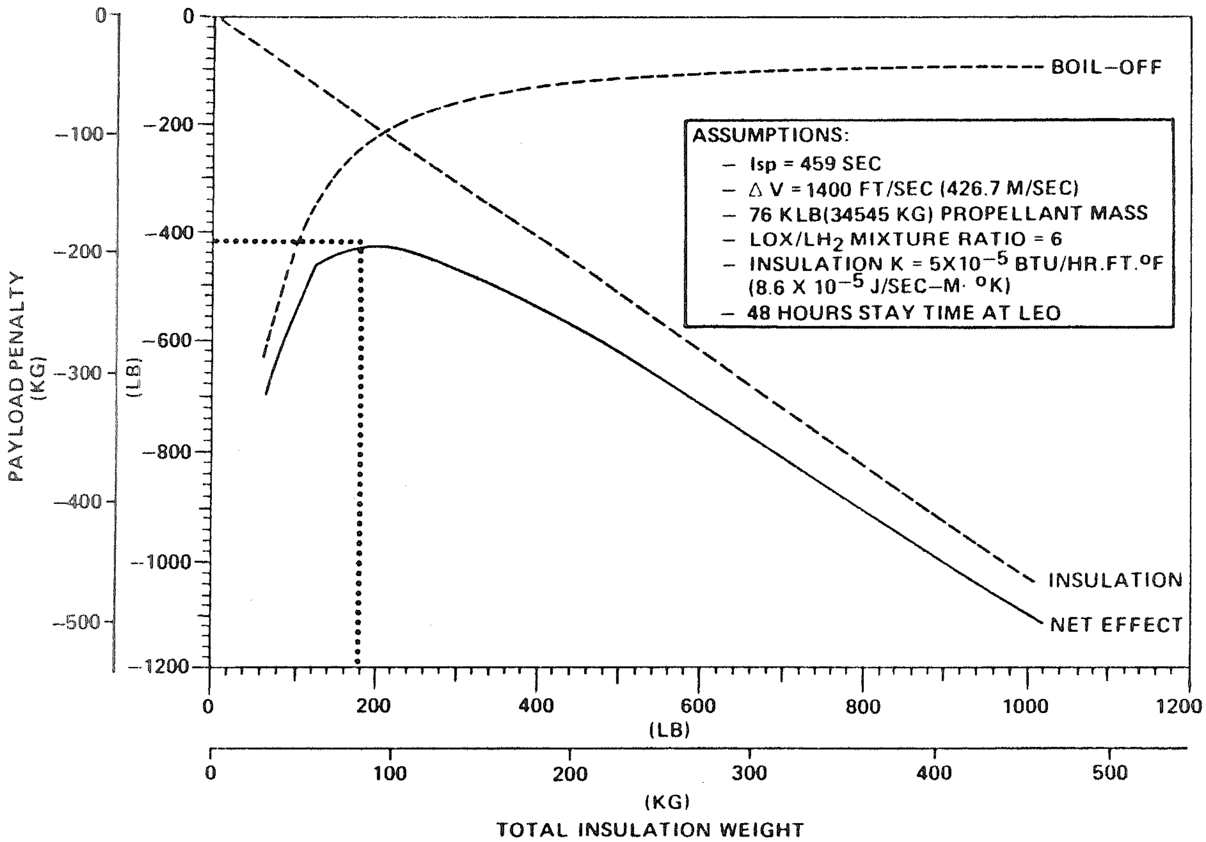

Figure 5

The design of the insulation system for both the hydrogen and oxygen tanks on a space based OTV will be optimized to provide maximum payload delivery capability to GEO. A tradeoff between insulation weight and propellant boiloff provides a characteristic curve such as shown on the opposite page. The design optimization is dependent on how much time after propellant loading will be required at LEO, during transfer from LEO to GEO and at GEO. Since the environment at LEO is generally warmer than at GEO and assuming equal stay times at both LEO and GEO, the LEO environment would dictate the insulation design. Based on the assumptions specified, a total insulation weight of 180 lb would be optimum.

OTV TANK INSULATION REQUIREMENTS

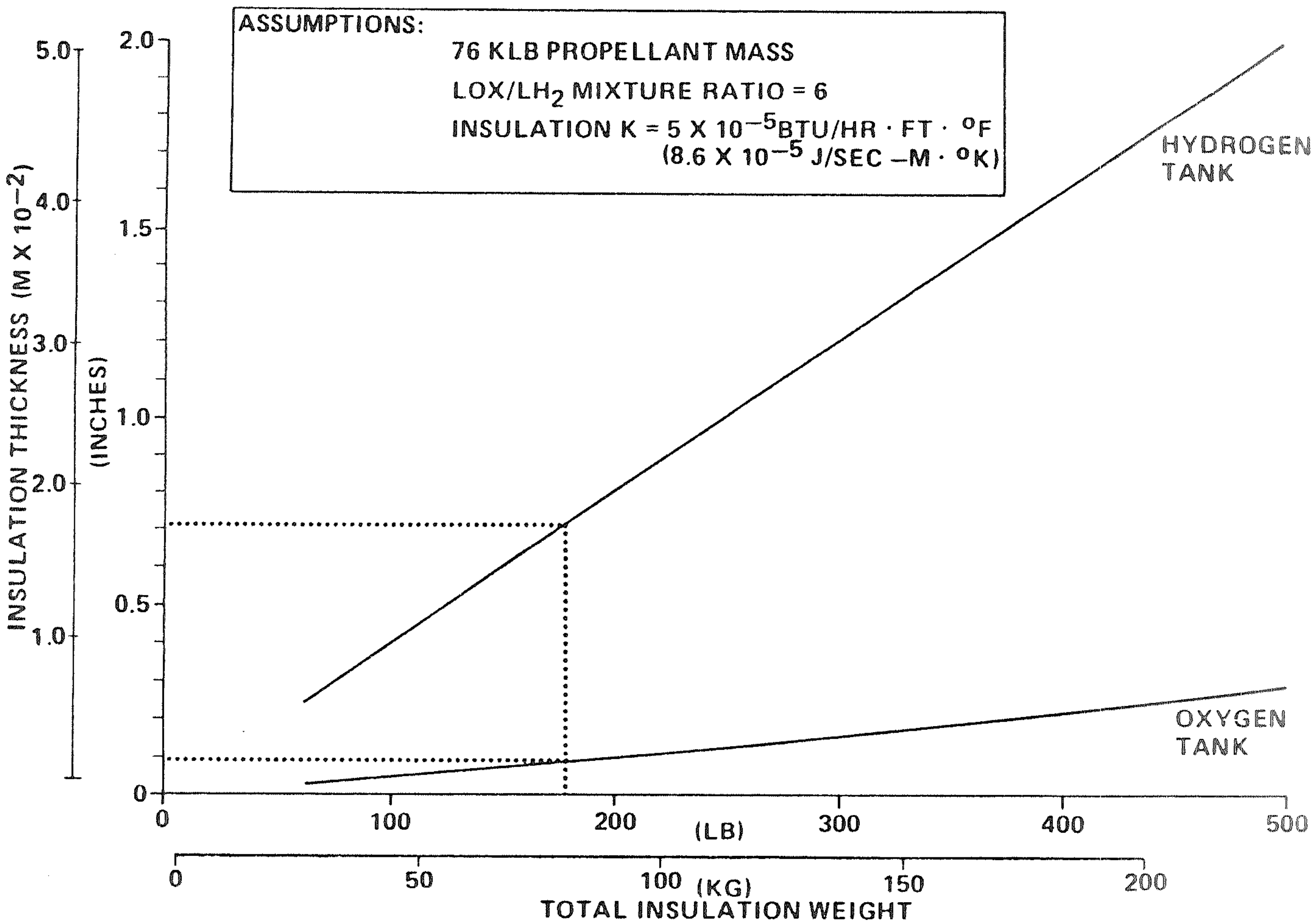

Figure 6

Based on the assumptions made on the previous page, the insulation requirements for both the hydrogen and oxygen tank are shown in the graph on the facing page. As indicated, based on an optimum total insulation weight of 180 lb, the resultant insulation thicknesses for the hydrogen and oxygen tank are approximately 0.7 and 0.1 inches, respectively. The insulation thicknesses on each tank are tailored to maintain the proper propellant mixture ratio.

OTV LH_2 TANK PRESSURES DURING ORBITAL COAST
STRATIFICATION/MIXING EFFECTS

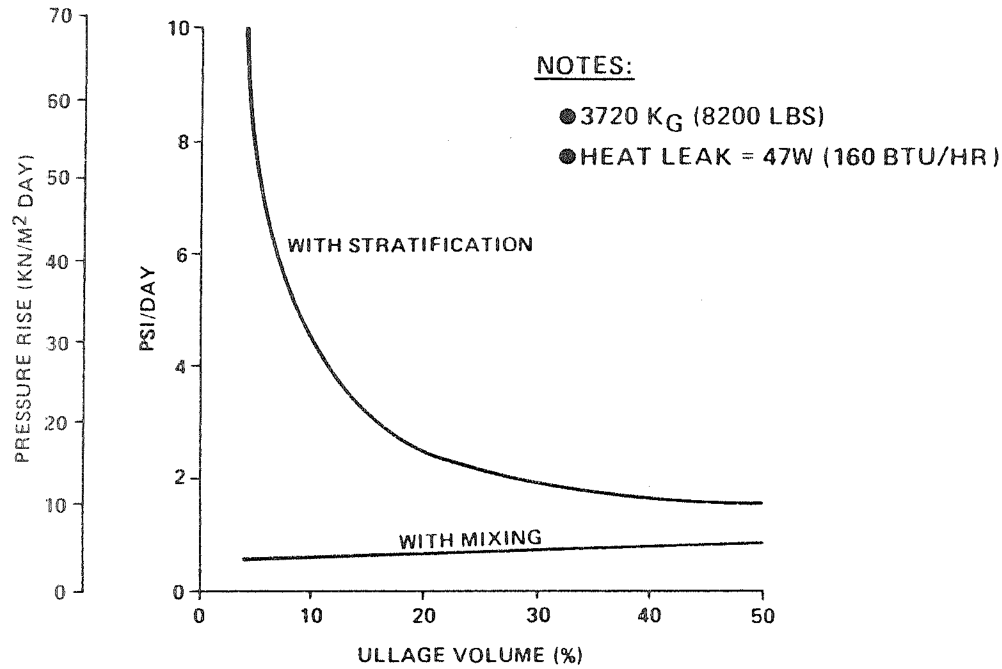

Figure 7

Propellant conditions during orbital coast periods between engine burns are important from several standpoints. For example, tankage heat leak and its distribution within the propellant determines the ullage pressure rise rates and resultant vent rates/cycling. To minimize pressure rise rate and transient thermodynamic uncertainties, the general approach is to assure that tankage sidewall and penetration heat leak is uniformly distributed within the bulk liquid, and that good heat exchange between the ullage and liquid exists. To remove uncertainties associated with passive mixing/destratification in reduced gravity, active mixing techniques are generally employed in OTV concept designs.

Additionally, the energy distribution within the tank can significantly affect other subsystem functions. If a capillary start basket is utilized, localized stratification within and near the basket should be prevented, i.e., localized superheating/boiling can occur. Also, proper feed system thermodynamic conditions must be established for each engine burn.

OTV LO_2 TORUS TANKS
PROPELLANT MANAGEMENT TECHNOLOGY ISSUES

- NO IN–FLIGHT EXPERIENCE WITH REDUCED GRAVITY FLUID/HEAT TRANSFER BEHAVIOR IN TORUS TANKS.
- ACQUISITION DEVICE R&D REQUIRED
  - PROPELLANT SETTLING
  - THERMAL ISOLATION
  - RESIDUALS
  - ORBITAL PERFORMANCE VERIFICATION
- PRESSURIZATION/VENTING
  - MULTIPLE ENGINE RESTARTS/PRESSURIZATION EFFICIENCY
  - ZERO G VENTING
  - STRATIFICATION/DESTRATIFICATION
  - ACQUISITION SYSTEM INTERFACES
- SLOSH
  - PROPELLANT C. G./VEHICLE CONTROL
  - BAFFLES
- INSULATION
  - UNIQUE TANK SHAPE EFFECTS ON PERFORMANCE
  - PURGE

Figure 8

The state-of-technology supporting LO_2 fluid management in torus tanks is weak. Due to its unique geometry, the torus shape introduces a wide range of issues that have not been addressed in past technology efforts. Propellant acquisition, pressurization, venting, stratification/destratification, sloshing, insulation, and heat leak distributions are all areas requiring R&D efforts specifically applicable to torus tanks.

TANK PRE-CHILL PREPARATIONS SUMMARY

- DILUTION OF HELIUM RESIDUALS PRIOR TO REFUELING REQUIRED TO PREVENT:
  - EXCESSIVE PRESSURES AT END OF FILL
  - INACCURATE KNOWLEDGE OF PROPELLANT VAPOR PRESSURES
  - START BASKET HELIUM ENTRAPMENT
  - INACCURATE THERMODYNAMIC MASS GAUGING
- APPROXIMATE DILUTION LEVELS REQUIRED
  - $LH_2 \leq .45$ KG (1 LBS)
  - $LO_2 \leq .09$ KG (.2 LBS)

  } FURTHER DILUTION REQUIRED IF THERMODYNAMIC MASS GAUGING UTILIZED
- PROCEDURAL/TECHNOLOGY CONCERNS
  - DURATION OF VENT/HOLD CYCLES
  - KNOWLEDGE OF HELIUM RESIDUAL MAGNITUDE

Figure 9

The initial phase of orbital transfer is "prechill preparations." If no helium pressurant gases have been used in the tankage to be filled, the prechill preparations would be minimal. However, if helium is present then the tankage must be purged and vented until the helium is reduced to an acceptable level. The "acceptable level" is determined based on end-of-fill pressures/achievement of maximum fill control, capillary screen acquisition system pressure, and thermodynamic mass gaging (if used). The LO_2 system sensitivity to helium is significantly greater than with LH_2. Lack of orbital experience and in-orbit measurement of residual helium magnitudes are the primary concerns in developing a suitable purge approach.

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OTV LH_2 TANK THERMODYNAMICS DURING CHILLDOWN

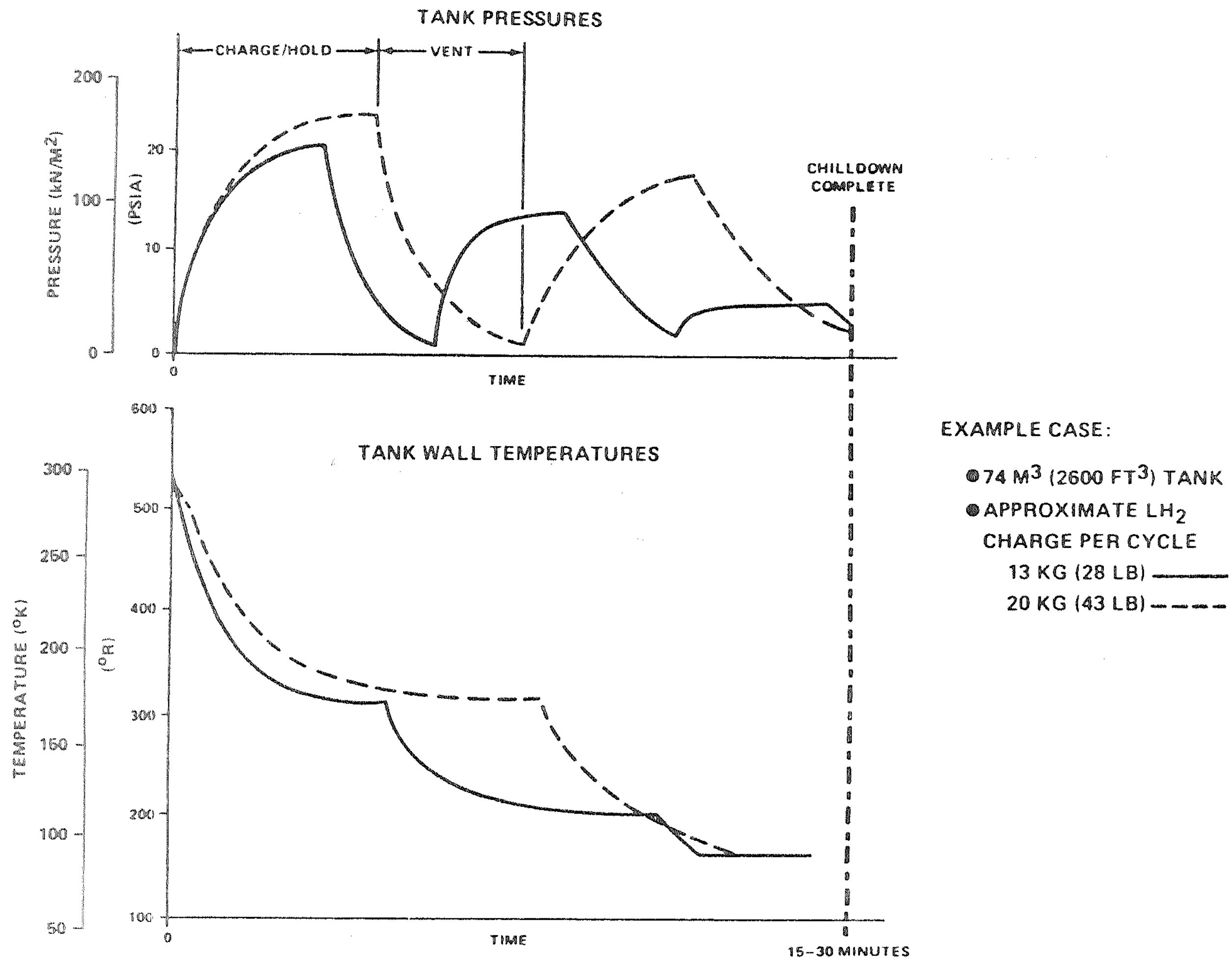


Figure 11

Chilldown is accomplished by introducing propellant into a tank in such a manner that good heat exchange between the high temperature walls and chilldown liquid is assured. Thermodynamic calculations indicate that the amount of propellant required to chill a tank should be relatively small. It is therefore doubtful that chilldown procedure selection will be driven by minimization of chilldown liquid. However, the complicated thermodynamic, boiling heat transfer, and fluid dynamic phenomena involved cannot be analytically modeled with confidence. Hence, issues involving definition of inlet flow distribution/velocity, charge/hold duration and maximum pressure, vent duration, and instrumentation to monitor chilldown progress remain.

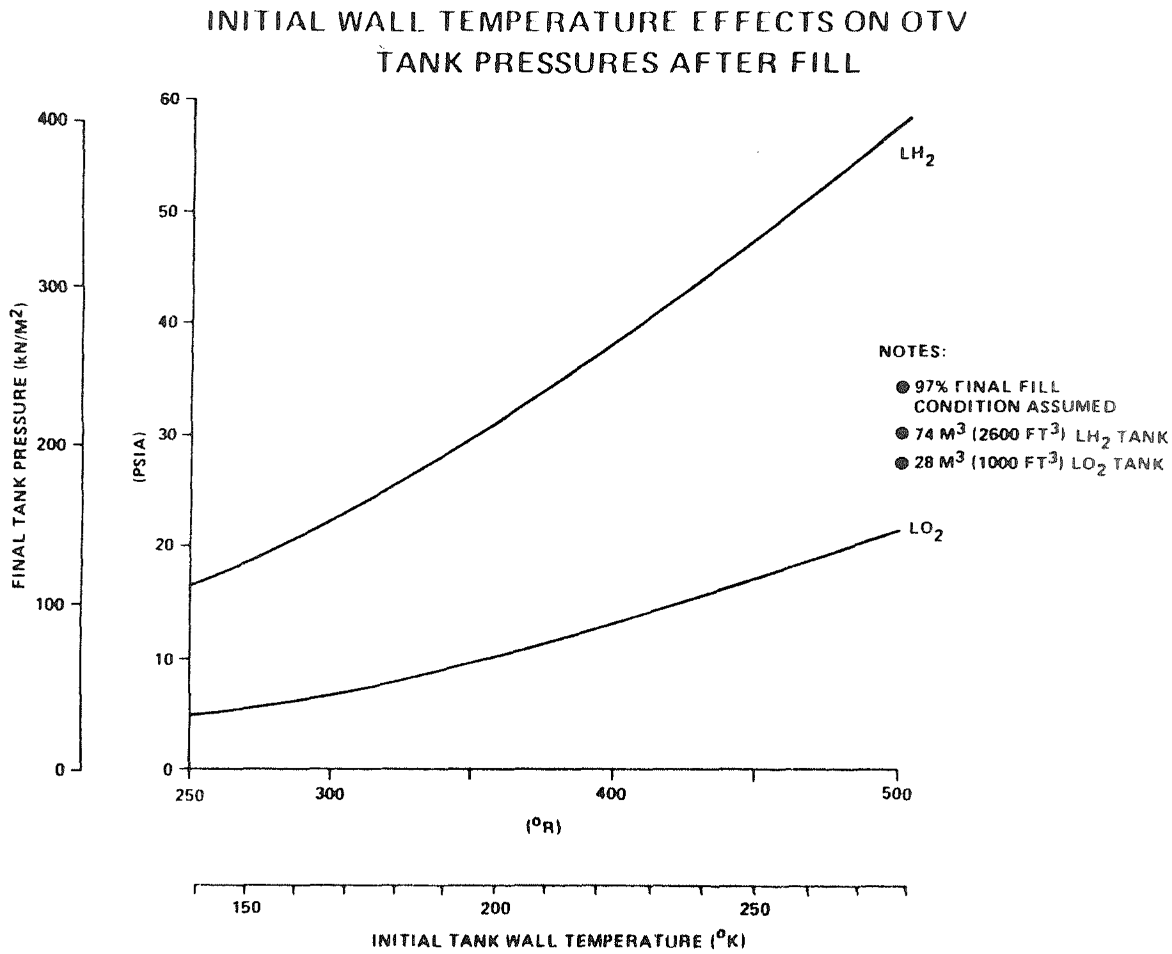


Figure 12

Receiver tank chilldown must be conducted whenever thermal energy stored in the tank walls is sufficient to preclude a nonvented fill operation. For example, with initial wall temperatures of 450^{o}R, the LH_2 and LO_2 tanks final pressures would be 48 psia and 18 psia, respectively; hence, LH_2 chilldown would be required, whereas LO_2 chilldown would be optional. A LH_2 tank wall temperature of less than 250^{o}R probably will be required.

TRANSFER LINE/TANK CHILLDOWN SUMMARY

- REQUIREMENT: REDUCE TRANSFER LINE/TANK WALL TEMPERATURES SUFFICIENTLY TO PREVENT EXCESSIVE LINE PRESSURE/FLOW SURGES AND TO ENABLE A NON-VENTED TANK FILL
- PROCEDURAL/TECHNOLOGY CONCERNS:
  - TANK CHARGE/HOLD/VENT CYCLE DEFINITION
    - SEMI-EMPIRICAL MODELING LACKS EXPERIMENTAL DATA
    - LACK OF HARDWARE EXPERIENCE
    - WALL CHILLDOWN CRITERION: CURRENT RANGE = 95°K TO 200°K (170°R TO 360°R)
    - CHARGE MASS/FLOWRATE SELECTION TBD
  - LACK OF TRANSFER LINE CHILLDOWN EXPERIENCE – PREVENTION OF EXCESSIVE SURGES AND LINE LOADS
  - INSTRUMENTATION TO MONITOR CHILLDOWN PROCESS

Figure 13

Based on the preceding discussions of chilldown issues, optimum operational efficiency and minimum complexity/crew time are apparently the primary goals (as opposed to minimizing propellants used for chilldown). However, definition of charge/hold/vent cycles that will allow achievement of these goals cannot occur until/unless orbital experience and data are acquired.

OTV LH_2 TANK PRESSURES DURING ORBITAL FILL

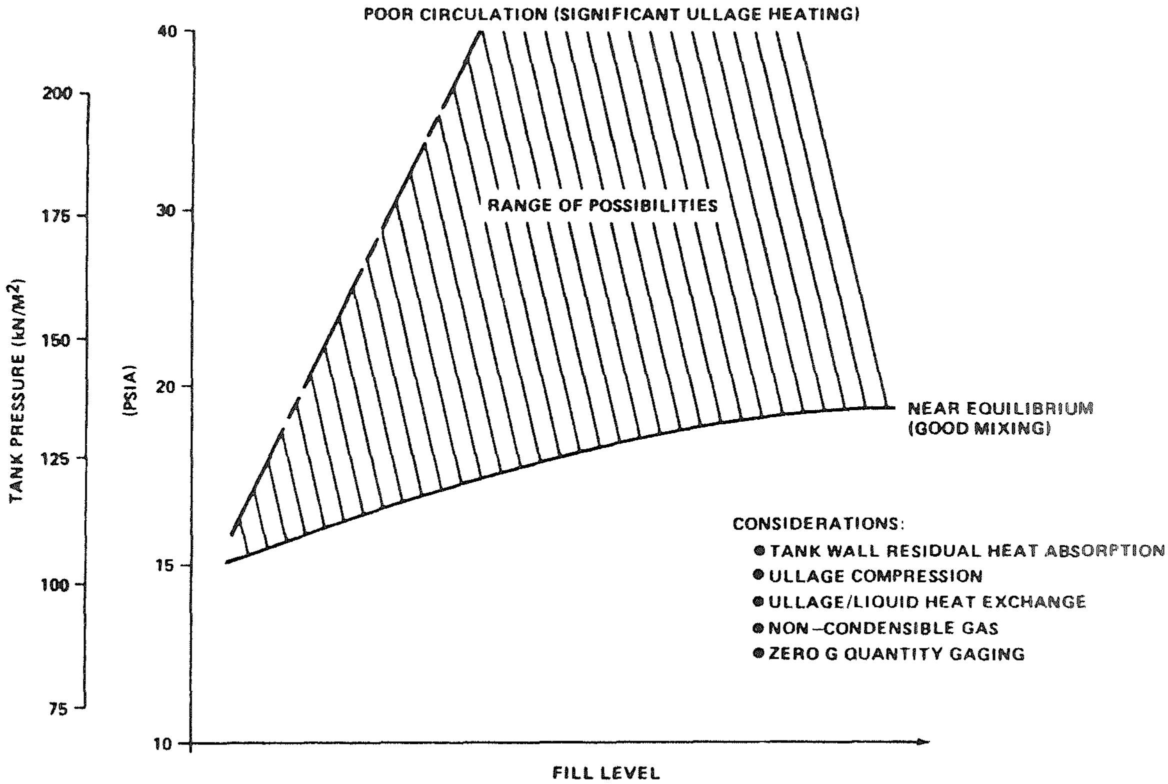


Figure 14

Assuming that the prescribed tank chilldown temperatures have been achieved, then the nonvented fill procedure can be initiated. However, care must be taken to assure that venting is not necessitated by excessive pressure during fill. Good mixing must occur throughout the fill process to prevent excessive heat transfer to the ullage and corresponding pressure increases. Additionally, tank wall residual heat absorption/distribution, ullage compression, noncondensible gases, and the measurement of transferred mass are issues that must be addressed.

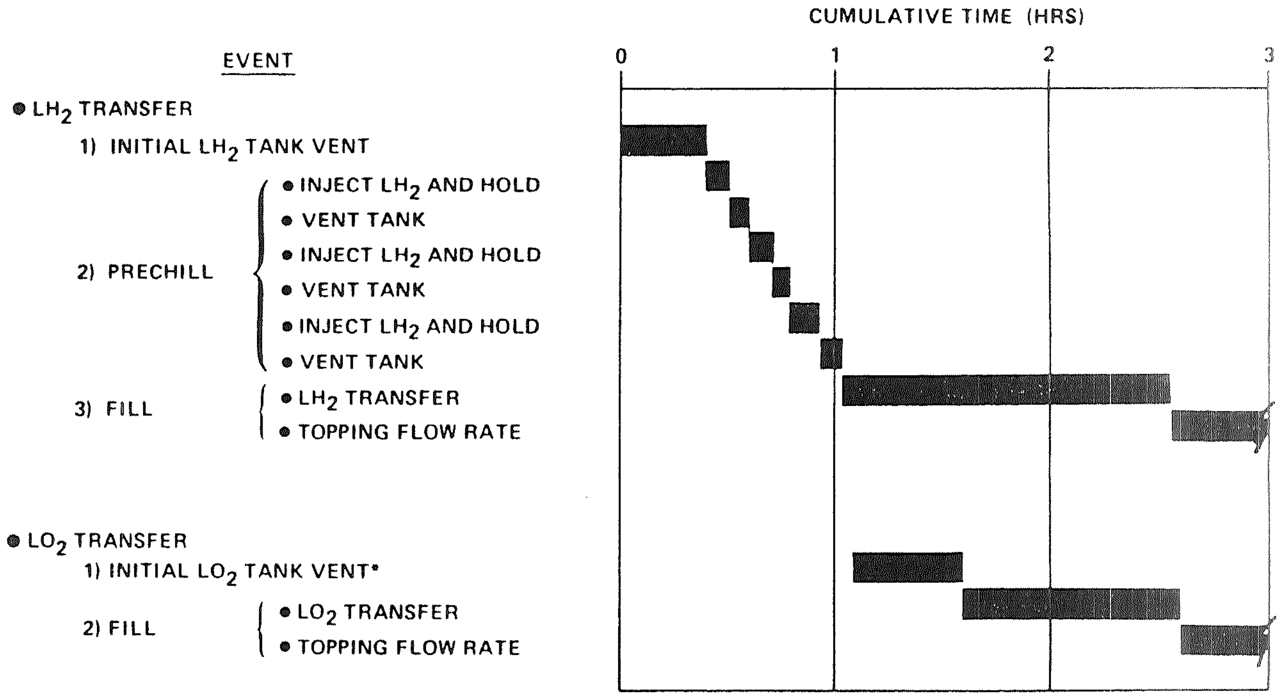
TANK FILL SUMMARY

- REQUIREMENT: LH_2 & LO_2 TANK FILL WITHOUT VENTING
- PROCEDURAL/TECHNOLOGY CONCERNS:
  - ASSURANCE OF ADEQUATE CIRCULATION TO MAINTAIN NEAR–THERMAL EQUILIBRIUM, i.e., LOW PRESSURES
    - GOOD MIXING/HEAT EXCHANGE BETWEEN ULLAGE/LIQUID REQUIRED
    - EXISTING SEMI–EMPIRICAL MODELS LACK EXPERIMENTAL DATA
    - LACK OF IN–FLIGHT HARDWARE EXPERIENCE
    - MECHANICAL MIXER PROBABLY REQUIRED
  - LACK OF ZERO–G MASS GAUGING DEVICE
  - SPECIAL FILL PROVISIONS FOR START BASKET
    - BLEED LINE FOR DIRECT FILL OF BASKET
    - ACTIVE CIRCULATION TO ASSURE ENTRAPPED VAPOR COLLAPSE
  - SUPPLY TANK VAPOR PRESSURE $\leq$ 2.2 kN/M^2 (15 PSIA), NO HELIUM PASSAGE ALLOWABLE
  - PREVENTION OF EXCESSIVE TRANSFER LINE LOADS

Figure 15

Semi-empirical modeling of the fill process is required to define the interacting fluid and thermal phenomena; however, existing models lack experimental verification. Active mixing probably will be required to assure near equilibrium thermodynamic conditions. The lack of a zero G quantity gauge is a significant handicap in achieving a 97% fill condition. Special considerations are involved in interfacing with capillary start baskets to assure that vapor entrapment does not occur during tank fill. Also, supply vessel conditions must be controlled to prevent excessive vapor pressures and the transfer of helium into the OTV.

OTV PROPELLANT TRANSFER TIMELINE



NOTE:

* TWO OR MORE ADDITIONAL VENT CYCLES REQUIRED IF HELIUM PRESENT

Figure 16

Definition of the transfer timeline cannot be accomplished with confidence until orbital experience and data become available. However, the sequence of events can be established with reasonable confidence. Based on current models, the total transfer time is expected to require on the order of 3 hours.

MSFC
CRYOGENIC MANAGEMENT BREADBOARD

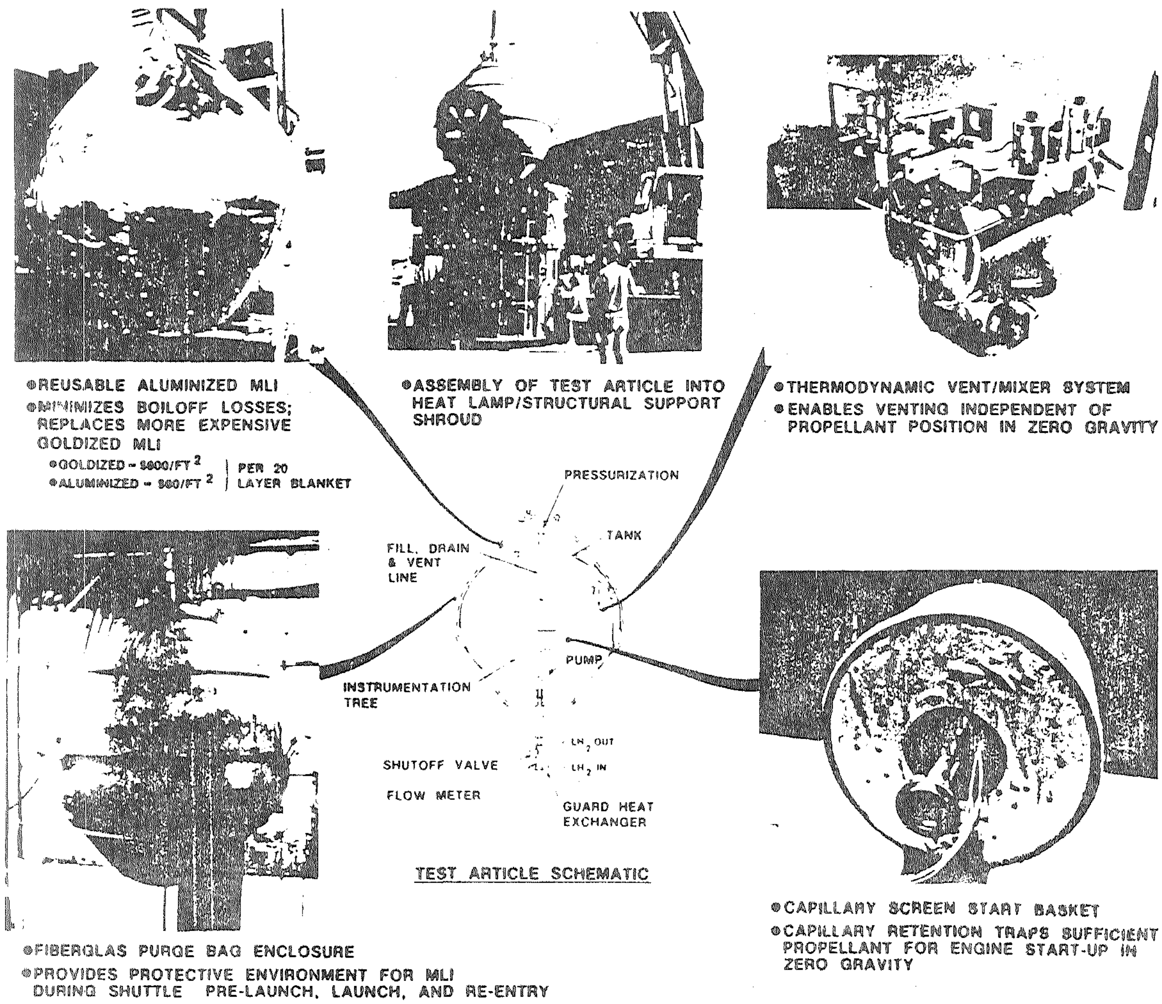


Figure 17

Various degrees of technology development are associated with the types of subsystems that will be required in an OTV cryogen management system, i.e., the technology backgrounds range from substantial to meager. However, these subsystems have never been integrated into a total OTV-type system and required to perform simultaneously. Therefore, a major objective of the cryogenic management breadboard program is to integrate advanced technology items into a system level LH_2 test article, thereby enabling evaluation of thermodynamic, heat transfer, and fluid mechanic interactions/controls/instrumentation within the limits of normal gravity testing. The breadboard data will be evaluated to determine normal gravity performance and to more specifically identify technology gaps/concerns that must ultimately be assessed with orbital experimentation, i.e., breadboard testing of this type is a prerequisite to the eventual experimental verification of OTV-type systems in orbit. Additionally, the system level experience will minimize the development risk of orbital cryogenic management experiments/flight systems in general.

The test article tank is an 88-inch diameter oblate spheroid with a 175 ft^3 volume. The test article contains all the basic elements of an earth-based OTV LH_2 system, i.e., a reusable multilayer insulation/purge bag system, zero gravity thermodynamic vent/mixer, GH_e/GH_2 pressurization, capillary start basket, and a pump/feedline system. The multilayer insulation, organically coated aluminized Kapton, was developed to replace the more expensive reusable goldized Kapton insulations. This breadboard installation represents the first system level demonstration of the aluminized insulation for cryogenic applications.

Final preparations are in progress at MSFC for the breadboard testing. Initial LH_2 loading is scheduled for the first week of April 1984. Various test phases will be conducted intermittently through October 1984.

CRYOGENIC FLUID MANAGEMENT FACILITY
OTV TECHNOLOGY

MISSION 1

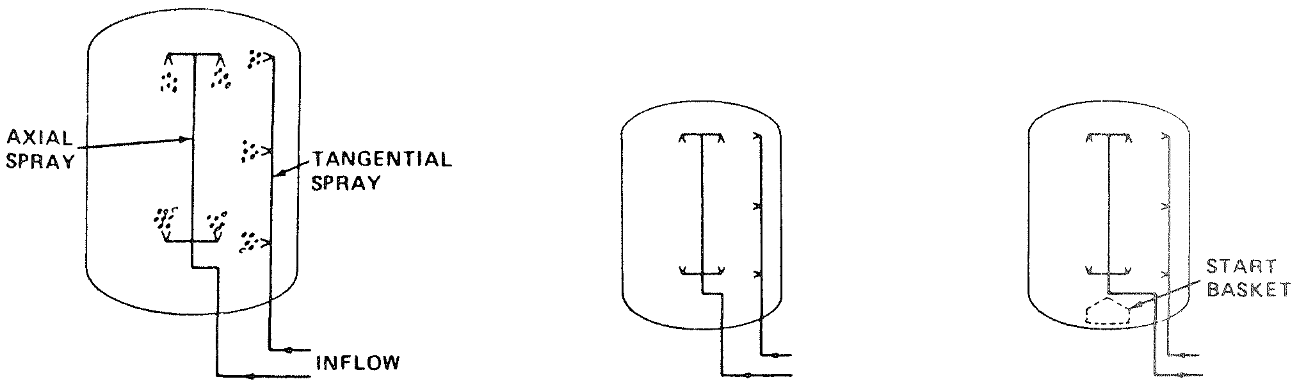
.28 SCALE
(48 IN. DIAMETER)

MISSION 2

.18 SCALE
(31 IN. DIAMETER)

MISSION 3

.18 SCALE
(31 IN. DIAMETER)



GOALS

- SCALABLE TANK CHILLDOWN DATA
- PURGED MLI (30 LAYER)

DELTA GOALS

- NO VENT FILL
- LH_2 SETTLING/OUTFLOW
- He PRESSURIZATION
- ON-WALL TVS
- STRATIFICATION

DELTA GOALS

- START BASKET INTEGRATION & PERFORMANCE
- TVS MIXER
- FOAM/MLI COMBINATION

Figure 18

The Cryogenic Fluid Management Facility (CFMF) is expected to provide significant technology inputs to OTV development. The initial mission will utilize a .28 scale OTV LH_2 receiver vessel. Although the CFMF supply tank can fill the receiver to only about the 30% level, the primary goal of obtaining chilldown data can be achieved. An OTV representative purged multilayer insulation (MLI) will be installed on the receiver. The second mission will utilize a .18 scale vessel that can accommodate a complete fill procedure. Additional data include LH_2 settling/outflow, helium pressurization, and performance of a thermodynamic vent system (TVS) with a wall mounted heat exchanger. The third mission will also utilize a .18 scale vessel. Chilldown / fill data will again be acquired to assess repeatability of the mission 2 results. An OTV type start basket will be utilized to assess thermodynamic and fluid mechanic interface effects on start basket performance, i.e., feed system heat leak, TVS operation, and tank pressurization. The TVS may include an active mixing system. The tank insulation will consist of a foam/MLI combination.

EXAMPLE CFMF DATA FOR OTV LH_2 TANK CHILLDOWN DURING TRANSFER

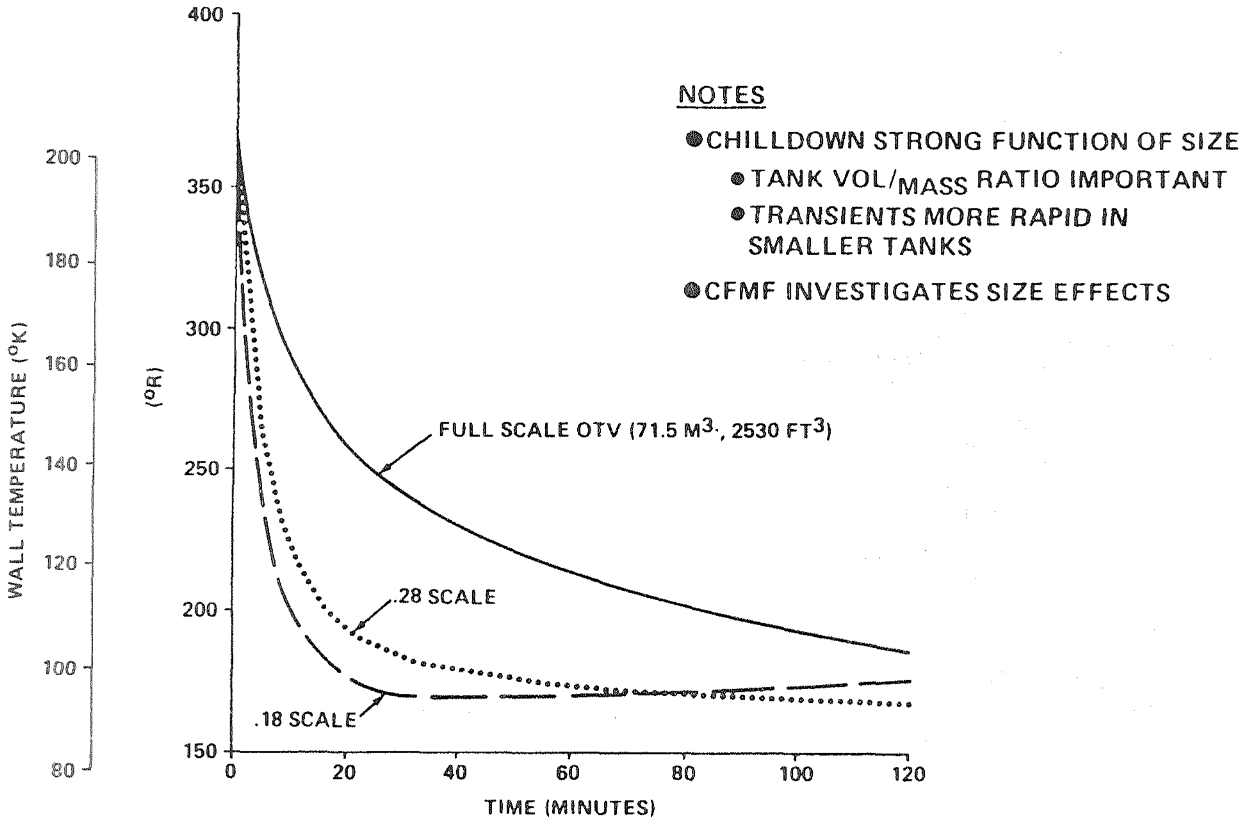


Figure 19

The relative chilldown responses of the CFMF .18, .28, and full scale OTV receivers can be illustrated using currently available analytical modeling. The smaller a vessel, the more responsive it is to heat leak and the nonequilibrium thermodynamics. This is basically because the tank volume relative to energy stored in the walls and structure becomes less with decreasing tank size. Therefore, there exists the concern that the rapid response of small vessel thermodynamics/fluid dynamics will differ significantly from the actual transients in prototype vessels. However, the CFMF design has incorporated the largest scale OTV vessel achievable (.28 scale) within the constraints of schedule and cost. Additionally, LH_2 transfer behavior in the .28 and .18 scale vessels can be compared, thereby providing valuable scaling effects data.

SPACE STATION TECHNOLOGY REQUIREMENTS
(SSTSC FLUID MANAGEMENT WORKING GROUP)

- CRYOGENIC FLUID RESUPPLY*
- NON-CRYOGENIC FLUID RESUPPLY*
- ZERO-LEAKAGE FLUID COUPLINGS
- FLUID LEAK DETECTION INSTRUMENTATION
- REUSABLE EARTH TO ORBIT CRYOGEN TRANSPORT
- FLUID QUANTITY GAUGING INSTRUMENTATION
- LONG TERM ORBITAL CRYOGEN STORAGE*
- CONTROL, INSTRUMENTATION & DIAGNOSTICS
- OPERATIONS (MANNED VS. AUTONOMOUS)
- FLUID SYSTEM STUDY

*MANDATORY FLIGHT TESTS

Figure 20

The Space Station Technology Steering Committee met in Williamsburg, Virginia in March, 1983, to discuss technology requirements and priorities. The Fluid Management Working Group recommended that technology be pursued in ten areas. The chart on the facing page lists these recommendations in order of their priority for Space Station application. Out of the ten areas, three were considered to require mandatory flight tests. These three items were considered to be enabling technologies.

SPACE STATION ADVANCED DEVELOPMENT

- ADVANCED DEVELOPMENT TEST BED
  - COMPONENT DEVELOPMENT TESTING
  - LOX/LH_2 SYSTEM LEVEL TESTING
  - FLUID LEAK PREVENTION/DETECTION
- PROPOSED SHUTTLE FLIGHT EXPERIMENTS
  - LONG TERM CRYOGENIC STORAGE FACILITY
  - REFRIGERATION/RELIQUEFACTION
  - REMOTE CONTROLLED OR AUTOMATED PROPELLANT SERVICING
- PROPOSED SPACE STATION TECHNOLOGY DEMONSTRATION MISSION (TDM)
  - PROPELLANT TRANSFER, STORAGE & RELIQUEFACTION
  - LONG TERM SYSTEM PERFORMANCE DEGRADATION

Figure 21

Based on the anticipated need for a cryogenic OTV at the Space Station, several proposals have been made to define the advanced development work that will be required to support such a goal. A combination of ground testing, shuttle flight testing and Space Station technology demonstration missions (TDM's) are evolving as the primary activity for achieving this goal. The opposite page provides a brief summary of the major proposed advanced development activity in the fluid management area.

N85-17008 D19

VEHICLE/ENGINE INTEGRATION

L. P. Cooper, Moderator, NASA Lewis Research Center
Tim Vinopal, Boeing Aerospace
D. Florence, General Electric
Roy W. Michel, Aerojet TechSystems Company
J. R. Brown, Pratt & Whitney Aircraft
R. P. Bergeron and V. A. Weldon, Rockwell International Corporation

Tim Vinopal
Boeing Aerospace

As a systems integrator, Boeing recognizes that the main propulsion system has a profound affect on vehicle development cost and schedule. Significant engine weight growth or unplanned changes in performance capability have important implications in vehicle design and mission capture.

Agreement is needed on man-rating requirements as these will greatly affect vehicle/engine integration. As a minimum, elimination of all single point failures requires re-examination of aeroassist concepts which require large, retractable engine nozzles. Placing the nozzles behind the heat shield moves large deployed payloads in front of the shield-making P/L return impossible. The manned transfer cab is small enough to either fit behind the unmanned aeroassist device or have a kittable heat shield, depending on aeroassist concept. Preliminary reliability analyses indicate that a single engine is unable to meet manned mission reliability goals. An increase in the number of engines corresponds to a decrease in performance and an increase in maintenance requirements. Performance analyses currently show a 5000 to 7000 lb engine thrust range as optimum; however, the cost analysis is expected to move the optimum to a level above 7000 lbs. The high cost of space based maintenance may have the dual effect of increasing the thrust level, and derating the engine components to reduce the amount of engine maintenance required.

VEHICLE/ENGINE INTEGRATION ISSUES

Q. FROM A PRIME CONTRACTOR STANDPOINT WHAT ARE KEY VEHICLE/ENGINE INTEGRATION ISSUES?

- IMPACT OF ENGINE INTEGRATION ON CONFIGURATION DEVELOPMENT (DEVELOPMENT TIME, DDT&E, AND PERFORMANCE).
- IMPACT ON MAN RATING AND MISSION RELIABILITY (OPERATING COST).

Q. HOW DOES SPACE BASING IMPACT VEHICLE/ENGINE INTEGRATION?

- MODULARIZE ENGINE INSTALLATION AND/OR CRITICAL COMPONENTS TO ALLOW EFFECTIVE ON ORBIT SERVICING.
- HIGH SERVICING COSTS (~ $20,000/HR) MAKES DERATING ENGINE FOR LONG SERVICE - FREE LIFE ATTRACTIVE.

Q. HOW DOES AEROASSIST IMPACT VEHICLE/ENGINE INTEGRATION?

- ENGINE NOZZLE RETRACTION REQUIREMENT INTRODUCES SINGLE-POINT FAILURE MODES.
- LARGE, HIGH EXPANSION RATIO ENGINES DIFFICULT TO SHIELD FROM FREE STREAM FLOW.

Figure 1

EFFECT OF ENGINE RELIABILITY ON SYSTEM RELIABILITY

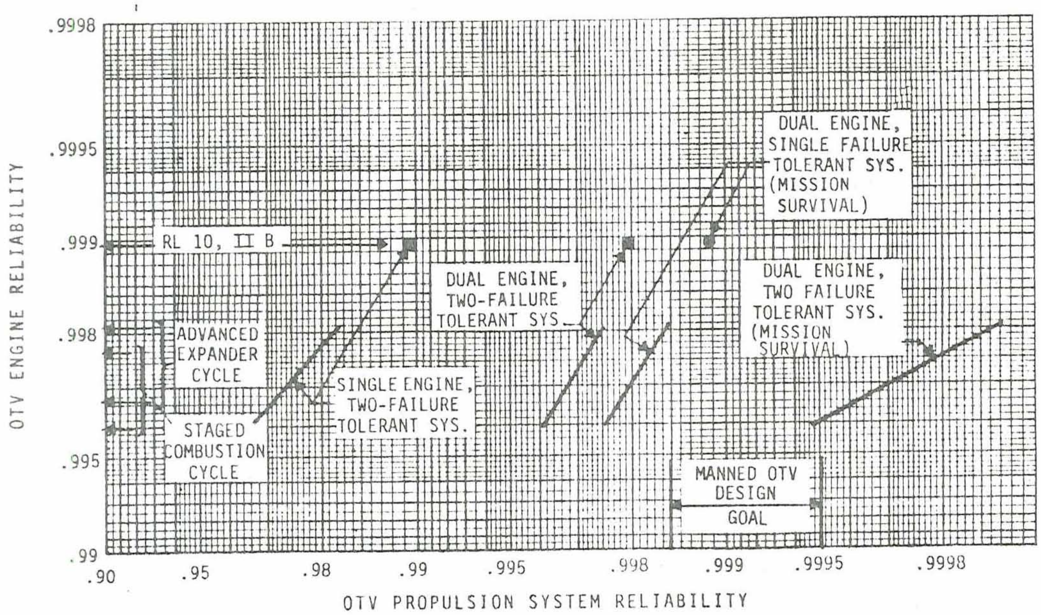


Figure 2

Fresh Look Lifting Brake Designed for Space Assembly

- COMBINE BEST FEATURES OF LIFTING BRAKE & AMOTV TO INCREASE L/D AND REDUCE SCAR WEIGHT
- SPACE ASSEMBLED PREFABRICATED COMPOSITE PANELS
- RIGID OR FABRIC REUSABLE TPS
- LARGE PLANFORM AREA REDUCES TEMPERATURES
- NO IMPINGEMENT PROBLEM

- STS COMPATIBLE OTV MOUNTED USING SHUTTLE FIXTURES
- OTV CAN BE EITHER GROUND BASED OR SPACE BASED
- NO NOZZLE RETRACTION REQUIRED
- GROSS TRIM ACCOMPLISHED BY SLIDING OTV ON RAILS
- CONTROL WITH AERODYNAMIC SURFACES & RCS

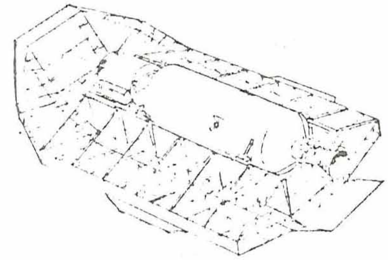


Figure 3

PRELIMINARY RESULTS OF VEHICLE/ENGINE INTEGRATION STUDY

- DUAL ENGINE INSTALLATIONS FAVORED
  - RELIABILITY VS. OPERATING COSTS
- ENGINES IN 7000 LB + TO 15000 LB + SIZE RANGE CURRENTLY FAVORED
  - FUNCTION OF HIGH EXPANSION RATIO NOZZLE EFFECTIVENESS
  - ENGINE DERATING WILL INFLUENCE SIZING TRADE
  - AVERAGE MISSION COST WILL BE SELECTION CRITERIA
- NON RETRACTABLE NOZZLES FAVORED FOR MAN RATING & MISSION RELIABILITY
  - PUTS PAYLOAD IN FRONT OF HEAT SHIELD
  - TREAT MANNED MISSIONS AS UNIQUE AND INTEGRATE HEAT SHIELD WITH MANNED MODULE

Figure 4

D. Florence
General Electric

Numerous propulsion subsystem related parameters impact the AOTV configuration development and ultimate performance. However, the major first order parameters appearing to have the greatest impact are engine specific impulse, Isp, propellant mixture ratio, MR, and packaging volume and length required for the engines and associated plumbing, Figure 1. It was demonstrated in Reference 1 that 1) improved specific impulse (443 to 480 sec) provides the largest benefit for both single stage and two stage AOTV's, 2) for the single stage AOTV, the combined effects of a smaller hydrogen tank due to increased mixture ratio and the shorter vehicle due to use of multiple small engines, provides a benefit nearly as large as the increased Isp.

For ground based AOTV's, the payload weight delivery or round trip capability, is highly dependent on the AOTV dry weight. Other major parameters effecting the payload magnitude include the engine Isp, low earth orbit payload capability of the launch vehicle, and AOTV L/D. For the GEO delivery mission, the vehicle L/D has a minor impact on payload delivery, for the round trip GEO mission, L/D is more important and for polar delivery, even more important, Reference 1. A single stage 38ft GEO delivery vehicle with propellant tanks sized for a mixture ratio of 7 and a single engine was described in Reference 1. Except for the advanced engine (Isp = 477 sec, MR = 7), this vehicle utilized state-of-the-art technology. Significant subsystem weight reductions are possible by incorporating advances projected due to state-of-the-art advances, Reference 1. The improved payload delivery of these lighter vehicles is illustrated in Figure 2, and compared to previous AMOOS results, Reference 2.

Configuration variations of the 38 ft GEO delivery vehicle identified in Reference 1, were explored for a Six Hour Polar Mission to determine effect on payload weight/length, Figure 3. Here, it is noted that incorporating an aft conical frustum angle of 1^{0} results in increased payload length. Lessor frustum angles are expected to produce even longer payloads, however, the axial center of gravity requirements become less attractive and more body flap (heavier) must be added to trim the vehicle at the desired angle of attack. The longer payload lengths are produced by the larger propellant mixture ratios. Additional payload length is obtained by blunting the nose, however, the loss of L/D reduces the payload weight delivery capability. In this evaluation, the AOTV structure and thermal protection subsystem weights were scaled as the vehicle length and surface are changed. Hence, we conclude that for increased allowable payload lengths in a ground based system, lower L/D is as important as higher MR in this range of mid L/D AOTV's.

References

1. Florence, D. & G. Fischer, "Moderate Lift to Drag Aero Assist", presented at the NASA Lewis Research Center Conference on Orbit Transfer Vehicle Propulsion Issues, April 3 & 4, 1984.

2. Program Development, NASA Marshall Space Flight Center, "Orbit Transfer Systems with Emphasis on Shuttle Applications - 1986-1991", NASA TMX-73394, 1977.

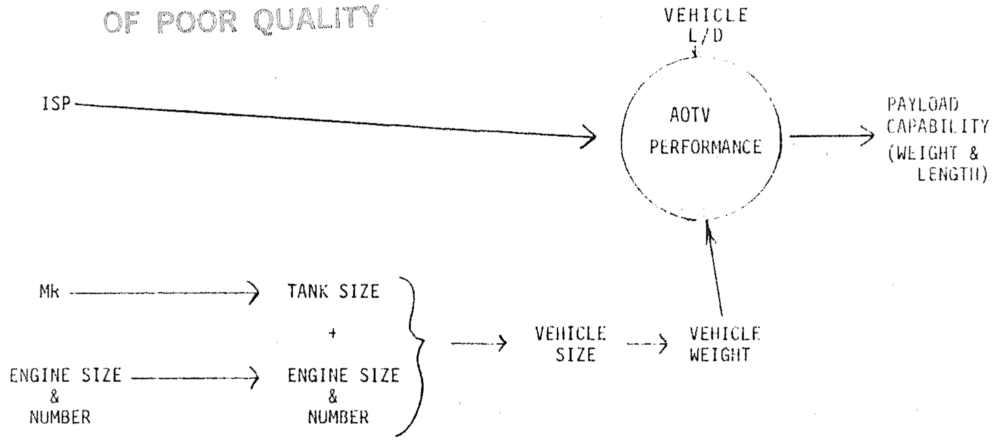


Figure 1. - Propulsion subsystem parameters with first-order impact on AOTV performance.

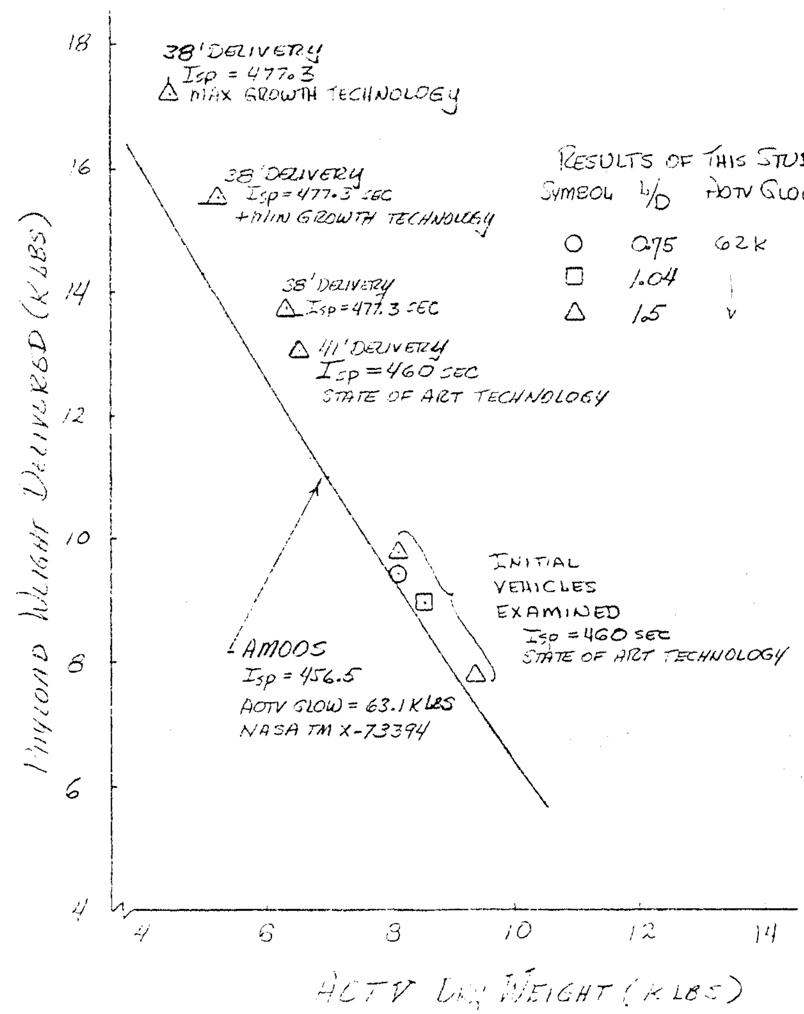


Figure 2. - Mid L/D AOTV GEO delivery capability for single-stage vehicles.

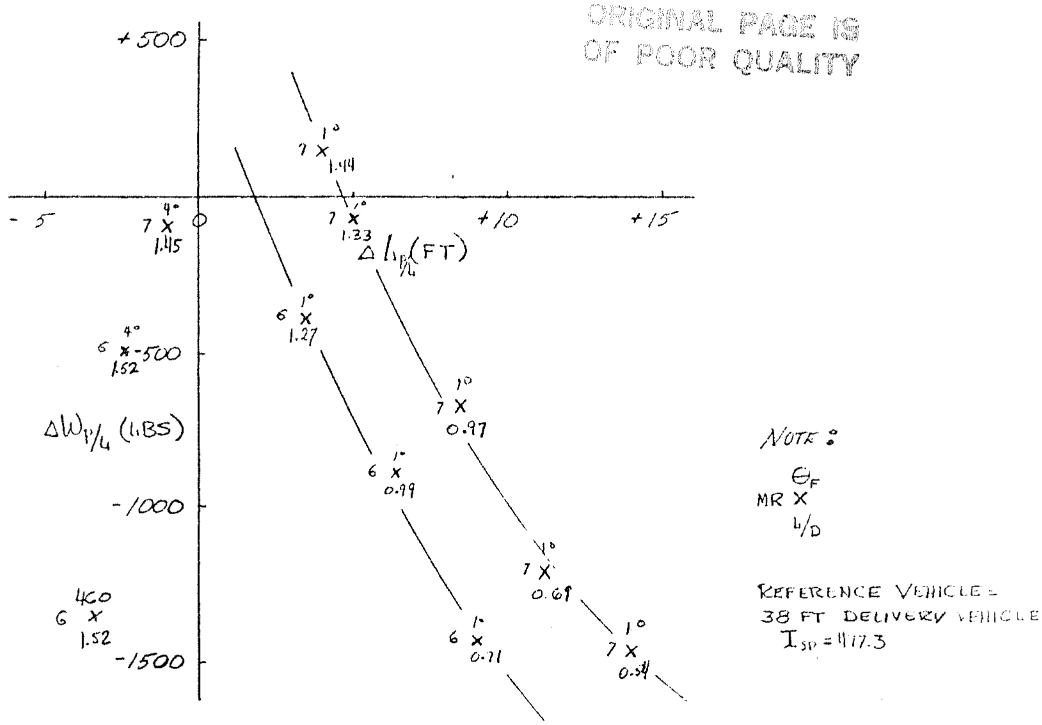


Figure 3. - Performance variations of a single-stage AOTV for polar delivery.

Roy W. Michel
Aerojet TechSystems Company

The Aerojet position is that the right approach to advanced OTV propulsion is with small multiple engines. In contrast to the other engine contractors, Aerojet has selected a nominal design thrust of 3000 lbF.

The small, multiple engine approach has several advantages, notably that crew safety and mission success are assured because of engine-out capability and that highest performance in a given length is obtained with small engines. Length is important both for earth-based OTVs and aeromaneuvering OTVs, and higher performance means greater payload capability.

Of several options for manned OTV reliability, only one provides the necessary reliability and is practical: redundant engines. Other options are far more costly or depend on back-up modes that simply do not exist.

The 3000 lbF thrust engine develops about 4 lbF sec/lbM higher performance than the 15,000 lbF engines within a given length, by virtue of higher area ratio. For the larger engine to achieve the same performance requires an additional three to four feet of length and two or three extendable nozzle segments. In an aeromaneuvering vehicle these extendable segments must also retract during passage through the atmosphere and thus constitute single point failure modes.

With multiple 3000 lbF thrust engines the whole mission model can be performed, efficiently, by a single propulsion system. Large space structures (LSS) are acceleration-limited and have a thrust requirement of 500 to 2500 lbF, which is met by one or two engines throttled. Many payloads are in the 3000 lbM class, which also requires one or two engines. High energy payloads and manned aeromaneuvering vehicles require 10,000 to 12,000 lbF thrust, obtained by a four engine configuration.

Aerojet's approach to space-based maintenance is to design the engine to be a space-replaceable unit, which is most plausible for small engines. If an engine component needs repair, the whole engine would be removed and returned to earth; repairs would be made by skilled technicians and the engine retested to assure its operation and performance.

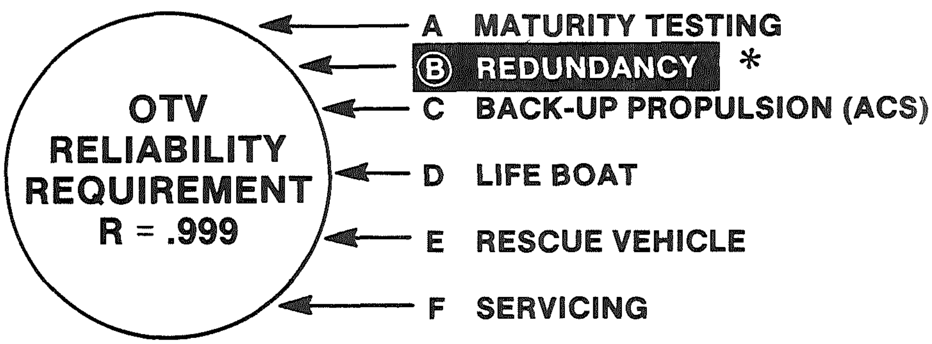
The several advantages of the small, multiple engine approach to OTV propulsion have a life cycle cost benefit on the order of $1 Billion. Altogether, the advantages and potential cost savings prove that the right approach to advanced OTV propulsion is with small, multiple engines.

ADVANTAGES OF SMALL, MULTIPLE ENGINES

- CREW SAFETY AND MISSION SUCCESS ASSURED
- HIGHEST PERFORMANCE FOR GIVEN LENGTH
- MORE PAYLOAD CAPABILITY
- GREATER MISSION FLEXIBILITY
- REAL SPACE-BASED MAINTENANCE
- SAVES $1 BILLION

Figure 1

OPTIONS FOR OTV RELIABILITY



* ASSURES MISSION SUCCESS

Figure 2

HIGHEST PERFORMANCE FOR GIVEN LENGTH

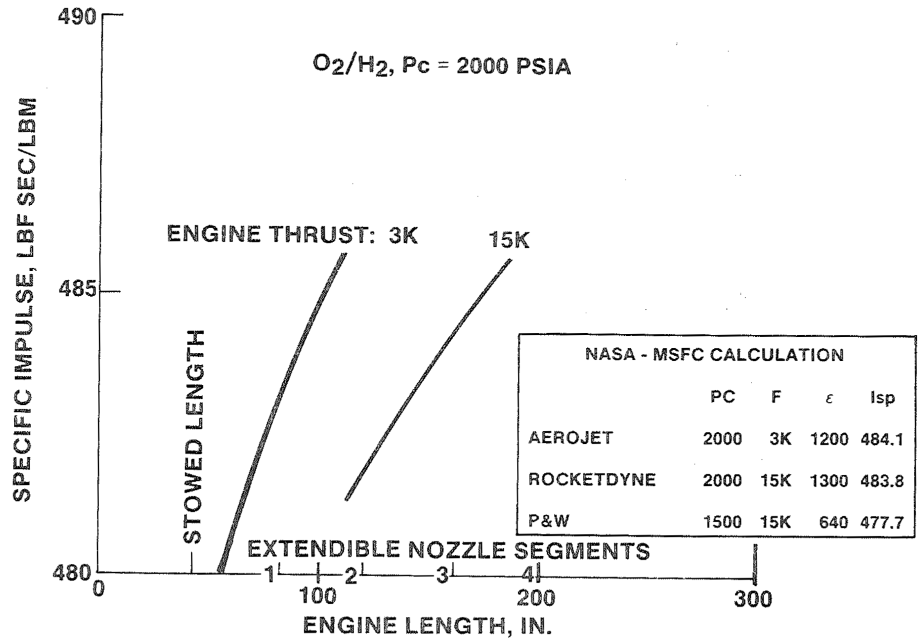


Figure 3

GREATER MISSION FLEXIBILITY

PAYLOAD	THRUST, lbF	# ENGINES
LSS	500-2500	1 OR 2 (THROTTLED)
3000 lbM	3000	1 OR 2
12,000 lbM	11,000	4
MANNED	9,500 (AEROMANEUVERING)	4

Figure 4

SMALL ENGINE MEANS REAL SPACE-BASED MAINTENANCE

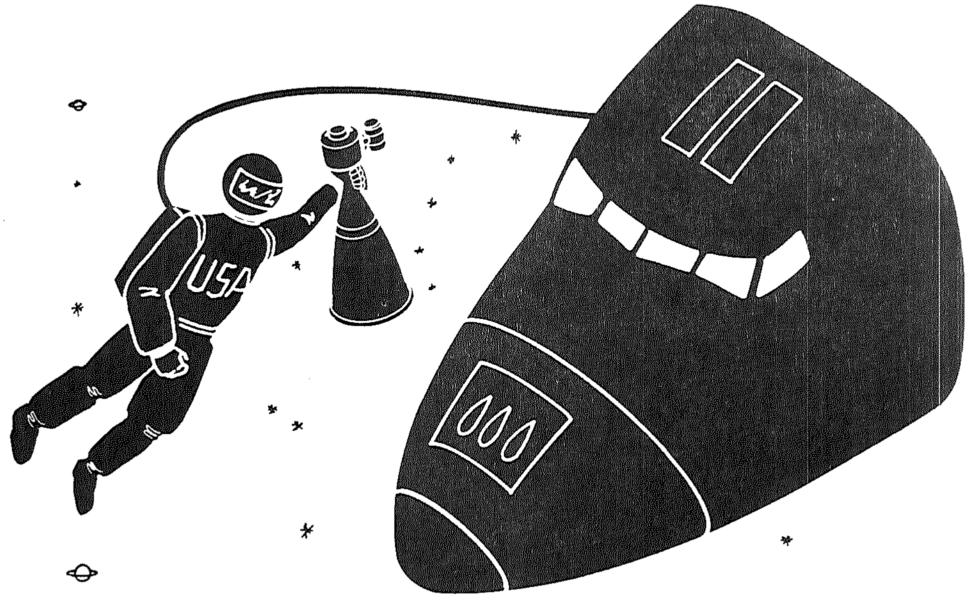


Figure 5

SMALL ENGINE APPROACH SAVES $1 BILLION

	VALUE
• RELIABILITY	$100 M
• WEIGHT	-40 M
• ENVELOPE/PACKAGING	70 M
• PERFORMANCE	400 M
• MISSION FLEXIBILITY	500 M
	$1000 M

Figure 6

J. R. Brown
Pratt & Whitney Aircraft

Pratt & Whitney Aircraft believes that several significant issues exist in the engine/vehicle integration area. These issues fall into the general categories of:

- scenario validity
- geometry constraints
- throttle levels
- reliability
- servicing

We believe that one engine cannot be optimized to cover all possible perturbations of these issues. Rather, the issues must be resolved in a coordinated effort between the engine and systems contractor and only then can the engine configuration be selected.

IS CURRENT SCENARIO VALID?

- Space based OTV
- Propellant depot
- Manned GEO missions
- Substantial LEO-GEO traffic
- Low thrust deployment missions
- Only one type OTV
- New driver mission (e.g., lunar lander)

Figure 1

WHAT ARE ENGINE GEOMETRY CONSTRAINTS?

- Available length
- Available diameter
- Vehicle total thrust required
- Number of engines

Figure 2

WHAT THROTTLE LEVELS ARE REQUIRED?

- Steps (1%, 10%, 100%)
- Continuous (1%, 3% to 100%)
- Mixed (1%, 3% to 10%, 100%)

What response rate(s) are required?

Figure 3

WHAT ARE ENGINE REQUIREMENTS DURING AEROASSIST MANEUVER?

- Nonfiring
- Firing
  - Thrust level(s), response
  - Extendable nozzle position
- Engine environment
  - Thermal
  - Flow field

Figure 4

HOW DOES ENGINE INFLUENCE VEHICLE RELIABILTY?

- Number of engines
- Mission logic (number of failures to abort)
- Back-up dependency
  - Main engine
  - ACS
  - "Life boat"
  - Rescue mission

Figure 5

WHAT ARE VEHICLE SERVICING REQUIREMENTS/CAPABILITIES?

- Routine maintenance (after every mission)
- Periodic maintenance (after every 10 missions)
- Unscheduled maintenance
- Back-up mission logic
  - One spare vehicle
  - One spare + components
  - Two spare vehicles
  - Etc.
- Dependency on diagnostic systems

Figure 6

ENGINE/VEHICLE INTEGRATION SUMMARY

The engine contractors need to know:

1. How does vehicle limit engine geometry?
2. What is engine required to do?
   - Primary mode
   - Aeroassist mode
3. What propulsion system reliability is needed?
4. What engine servicing capability is available?

Figure 7

D20 N85-17009

ORBITAL TRANSFER VEHICLE PROPULSION ISSUES

VEHICLE/ENGINE INTEGRATION

R. P. Bergeron and V. A. Weldon
Rockwell International Corporation

The development of a reusable and space-basable orbital transfer vehicle (OTV) necessitates an integral approach toward structural and propulsion subsystems design. Key drivers include gimbal/actuator location, feed line gimbal provisions, and accessability for orbital maintenance. Recent studies have considered the use of toroidal tank configurations with the engine(s) located within the central cavity of the toroid. The primary objective of that approach is to achieve minimum stage length. Dependent upon engine size and number, that concept introduces unique vehicle/engine integration requirements that necessitate special design considerations. Of particular concern is vehicle center-of-gravity (CG) location when the propellant tanks are more than 75% expended. A single engine installation will necessitate moving the engine further aft and/or relocation of the engine gimbal point to accommodate vehicle control requirements. Penalties associated with gimbal point relocation without increasing stage length or modifying typical advanced engine concepts, as well as a method for minimizing such penalties, are presented for a single engine toroidal tank OTV configuration. Alternative integrated vehicle structure/engine concepts are also presented for multi-engine configurations. Features of these potential concepts are presented which indicate the need for substantial additional study of feedline gimbal alternatives before firmly establishing advanced engine design.

INTRODUCTION

The issue of vehicle/engine integration is addressed in three areas; interfaces (physical and functional), installation requirements, and reliability apportionment (i.e., number of engines required to assure mission completion). Typical elements of each area are presented below.

- INTERFACES
  - -THRUST STRUCTURE GIMBAL ATTACH
  - -PRESSURANTS
  - -ACTUATOR(S)
  - -PUMP INLET(S)
  - -PURGE REQUIREMENTS
  - -ELECTRICAL/AVIONICS

- INSTALLATION
  - -ACCESSABILITY
  - -STIFFNESS
  - -INLET CONTOUR CONTROL (UPSTREAM)
  - -GIMBAL/ACTUATOR LOCATION
  - -FEED LINE(S) GIMBAL PROVISIONS

- EXTENDABLE NOZZLE COMPATIBILITY
- AERO-ASSIST KIT COMPATIBILITY

- RELIABILITY APPORTIONMENT
  - FAILURE MODES(S)
  - ENGINE-OUT CAPABILITY

The necessary vehicle/engine interfaces are defined by overall mission, system, and performance requirements. Although some interface requirements are subject to trade study analyses (i.e., autogenous vs helium pressurization, thrust vector control (TVC) vs Reaction Control System (RCS), once the interfaces are defined, their characteristics are established and it remains for the designer to provide an installation that will satisfy other program objectives (i.e., simplicity, accessability, cost, etc.). An efficient overall configuration can be achieved if only an integrated approach toward vehicle structure/engine design is implemented. The objective of this paper is to provide an example of the significant need for such an integrated design approach. To accomplish this, typical OTV concepts, which have been suggested in prior studies, are used to illustrate the potential problem areas that must be addressed prior to advanced engine definition.

DISCUSSION

A typical OTV concept which has received considerable attention in recent years utilizes a conventional propellant storage tank for LH_2, but an advanced toroidal tank design for LO_2 storage. A single engine is installed in the cavity of the toroidal tank in order to minimize stage length and Space Transportation System (STS) launch costs and/or maximize payload length. When operating in an expendable mode, with payload attached forward, this concept is viable. However, when operating in a reusable mode with stage return after payload deployment, the vehicle C. G. moves aft of the engine gimbal point (assuming a conventional engine design with front end gimbal). A potential solution is to move the engine further aft, but this defeats the original objective of shortest stage length. An alternate method, Figure 1 (using a Rocketdyne early RS-44 engine version as an example), is to add a throat gimbal kit which provides a "pseudo" gimbal axis about the engine throat. In this configuration the thrust loads are still transmitted through the power head and thrust structure into a bearing plate on the vehicle. Some redesign of the engine to attach the throat gimbal links is of course required. Another alternative is to redesign the engine for integral throat gimbal and thrust load transfer (i.e., similar to the Apollo Service Module Engine). This change would also necessitate relocation of the feed line interface to the throat gimbal ring. A comparison of the suggested engine modifications are presented below.

REDESIGN FOR THROAT GIMBAL & THRUST LOAD TRANSFER	REDESIGN FOR THROAT GIMBAL WITHOUT THRUST LOAD TRANSFER
• CONVENTIONAL GIMBAL STRUCTURE • MAJOR ENGINE REDESIGN • MINOR WEIGHT IMPACT • FEED LINE SYSTEM-MINOR INCREASE IN WEIGHT & COMPLEXITY • LIMITED POWER HEAD ACCESSABILITY • EXTENDABLE NOZZLE IMPACT	• COMPLEX GIMBAL STRUCTURE • MODERATE ENGINE REDESIGN • MAJOR WEIGHT IMPACT • FEED LINE SYSTEM-COMPLEX & HEAVY WITH INCREASED HEAT LEAK • LIMITED POWER HEAD ACCESSABILITY • EXTENDABLE NOZZLE IMPACT

To resolve such issues, further study of gimbal/feed line alternatives are recommended prior to establishing advanced engine configuration requirements.

The reusable and space-basable OTV is planned to evolve into a man-rated system. In order to achieve this objective, the issue of engine reliability and redundancy requirements must be addressed. The single engine reliability will dictate the number of engines required to satisfy overall mission probability of success, Figure 2. In order to meet manned mission requirements, the reliability apportionment for the propulsion system is in the order of 0.999. As indicated in Figure 2, a two engine configuration (with on engine capable of accomplishing the mission) is equivalent to a three engine configuration (with one engine capable of accomplishing the mission) and superior to a three engine configuration (with two engines required to accomplish the mission). A two engine OTV concept was therefore selected to evaluate vehicle/engine integration issues.

When using multiple engines of RS-44 size, the engines can no longer be installed within the toroidal tank cavity, Figure 3. In this configuration a key integration issue is propellant feed line gimbal requirements. In order to integrate the currently suggested RS-44 engine into a vehicle, feed line gimbal must be accomplished upstream of the pump inlets located on the power head. This would result in complex line routings within the toroidal cavity thus providing limited access for assembly and/or on-orbit maintenance. In addition, the greater articulation of relatively long line lengths (especially in an engine-out condition) would probably limit to two the number of engines that could be installed without significant increase in stage length. A preferred concept may be that which has been employed on the STS orbiter for the SSME; feed line gimbal downstream of the pump inlet. This installation could be lighter and simpler, and provide better access for assembly and on-orbit check out and maintenance. Using this approach, the feed line gimbal system would be included in the advanced engine design. This concept could also be beneficial in that changes in propellant flow characteristics with feed line contour changes during gimbal can be evaluated during engine analyses, design and testing. Propellant feed line gimbal for conventional tank design concepts, Figure 4, have also been evaluated and similar results obtained. Again, this is mainly due to the longer line lengths required and greater feed line articulation needed to satisfy the engine-out condition.

CONCLUDING REMARKS

The objective of this paper is not to recommend the potential changes in advanced engine design discussed above, but rather to emphasize the need for an integrated approach toward vehicle/engine design. This integrated approach becomes even more necessary when aero-assist concepts are considered, especially for those concepts that rely on engine firing during aero-assist maneuver.

An on-orbit checkout and maintenance philosophy must also be established to provide effective guidelines for engine design and self-monitor requirements. With the exception of oils and greases, the aviation industry trend is toward no scheduled maintenance between major overhauls. A similar objective might be considered for the reusable, space-basable OTV.

Engine redundancy, thrust level, throttling, etc. requirements remain as open issues. If properly executed by the selected contractors, the currently planned NASA MSFC OTV Concepts Definition and Systems Analysis Study should provide answers to these and most other vehicle/engine integration issues. In the interim, it appears prudent to maintain as much flexibility as possible in defining an advanced cryogenic engine configuration.

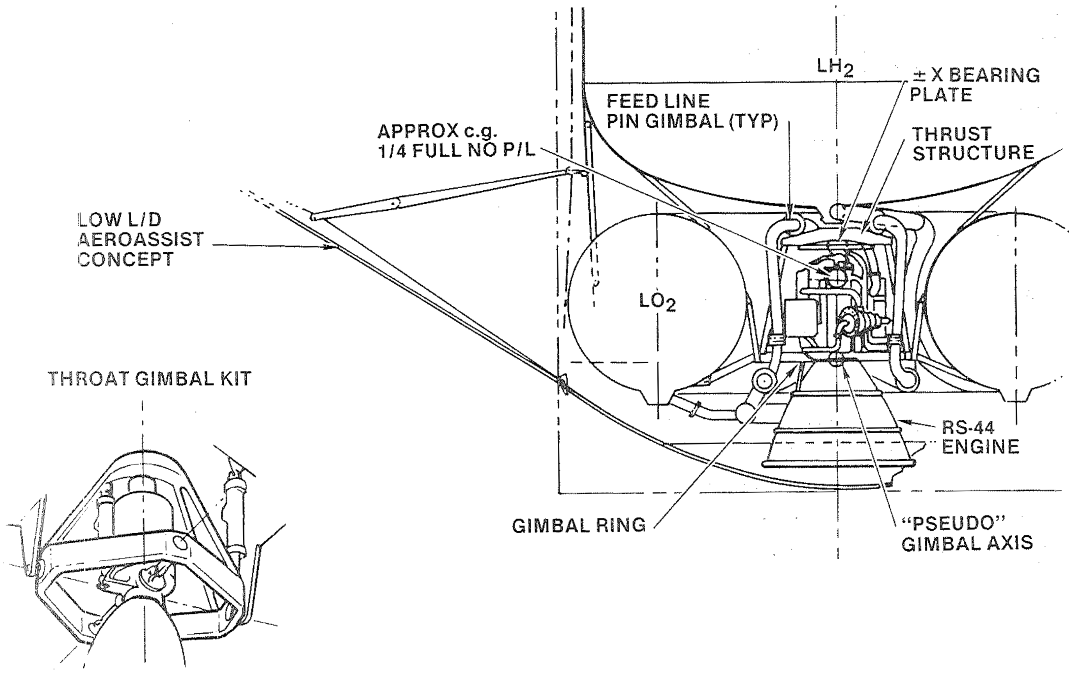


Figure 1. Single Engine Installation for Toroidal Tank Configuration

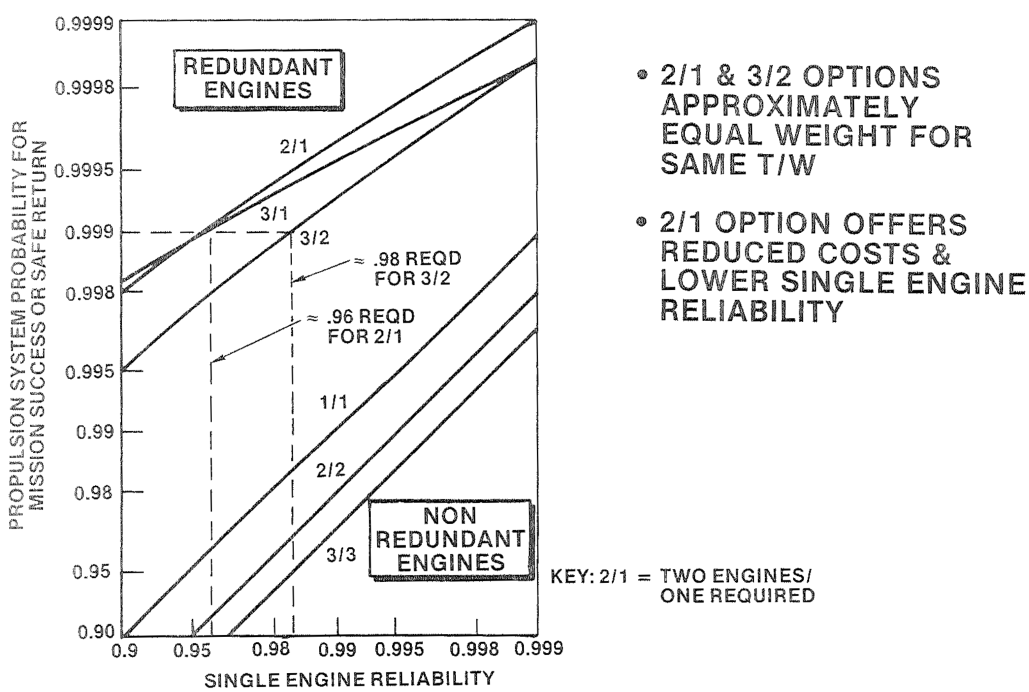


Figure 2. Engine Redundancy Requirements

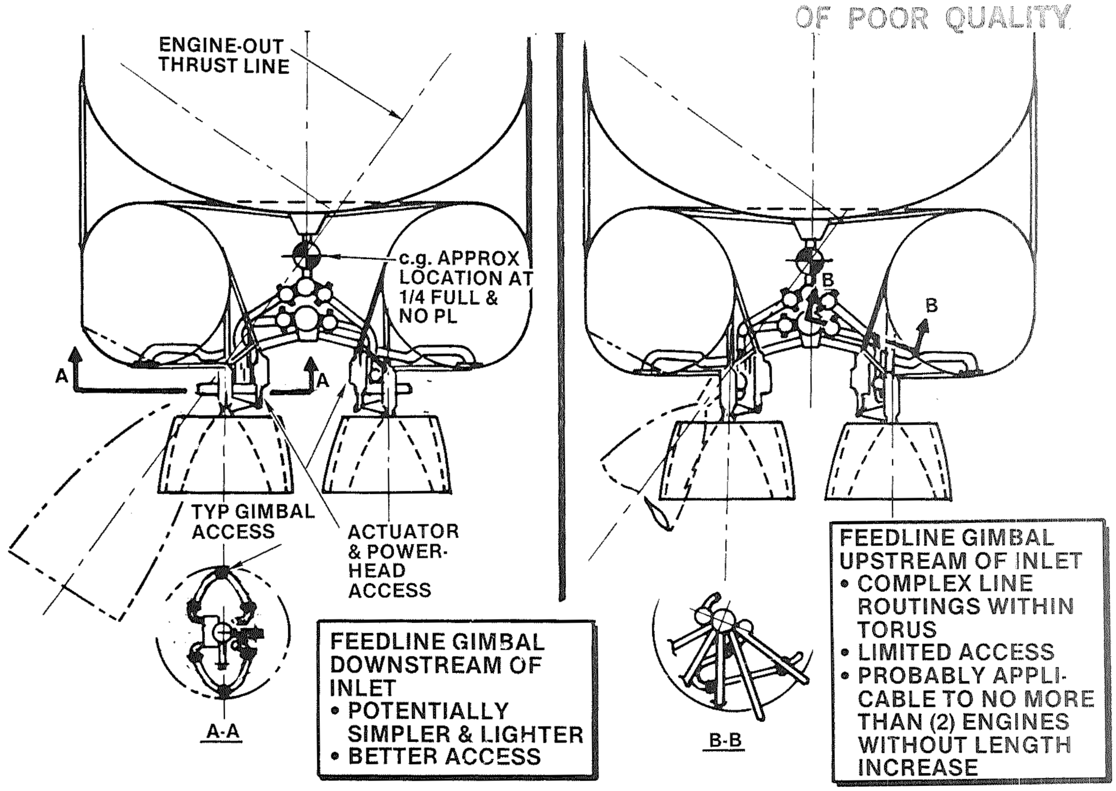


Figure 3. Multiple Engine Installation Options for Toroidal Tank Stage Concept

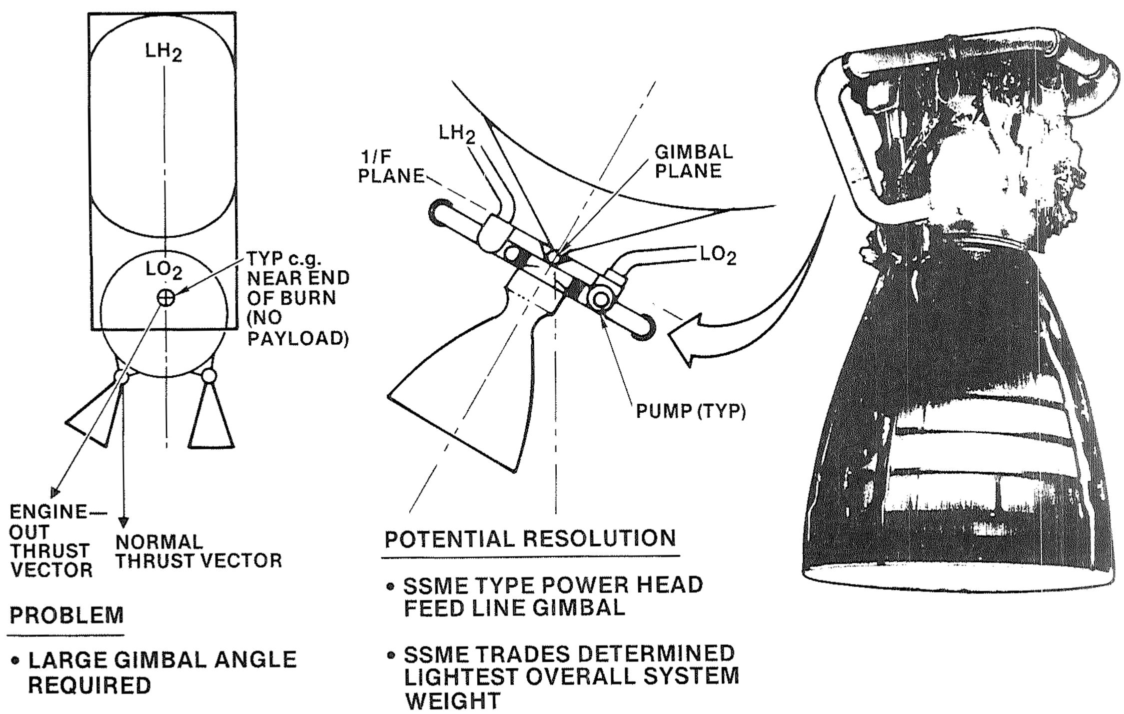


Figure 4. Multiple Engine Installation for Conventional Tank Configuration

OTV SERVICING AND OPERATIONS

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SPACE BASED OTV SERVICING*

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Space based servicing of an Orbit Transfer Vehicle (OTV) has been outlined in sufficient detail to arrive at OTV and support system servicing requirements. Needed space station facilities and their functional requirements have been identified. The impact of logistics and space servicable design on the OTV design is detailed.

INTRODUCTION

The President's proposed Space Station (SS) will provide an excellent base from which to operate a reusable space based Orbit Transfer Vehicle (OTV). Using the SS as a launch and refueling platform will allow the decoupling of the Space Transportation System (STS) earth to low earth orbit (LEO) and the OTV LEO to geosynchronous earth orbit (GEO) legs of payload delivery to GEO. The shuttle will no longer be forced to launch in a window dictated by the payload delivery, but rather on a periodic basis which would allow optimization of ground resources for routine flow. The burden of meeting the launch window then falls upon the SS/OTV system. This implies the need for a highly dependable OTV and OTV support system if the launch windows are to be reliably met.

The OTV support system will in part consist of SS facilities capable of doing routine maintenance and certain contingency repair procedures. It will need an efficient logistics function, as well, to provide needed spares and consumables in a cost effective, timely manner. Implied by this is a highly developed health monitoring system for the OTV and its subsystems. This system must be capable of diagnosing items in need of attention early enough so that the necessary preventative action can be scheduled and lengthy downtimes avoided. All this is made very challenging by the fact that the SS will be able to provide only very limited manned support due to the restricted number of men available, the extreme difficulty of working in the space environment, and the demands of other SS activities.

GROUND RULES AND ASSUMPTIONS

Since none of the hardware actually exists, it is necessary to make a few assumptions and establish sensible ground rules to allow the task to proceed. These are shown in Table 1 and will be briefly discussed below. Since the objective of the present study is to identify engine impacts with regard to servicing, detailed design of the SS support facilities, etc., won't be attempted. Also, assumptions which ease the task of determining representative OTV design and subsequent engine impacts will be made fully realizing that they may not be real.

* This work was performed under Contract No. 127985 with Pratt and Whitney Aircraft, West Palm Beach, Florida

For instance, all refueling operations are assumed to be performed on the SS instead of at a remote propellant farm. Operationally, the only impact is to the timeline. The operations to be performed remain similar. The major assumptions show up in Table 1 while many of the smaller assumptions will be noted in the text, as appropriate.

The present study ground rules are: the use of the space station as the OTV base, STS shuttle as the launch vehicle, manrating of the OTV, LO_2/LH_2 propellants, and the use of an aerobrake with a low lift to drag ratio. From a servicing standpoint, LO_2/LH_2 propellants, manrating, and the aerobrake present the greatest drivers. While the aerobrake itself may not need much servicing, a fixed aerobrake restricts OTV maneuvering about the SS, drives hangar design, and complicates engine servicing. Manrating implies a high degree of reliability/redundancy which in turn impacts the integrity of servicing operations. LO_2/LH_2 propellants have a major impact on the propellant storage and transfer systems and to a lesser extent impacts the engine servicing requirements. Principally, the latter will be concerned only with engine changeout implications and the required health monitoring system and its requirements.

As mentioned above, all space based OTV servicing is assumed to be at the SS and means to maneuver the OTV about the SS are provided. Specifically, the hangar and refueling depot are assumed attached and controlled from a permanent OTV control station at the SS. The OTV control station will control all OTV related operations: data-handling, refueling, line of sight (LOS) proximity operations, maintenance scheduling and procedures (except extra-vehicular activity (EVA)), and SS inventory control. The OTV is assumed to be under ground control for the LEO-GEO-LEO phase of the delivery missions. Both the baseline Rev 6 mission model and the SS Mission Model (ref. 1 Vol. 3) indicate an OTV launch frequency of one every two weeks to one month. Therefore, a two week turnaround will be used as the groundrule.

OTV MISSION FLOW

Given the above assumptions and ground rules, the general OTV servicing flow can be sketched as shown in Table 2. From this list of operations, those pertinant to engine and OTV servicing are further broken out so that an operational and functional analysis can be performed which will reveal the SS facilities needs and the engine servicing impacts. These will be used as a baseline against which alternate servicing concepts will be explored/evaluated. Also, contingency operations such as unscheduled maintenance will be discussed relative to the impact on the baseline functional flow.

A "top down" approach was first used to divide up the nominal two week turnaround so that the maximum available time to do tasks could be delineated. Next, specific individual tasks were considered "bottom up" in that actual times and equipment needed to perform comparable tasks on the ground were determined. In this fashion, areas of further research were identified. For the purposes of this study, the shorter of the two times were used to assemble the timelines shown. Included with the operational analysis are columns indicating facilities needs, and intra-vehicular activity (IVA), EVA and delta time.

Tables 3 and 4 indicate tasks, facilities, and time data for the baseline OTV turnaround. Table 2 presents a top level OTV servicing flow while Tables 3 and 4 break out OTV servicing and engine servicing in further detail, respectively. Complete mission turnaround is shown to take approximately 10 days. This is driven primarily by the LEO-GEO-LEO time and the OTV post mission processing.

The following discussion will cover the OTV mission flow. The "generic" OTV mission is anticipated to begin early with the mission planning activities and other operations by the payload program. The SS begins its preparations 2 to 3 days before the payload is delivered by the STS. The payload is delivered a nominal one week early principally at the convenience of the shuttle and is stored on-board the SS awaiting pre-mission processing. Facilities for handling the payload are presumed available. Their exact manifestation is immaterial, but should include means of mechanically restraining the payload, providing dormant power, data handling, and thermal protection.

A day before the mission the payload is moved into the servicing hangar for final check-out operations. No EVA is anticipated, but could be used if the payload had non-standard interfaces or required some minor contingency repair. For normal operations all pre-mission payload check-out operations shall be handled remotely. The four hours of check-out time are primarily to allow for P/L operations which may be more economically performed on the SS than on the ground. For example, payloads could be launched without fluids to relieve designing for launch loads.

Following successful payload check-out, the OTV will be moved to the servicing hangar for mating with the payload. A final health check will be made of the OTV and the mission parameters will be loaded into the OTV main computer. The OTV to payload interface (I/F) is assumed to be primarily mechanical with a minimal electrical I/F provided. The electrical I/F would be standardized as well as the mechanical I/F. If non-standard I/F's were used, the timeline would need to be modified to allow for OTV I/F modification. No fluid I/F's are anticipated. Two P/L interfaces are implied here: one for the OTV and one for the STS. Once mated and the I/F's verified, the OTV and payload will be moved to the OTV refueling area.

Refueling is performed as the last major operation in the pre-launch flow to avoid bringing a fully loaded OTV into the hangar and to minimize boil-off. This implies a refueling area capable of accommodating the OTV, aerobrake, and payload. The OTV is docked and refueled on the aft end. A fixed aerobrake will complicate the refueling area design. Presumably, a door will be provided in the aerobrake to allow the fluid umbilicals access to the OTV fluid interfaces. The refueling operation itself is the subject of much debate and is simplified here into a tank chilldown operation followed by the bulk fluid transfer. Simultaneous fluid transfer is assumed. Non-hypergolic fluids and "no leak" quick-disconnects (QD's) should allow this. Also, reaction control system (RCS) propellants and pressurants are resupplied in parallel with the main propellants. Pressurant needs should be minimized as much as possible due to the inordinate costs of resuppling pressurants.

Following resupply, a final OTV checkout can be performed (gimbal actuators, pressure checks, etc). The OTV and payload are then disconnected from the refueling area and deployed from the SS. The timeline shown assumes that the SS remote manipulator system (RMS) releases the OTV and payload combination with a small delta-V relative to the SS. The OTV uses a "small" RCS burn to give additional delta-V (about 3 fps) allowing swifter OTV and SS separation. At a safe distance from the SS the OTV control is passed to the ground and the delivery mission begins. An orbital maneuvering vehicle (OMV) could be used to accomplish the same thing.

Space Station control resumes following the return of the OTV to a safe area within LOS of the SS. Here, OTV safing is performed. This may be comprised of venting the OTV propellant residuals. However, this timeline assumes that the cost of propellants is of sufficient importance to warrant recovery. Safing would then entail, primarily, deactivation of the main engines (and the RCS if an OMV is used to recover the OTV). The OTV is returned to the SS following safing either by the OTV RCS or an OMV. The OTV is berthed at the refueling area.

If safing were to entail venting of propellants, this may have a major impact on the OTV. Non propulsive vents must be provided with the appropriate valving and controls. Venting through the engines would be possible but could impose undesirable characteristics on the engine. Additionally, the resulting thrust would need to be accounted for. An OMV would not be able to do this as the OMV would likely be mated to the aft end of the OTV (so its thrust can act through the OTV-P/L center of mass).

Post mission processing is essentially the reverse of the pre-mission flow. The residual propellants are removed after docking at the refueling area. Liquid propellants are returned to the SS cryogen tanks and gaseous propellants are recovered for use by the SS. RCS propellants would also be returned to storage to aid in the accuracy of pre-mission loading (mass measurement errors would otherwise accumulate). It may be desirable to leave a blanket pressure of propellant gases in the tanks for structural reasons.

During the propellant off-loading the SS data handling system will down link mission data from the OTV and return the bulk of this data to the ground where it will be processed. Additional data will have already been sent to the ground during the mission. Some data will also be retained by the SS computer to allow SS personnel to begin post processing scheduling. Quick data analysis and turnaround will be essential to efficient OTV servicing. The bulk of the analysis software is assumed to reside on the ground because it isn't cost effective to burden the SS computer or personnel with this task. Two days are allowed for the ground to return to the SS a preliminary post mission maintenance schedule. During these two days, the OTV would be returned to the hangar if it still has a payload attached. Otherwise, the OTV is moved to its storage area (which may be the servicing hangar - more on this later).

Since post mission OTV servicing is highly dependent upon which maintenance needs to be performed, the routine servicing flow will be discussed along with a separate discussion of major contingency operations such as engine removal or aerobrake repair. Crew time is expected to be an extremely valuable commodity, therefore, routine operations will be highly automated. In addition, the ground processing of mission data will perform an optimization of servicing tasks and return a time table detailing the exact operations to be performed. An approximation is only possible now because both routine, (every mission) and contingency operations will be interwoven to effect the optimization. This approximation appears in Table 2 made up of the scheduled maintenance tasks from Tables 3 and 4.

While the OTV is still berthed at the refueling area, a propulsion system check will be performed. This check will be in support of ground analysis of flight data to determine items in need of maintenance and to execute tests designed to isolate any anomalies detected in the flight data. The objective of this checkout is to drive out any failures which can be remedied in the OTV maintenance to follow. Also, tests which require pressurants will need to be performed here. If the OTV was equipped with removable tanks, the tank operations would be performed in this area. However, removable tanks are not currently envisioned.

All maintenance operations will be performed in the servicing hangar after the schedule has been returned. The first operation will be an overall OTV visual inspection. This could be done EVA, but will likely be done with a closed circuit TV (CCTV) and monitor. In this case, sufficient mobility must be given to the CCTV to allow it to reach all areas of the OTV. Most likely, only specific areas will routinely be inspected such as the engine nozzles, aerobrake, and OTV exterior. CCTV movement could then proceed in a pre- programmed manner and the crew would only override to inspect questionable areas.

It is anticipated that the servicing hangar will provide for checkout umbilicals more extensive than those provided at the refueling area so that specific tests can be run on the avionics. All umbilical actuation will be automated to avoid EVA costs. EVA is anticipated only for non-routine module change out operations, non-routine inspection, and other infrequently performed operations where it won't be cost effective to automate. In any case, after checkout umbilicals are attached the avionics will be checked via checkout software and equipment carried for this purpose. Any anomalies will be noted and factored into the maintenance schedule relayed from the ground. Any EVA operations would be performed following schedule finalization. EVA module changeout would be performed on all items so identified in the preceding checks. This assumes that the proper modules are already on board the SS and the modules were designed for EVA replacement. Both of these assumptions will be discussed more completely later. No modules have yet been identified which will require changeout after every mission. If this were the case, this would likely be accomplished robotically using only one IVA crew man; once again, to avoid EVA costs. A candidate list of EVA replaceable modules is shown in Table 5. This table includes estimated times and anticipated interfaces. Since RCS modules may involve fluid disconnects, two operations are shown to illustrate the differences. The fluid QD's lengthen the time due to the additional effort required to assure the crew's safety (installation of spill containment shrouds and check out following installation).

Two major contingency operations identified are engine removal (which could also be routine) and aerobrake repair. Aerobrake repair is included at this point as a possibility. It is too early to say exactly what aerobrake repair implies or what type of failure it may suffer. Holes could be repaired either by patching or panel replacement. Aerobrake removal to ease servicing would be desirable but isn't a contingency operation. This would be included in overall processing flow near the end of pre-mission processing and the beginning of post mission processing.

ENGINE SERVICING

Several levels of engine maintenance are identified as detailed in Table 4. Two types of scheduled maintenance are shown, operations performed after every flight and those performed every 10 missions. The latter operations are more extensive and performed in addition to the regularly scheduled maintenance. They also include EVA operations (turbo pump inspection and line replaceable unit (LRU) replacement). The engine removal and replacement operation is detailed as well as three possible unscheduled engine repair operations. Unscheduled maintenance could occur on the engine while it is attached to the OTV. This would involve essentially replacement of an LRU that failed prematurely. A removed engine could have a failed LRU repaired in a SS workshop if future analysis showed this to be desirable. Any major repair of the engine will entail removal and return to earth for repair.

The tasks listed are indicative of the types of operations viewed as feasible. Table 6 is an example of the ground maintenance planned for the RL-10 Space Tug Engine (ref. 2). When the engine is further defined, the tasks will need to be re-evaluated. The turnaround maintenance tasks are to be fully automated so they may be performed with IVA. The inspection tasks will need manned involvement, hence the greater man hours assigned to the tasks. These tasks are listed separately from the OTV tasks previously discussed simply for ease of discussion. They would be fully integrated with the OTV tasks as part of the ground timeline optimization performed to arrive at the appropriate maintenance schedule. If an engine removal were scheduled, the inspection would be eliminated.

The engine is expected to be the major item to be serviced and is also the main topic of the present study. The view just presented is the baseline and represents a middle ground; one between the two extremes of the long life, zero maintenance engine and a fully modular, space rebuildable engine. The baseline engine has some LRU's specified (but not identified) so they may be included in the timelines. These LRU's are envisioned to be small items such as transducers or ignitor boxes which can be scarred with EVA compatibility without encurring a large weight or functional penalty. No major items like turbopumps, heat exchangers, thrust chambers etc., are included as LRU's. Failure of these items would entail engine changeout and ground servicing of the failed engine. The weight and functional penalty of making these LRU's is felt to outweigh the advantage of making these EVA compatible. The one major item which may lend itself to EVA (or remote) replacement would be the radiation cooled portion of the nozzle if there were a proven advantage to this. Therefore, the philosophy developed here is that any engine failure other than in a LRU will result in the replacement of that engine. In fact, engines will be replaced prior to failure if the health monitoring system detects an impending failure.

Now that engine removal has been specified, some discussion is warranted on what this will entail. An experienced ground crew under ideal conditions (air conditioned test cell fully equipped with the necessary tools) can remove an RL-10 in about five hours. The EVA crewmen are expected to replace an engine in four hours in the SS hangar. This short time is a goal which makes efficient OTV turnaround a possibility. Two concepts for the engine-OTV interface are shown in Figures 1 and 2. Concepts of this nature will be required. It will be necessary to simplify the OTV-engine interface as much as possible to enable both the engine removal itself and provide the necessary functional integrity to the interface once the engine replacement has been effected. For this reason, it is desirable to eliminate pressurant activated components as this eliminates a gaseous QD from the OTV-engine interface. If the propellant tanks are left with a blanket pressure, a set of valves will be needed on the vehicle side. The main engine valves should remain with the engine so they can be serviced after the engine has been removed (possibly on the SS, likely on the ground). The simplest interface design has all QD's aligned along a plane which separates the engine and the vehicle. This design type would lend itself to remote engine removal, which is a desirable feature. This approach would likely incur a weight penalty relative to an approach which minimizes weight at the expense requiring EVA assistance. Cost modelling of OTV servicing scenarios is expected to aid in recommending which approach to use.

SPACE STATION FACILITIES

The functional and operational analysis just presented have identified five basic space station facilities which will be needed to support a space based OTV. The facilities are shown in Table 7. While the facilities are treated as separate items dedicated entirely to the OTV, in the actual space station they will be more general purpose facilities designed to support the OTV, OMV and other spacecraft designed for SS servicing. At this point, the facilities are separated more for functional reasons than for hardware reasons. The actual SS facilities will probably recombine the functions into units logically arranged as part of the SS design effort. Therefore, the following facilities discussions emphasize the needed functions divided functionally. Possible overlaps are included in the individual discussions.

The servicing hangar will house all the necessary items used for servicing the OTV and other spacecraft. It should be a general purpose facility with some dedicated items specifically for servicing the OTV and the SS OMV as these two spacecraft will comprise the majority of the servicing requirements. A means of mechanically holding the various spacecraft will be needed. A variety of umbilicals will also be needed, mostly electrical. It may be desirable to provide a pressurant umbilical as well. Propellants and other hazardous fluids will be handled at another facility. Power for lighting and power tools should be supplied as well as means of securing the astronaut, his tools, and any other loose items necessary. One current hangar concept (Figures 3 and 4) involves a translation mechanism for the crewmen and a rotary carriage for the spacecraft. This would allow the possibility of a quasi-EMU (extra-vehiclular maneuvering unit) in which the EMU (or spacesuit) shares the SS atmosphere through an umbilical carried with the translation mechanism. In this hangar, total portability would not be necessary since a combination of translation and spacecraft rotation will allow access to all portions of the spacecraft.

As with the servicing hangar, many functions of the SS computer system have already been mentioned. Therefore, they will only be summarized here. Only a small portion of the SS computers' responsibilities will be represented by the OTV activities. The SS computer will function primarily as a link between the OTV computer, ground facilities, and the SS crewmen. OTV data stored during the mission will be down linked to the ground through the SS computer with a portion being retained for the SS crewmen to act upon (SS safety related items, for instance). After ground processing, an estimate of the OTV maintenance schedule will be returned to the SS. The SS computer will then factor in maintenance tasks discovered during post mission processing of the OTV and prepare a final maintenance schedule. The SS computer will also handle loading of the OTV computer with mission specific data prior to the OTV mission. Part of the SS computer will also handle control of the many automated servicing mechanisms. These will include the SS RMS(s), refueling, and CCTV movement.

The above mentioned functions may more logically be part of the OTV control station. Certainly items which are entirely OTV specific will be functions of the OTV control station. The major item here is OTV refueling and OTV LOS control. The SS computer will probably just monitor safety related items so it can respond properly if an emergency were to occur. The bulk of the OTV related software and systems will reside in the OTV control station (functionally at best). The OTV control station will be the primary man-machine link between the OTV and the SS crew. Several OTV display and equipment controllers will be logically arranged here to enable efficient IVA control of the various phases of the OTV mission. The OTV control station, as with the servicing hangar, will probably share hardware with other spacecraft. That, however, is a Space Station issue.

The OTV refueling area will work closely with the control station. The primary function here is, obviously, refueling of the OTV. However, several other propellant and fluid related functions will also be accomplished here. The refueling area will represent a significant portion of the SS mass so its location will be critical to the SS control. The disturbances due to the propellant transfer will also need to be accommodated.

The refueling area will house the cryogen tanks, an OTV mechanical interface, and the necessary umbilicals to allow refueling of all propellants and pressurants. An electrical umbilical is also necessary to allow control of the OTV and down linking of OTV data stored during the OTV mission. It is not envisioned that other spacecraft will be able to utilize this hardware for their refueling. This is due mainly to the physical size of the OTV compared to other spacecraft. Another refueling station will likely be provided by the SS for these smaller spacecraft. (They are also likely to require earth-storable propellants, not cryogens.) Spacecraft wishing to utilize this facility will accomodate the OTV and not vice versa. All the necessary control hardware will reside here (valves, pumps, plumbing, etc.) while the control software will be housed at the OTV control station. One or more CCTV's will be necessary if the refueling area is not visible from the control station.

The space station will need to provide some sort of storage facilities for both the OTV and the various payloads. These facilities will at least provide mechanical hold-down and minimal power and data interfaces to sustain the vehicles in a dormant mode. Desirable features would be thermal and meteoroid protection. The servicing hangar could provide all of these at a loss in utility. These are, of course, space station issues. However, they are worth some discussion here as there are several modifications possible to the baseline timeline. For instance, payload and OTV mating could be performed at the storage area if the proper alignment capability existed. The payload check-out could be performed here as well. This could save time as well as minimizing the movement of masses about the SS thereby saving SS propellant.

As an aside, this brings up the subject of the multiple payload interfaces necessary on the payload that it otherwise wouldn't need. Currently, the STS interface manifests itself as trunnion fittings and an electrical umbilical. The OTV, on the other hand, would require some sort of axially acting mechanical interface and a separate electrical umbilical to that utilized for the shuttle. Presumably one of these two interfaces could be used by the space station storage facility. A trade-off exists between requiring the payload to supply these interfaces and scarring either the shuttle or OTV to eliminate one of the interfaces. Since the payload is launched only once while the STS and OTV make multiple trips, the mass penalty may be best assigned to the payload. This is a subject for further study.

DOWNTIME AND LOGISTICS

The timelines discussed so far are for a routine mission where no major failure has occured which requires a delay to allow the STS to bring up the needed spares or, worse yet, return of the OTV to the ground for extensive servicing. Very few missions are likely to be "routine" and may well require delays which impact the baseline timeline. The learning curve is likely to extend through much of the "routine" mission time frame of the early to late 1990's. A fully debugged OTV-SS system by 1994 is unrealistic and an operational OTV by then is an ambitious goal. However, all the mission analysis to date suggest large payoffs for the ability to fly LEO-GEO missions on a two week schedule. A case for an OTV fleet is emerging.

The other response to downtime impacts is a sufficient spares inventory at the SS to avoid the majority of the delays. Since failures are by nature unpredictable, this implies storing many spares which may never be needed. Unnecessary spares cost both in launch mass for the spares and in the mass of the facilities needed to house them. The space station is not yet envisioned as a flying warehouse. It is bad enough that it is becoming a flying service station - (OTV servicing view point). As a part of the evolving SS and OTV, a comprehensive inventory management effort is recommended which will minimize simultaneously the required mass at the space station and the down time incurred by the OTV. This would entail a high reliability OTV coupled with a component-by-component failure analysis to pin-point likely failures. In addition, grouping the high failure items such that they may be replaced as a unit(s) is required. From day one, the OTV design must adhere to this modular philosophy to some degree. One spare unit capable of remedying several failures will be very valuable. It has already emerged as a conclusion to change out an engine for any major failure rather than repair the engine on the OTV. A reusable space-based OTV cannot be optimized alone. But rather the OTV and its support system should be optimized.

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1. Space Station Needs, Attributes, and Architectural Options Study. Final Report, Vol. III Mission Requirements, Martin Marietta Aerospace, NASA Contract NASW-3686.

2. Design Study of RL-10 Derivatives. Final Report, Vol. III, Part 2, Pratt & Whitney Aircraft, NASA Contract NAS8-28989.

Table 1 Ground Rules and Assumptions

Study Ground Rules
STS as launch vehicle
Space Station facilities in place for OTV
Space based, reuseable LO2/LH2 OTV
OTV to be man rateable and to include an aerobrake
Rev. 6 mission model, 1994 IOC
Structural interface only with payload
Study Assumptions
Space Station to provide up to 4 men/day , 8 hrs/day
Hangar, refueling area, and storage facilities attached to Space Station
OTV ground controled except when within Space Station line-of-sight
2 EVA/1 IVA men per EVA operation
Routine tasks automated as much as possible
Space Station to provide storage for necessary spares
OTV missions to average two week intervals
The OTV to be moved about SS by SS RMS(s)
Times for "routine" operation, not development period
OTV RCS to provide OTV control for "prox ops"
Payload mated to OTV prior to OTV refueling

TABLE 2 OTV Mission Flow

OPERATIONS	TIMELINE
PRE MISSION OPERATIONS	*- 7 days*
- Payload and payload ASE delivery to SS by STS	- 7 days
- uplink mission specific software to SS computor	- 7 days
MISSION PREPARATION OPERATIONS	*- 2 days*
- Move P/L to hangar - perform P/L c/o	- 24 hr
- Move OTV to Hangar	- 20 hr
- Verify OTV readiness - update OTV mission computer	- 19 hr
- Mate OTV and P/L - verify OTV-P/L interfaces	- 18 hr
- Move OTV-P/L to refueling area	- 17 hr
- Secure OTV-P/L and connect umbilicals	- 16.5 hr
- Perform propulsion system check	- 8 hr
- Chill down and fill main tanks with required propellants	- 7 hr
- Resupply RCS and pressurant (if necessary)	- 6 hr
- Disconnect umbilicals - release OTV-P/L - deploy from SS	- 1 hr
- "Small" RCS burn to seperate OTV and space station	- 0.5 hr
- Pass mission control from SS to ground	0.0 hr
PERFORM MISSION, RETURN TO SS LOS	*0 - 3.0 days*
POST MISSION PROCESSING	*3.0- 8 days*
- Pass mission control from ground to SS	80 hr
- Safe main propulsion at TBD miles from space station	80.5 hr
- RCS burn to SS rendesvous	81 hr
- SS capture of OTV-P/L , berth at refueling area	81.5 hr
- Connect umbilicals and off load propellant residuals	82 hr
- Down link OTV mission data to SS and ground computors	83 hr
- OTV exterior visual inspection	85 hr
- Perform post mission propulsion system checks	86 hr
- Disconnect umbilicals and move OTV to Hangar	87 hr
- Demate P/L (if attached)	96 hr
- Prepare P/L for ground return (if necessary)	100 hr
- SS and Ground computors return OTV status and service requirements	120 hr
- Perform OTV service as required	122 hr
- Prepare OTV for storage and move to storage	168 hr

Table 3 OTV Servicing Overview

Operation	Facilities	Tools	Delta Time	SS Man Hours		
				IVA	EVA	Total
Main Propellant-Resupply	Resupply Area,RMS Cryo Tanks,Umbil.	Resupply Software, Control System	7.0	7.0	-	7.0
Avionics - Sched. Maint.	SS Hangar		6.0	6.0	6.0	12.0
- Module Test	SS Computor	Test + Access Tools	3.0	3.0	-	3.0
- Module Replacement	EMU,HPA,Lighting	LRU ASE,Remov. Tools	3.0 (1)	3.0	6.0	9.0
- ACS Update	SS Computor	SS & ACS Software	0.5	0.5	-	0.5
Avionics - HM Maint.	SS Hangar					
- Module Replacement	EMU,HPA,Lighting	LRU ASE,Remov. Tools	1.0 (1)	1.0	2.0	3.0
- Module Repair	SS Work Shop	Electronics Tools	1.0 (1)	1.0	2.0	3.0
Avionic-Mission Peculiar	SS Hangar					
- Module Replacement	EMU,HPA,Lighting	LRU ASE,Remov. Tools	1.0 +	1.0 +	2.0 +	3.0 +
- Reconfiguration	EMU,HPA,Lighting	LRU ASE,Remov. Tools	2.0 +	2.0 +	4.0 +	6.0 +
Tanks - Sched. Maint.	Resupply Area		2.0	2.0	-	2.0
- External Inspection	CCTV Monitor	RMS + CCTV	1.7	1.7	-	1.7
- PU and TVS System	SS Computor	Test + Access Tools	0.3	0.3	-	0.3
Tanks - Unsched. Maint.	SS Hangar					
- Tank Removal	Res. Area,RMS Console	RMS, CCTV, Tank ASE	2.0	2.0	4.0 (2)	6.0
- Insulation Repair	EMU,HPA,Lighting	Insul. Rep. Kit	3.0	3.0	6.0	9.0
- X-ducer Replacement	EMU,HPA,Lighting	LRU ASE,Remov. Tools	1.0	1.0	2.0	3.0
- PU and TVS System	SS Hangar, EMU,HPA	LRU ASE,Remov. Tools	2.0	2.0	4.0	6.0
Tanks - Mission Peculiar	Resupply Area					
- Tank Reconfiguration	Res. Area,RMS Console	RMS, CCTV, Tank ASE	2.0	2.0	4.0 (2)	6.0
RCS - Scheduled Maint.	Resupply Area,Umbil.		1.0	1.0	-	1.0
- Leak check	SS Computor,Press	RCS Software	0.5	0.5	-	0.5
- X-ducer Check	SS Computor	RCS Software	0.5	0.5	-	0.5
RCS - Resupply	Resupply Area,Umbil.	RCS Software	2.0	2.0	-	2.0
RCS - Health Maint.	SS Hangar					
- X-ducer Replacement	EMU,HPA,Lighting	LRU ASE,Remov. Tools	2.0	2.0	4.0	6.0
- Thruster Replacement	Hangar,EMU,HPA,Light	LRU ASE,Remov. Tools	2.0	2.0	4.0	6.0
Struc.& AB - Sched. Maint.	SS Hangar					
- Inspection	CCTV Monitor	RMS + CCTV	1.0	1.0	-	1.0
Struc.& AB - HM Maint.						
- AB Refurbishment	Resupply Area,EMU,MMU	AB Repair Kit & Tools	3.0	3.0	6.0	9.0
- Structure Repair	SS Hangar,EMU,HPA	LRU ASE,Remov. Tools	2.0	2.0	4.0	6.0

NOTES :

(1) Mission average expected for all avionics modules, 1 hr for contingency

(2) Tank replacement only for Modular OTV, Some EVA assistance anticipated

Table 4 Engine Servicing Overview

Operation	Facilities	Tools	Delta Time	SS Man Hours IVA	SS Man Hours EVA	SS Man Hours Total
Engine-Turnaround Maint.	Space Station Hangar		3.4 hr	5.4	-	5.4
- Analysis of flight data	SS & Grnd Computor	Engine Software	2 days	2	-	2
- Lock up pressure decay	SS Computor, Refuel	Engine Software	0.5 hr	0.5	-	0.5
- Engine valve op check	SS Computor, Refuel	Engine Software	0.5	0.5	-	0.5
- Nozzle visual inspec.	SS Computor, Refuel	RMS + CCTV	0.6	0.6	-	0.6
- Nozzle extension check	SS Computor, Refuel	RMS + CCTV	0.2	0.2	-	0.2
- Gimbal actuator check	SS Computor, Refuel	RMS + CCTV	0.2	0.2	-	0.2
- Connect Umbilicals	SS Hangar	RMS	0.3	0.3	-	0.3
- Turbopump torque check	SS Computor	Engine Software	0.2	0.2	-	0.2
- Ignition system check	SS Computor	Engine Software	0.3	0.3	-	0.3
- Instrumentation c/o	SS Computor	Engine Software	0.5	0.5	-	0.5
- Solenoid c/o	SS Computor	Engine Software	0.3	0.3	-	0.3
- Disconnect Umbilicals	SS Hangar	RMS	0.2	0.2	-	0.2
Engine - Periodic Maint.	SS Hangar		4.0	4.0	6.0	10.0
- Setup operations	SS hangar	Engine tools, LRU ASE	0.5	0.5	1.0	1.5
- Turbopump boroscope	Power, Lights	Boroscope	1.0	0.5	1.0	1.5
- Thrust chamber inspec.	CCTV Monitor	RMS, CCTV	1.0	1.0	-	1.0
- Engine LRU replacement	Power, Lights, EMU, HPA	Engine Tools, LRU ASE	2.0	1.5	3.0	4.5
- Tool stowage	SS hangar	Engine tools, LRU ASE	0.5	0.5	1.0	1.5
Engine -OTV Engine Remove and Replace	SS Hangar, RMS, EMU Foot restraint Lighting	Engine Fixture, Engine Discon. tools, Protective covers	5.0	3.8	6.0	9.8
- Setup tools			0.5	0.5	1.0	1.5
- Attach engine fixture			0.5	0.5	1.0	1.5
- Disconnect engine			0.5	0.5	1.0	1.5
- Move engine to storage			0.2	0.2	0.4	0.6
- Pickup replacement			0.1	0.1	0.2	0.3
- Align and attach			0.7	0.7	1.4	2.1
- Check/verify QD's			2.0	0.8	0.5	1.3
- Store tools			0.5	0.5	0.5	1.0
Engine - Unsched. Maint.						
- Repair in hangar	SS Hangar, RMS, EMU, HPA	Above	2.0 +	2.0 +	4.0 +	6.0 +
- Repair LRU in SS	SS Hangar, RMS, EMU, HPA	Above plus LRU ASE	3.0 +	4.0 +	4.0	8.0 +
- Repair on Ground	SS Hangar, RMS, EMU, HPA	Above plus Engine ASE	2.0	1.7	3.4	5.1

Table 5 EVA Replaceable Modules:

SUBSYSTEM	MODULE	CONTENTS	INTERFACES	R&R TIME
Avionics	Main computor	CPU, I/O unit, Memory	Mech, Elec.	0.5 hour
	TT & C	Antenna (s)	Mech, Elec.	0.2 hour
	C &DH	RF Electronics	Mech, Elec.	0.5 hour
	Guidance	Gyros	Mech, Elec.	0.4 hour
Reaction Control System	RCS Module	Tanks, valves, thrusters heaters, & transducers	Mech, Elec	0.6 hour
	Thruster	REA valves, thrust chamber, heater	Mech, Elec & Fluid	1.5 hour
Electical Power System	Power supply	Fuel cells, valves, tanks heat exchanger,& pumps	Mech, Elec	0.5 hour
	Fuel cell	Fuel cell module	Mech, Elec & Fluid	1.5 hour
Structure & Aerobrake	Aerobrake	Aerobrake module	Mechanical	1.0 hour

Table 6 RL10 Derivative Rocket Engine Inspection Task Times

Inspection Area	Type of Inspection	Type of Fault	Inspection Technique	Access	ML	Total MMH	Elapsed MMH	
			Periodic/Phase Inspection Operations					
Thrust Chamber Assembly	External-Thrust Structure	Deformation and Structural Integrity	Tolerance Measurement, Dye Penetrant and Radiography	Directly Accessible	I	1.50	0.75	2 men
Extendible Nozzle	External-Structure	Thermal Damage	Visual	Directly Accessible	I	.17	.17	
Turbopump Assembly	Internal-Bearings	Signs of Thermal Damage, Cage Damage	Use of Borescope	Typical Access Ports No. 1, 2, and 3	I	2.00	1.00	2 men
	Internal-Seals	Excessive Seal Leakage	Pressurize Sub-System	Turbopump As Installed	I	.50	.50	
	Internal-Seals	Excessive Wear	Radioisotope	Turbopump As Installed	I	1.50	.75	2 men
	Turbopump Gears	Signs of Excessive Tooth wear	Use of Borescope	Typical Access Ports No. 1, 2, and 3	I	Include with bearings		
Turbopump	Torque-Check	Bearings and Shaft Fit	Torque Tool	Oxidizer Pump Closure Plate	I	.25	.25	
Helium System	Internal Configuration	Internal Leaks	Helium Consumption Rate	Engine-As Installed	I	.17	.17	
Flow Control	External-Total Valve Inventory	Leak Check (Internal) and Actuation Cycle	Pressure to Verify Seal Operation and Position Indicators	Engine-As Installed	I	.50	.50	
	Automatic Checkout	Actuation Timing, Position Indications	Comparison to Historical Data	Engine-As Installed	I	.25	.25	
	External-Valve Weldments & Flanges	Leak Check (External)	Visual - Leaks	Engine-As Installed	I	Include with eng. plumbing		

Table 6 RL10 Derivative Rocket Engine Inspection Task Times (Continued)

Inspection Area	Type of Inspection	Type of Fault	Inspection Technique	Access	ML	Total MMH	Elapsed MMH	
Gimbal Assembly	Load Checkouts	Excessive Wear	Gimbal Power Requirement Check	Engine-As Installed	I	.50	.50	
Engine Plumbing	Leak Check	Leaks	Visual	Engine-As Installed	I	2.00	1.00	2 men
				TOTALS		9.34	6.09	
			Turn Around Inspection Operations					
Engine Assembly	External-Weldments, Ducts, Components, Fluid Lines, and Hardware	Damage, Component Security, Loose Hardware	Visual	As Installed	I	.50	.25	2 men
	Diagnostic Review	All	Computer Comparison of Operating Signature	N/A	I	.25	.25	
Thrust Chamber Assembly and Extendible Nozzle	Internal Combustion Chamber Wall and Injector Face	Signs of Thermal Damage (Corrosion, Cracking, Plugging)	Visual	Throat	I	.17	.17	
"Hot Section"	Weldments, Ducts, Manifolds and Chamber Tubes	Damage	Visual	Directly Accessible	I	.25	.25	
	Expansion Nozzle	Tube Cracks, Splits, Holes	Visual	Directly Accessible	I	.17	.17	
	Extendible Nozzle	Signs of Thermal Damage	Visual	Directly Accessible	I	.17	.17	
Ignition System	Internal-Spark Ignition	No spark	Visual	Directly Accessible	I	.17	.08	2 men
				TOTALS		1.68	1.34	

Table 7 Space Station OTV Servicing Facilities

SPACE STATION HANGAR - Provides meteor and thermal protection for OTV and payloads. - Provides power, data, command, and pressurant umbilicals - Storage and use of OTV and payload handling cradles - General purpose RMS's, astronaut foot restraint/positioning aid (HPA), tool and LRU caddies
SPACE STATION COMPUTOR - Refers to entire SS C&DH system, including ground and S/C links as appropriate - Stores and executes routine servicing tasks, updating as needed from ground - Assumes major portion of task scheduling operations - Assumes major portion of RMS control and other robotics
OTV CONTROL STATION - Used for OTV-P/L control for LOS operations - Allows mission monitoring while OTV-P/L are under ground control - Used to monitor/control OTV refueling operations
OTV RESUPPLY AREA - Provides all mechanical, electrical, and fluid interfaces for OTV main engine and RCS - Provides propellant storage and fluid transfer control hardware - OTV removable propellant tank handling hardware
OTV STORAGE AREA - Provides mechanical Hold-down , dormant power ,and health monitoring - Could provide thermal and meteoroid protection

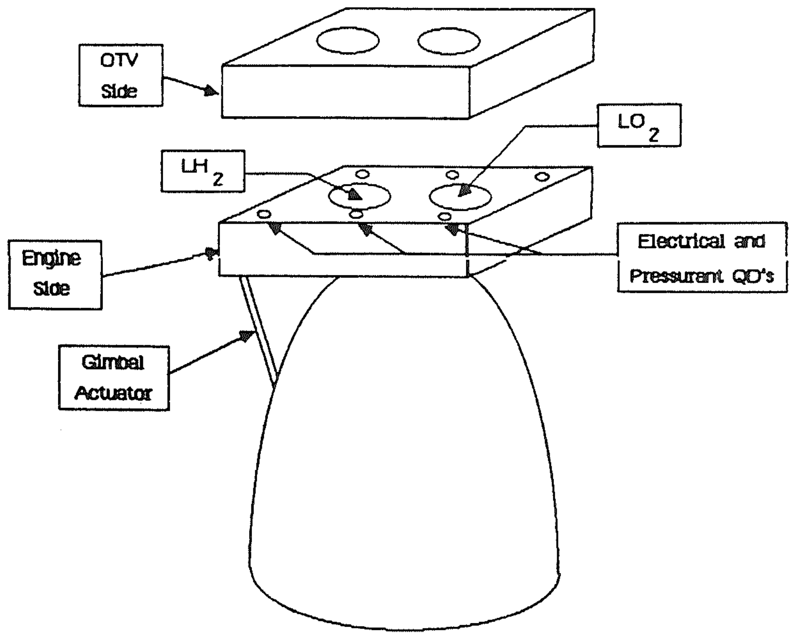

Figure 1 Planar OTV-Engine Disconnect Concept

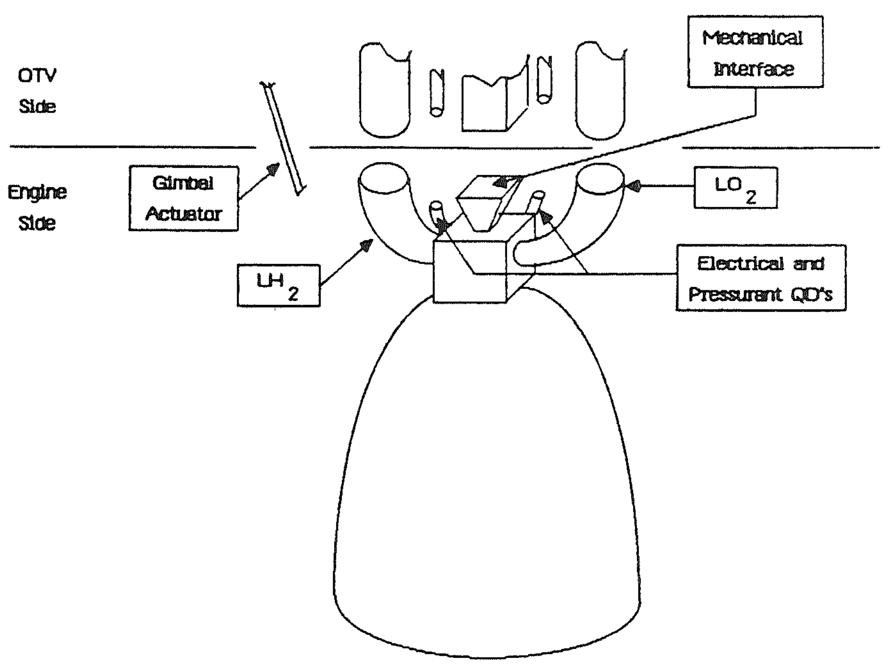

Figure 2 Discrete OTV-Engine Disconnect Concept

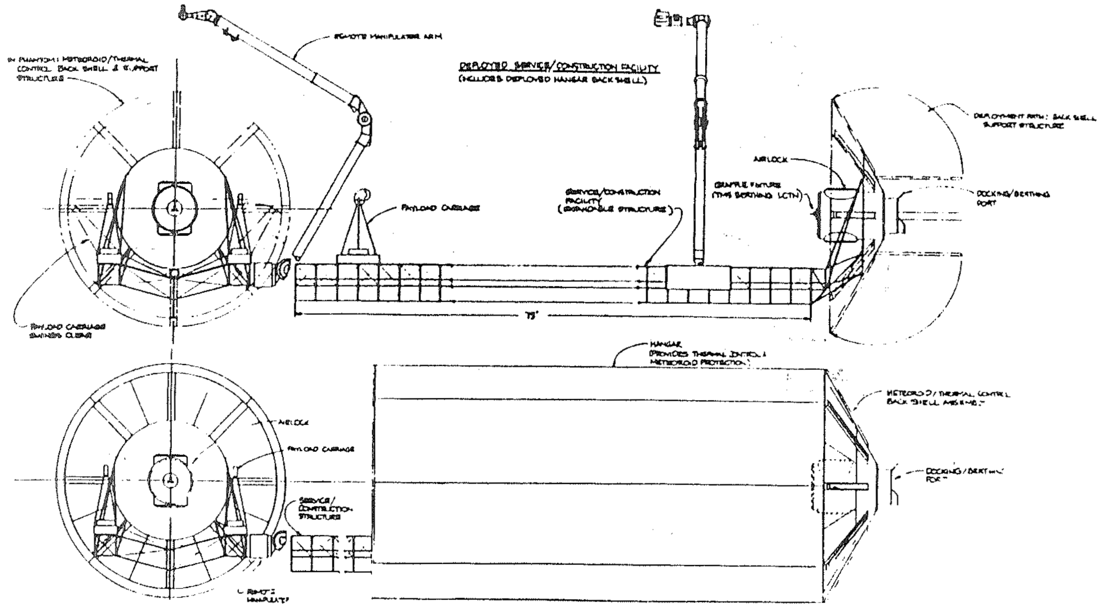

Figure 3 - Deployable Hangar Concept

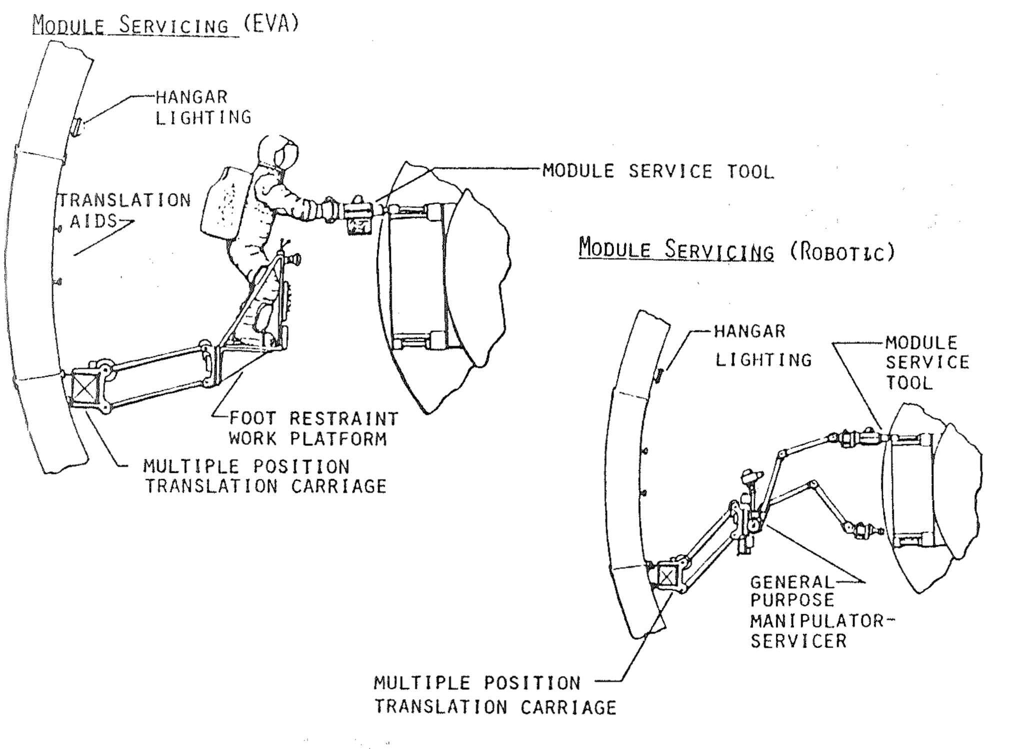

Figure 4 Hangar Servicing Concepts

OMIT TO END

W. J. Kitchum
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The major goals of a reusable space-based OTV are identified. In addition, the benefits that a space-based OTV offer over a ground-based OTV such as increased performance due to reduced inert weight/propellant requirements and reduced cost due to elimination of repeated vehicle deliveries from Earth are discussed. The OTV mission requirements for 1991 to 2000 are delineated including satellite delivery, low thrust, and manned missions. Several candidate configuration options to meet these mission requirements are presented. A comparison of the cost of a pound of payload to GEO for existing upper stages and proposed reusable ground-based and space-based OTV's is presented showing the space-based OTV to be the most economical.

A representative space-based OTV servicing facility on the Space Station is presented showing the required berthing trusses, maintenance shelter, propellant storge tanks, etc. The maintenance philosophy which was followed in deriving the space operations tasks and timelines, is delineated indicating that the space-based OTV design is predicted on a modular approach with good maintainability/accessibility features as a major design driver. General Dynamics extrapolated its experience on current ground-based operations to arrive at the tasks and timelines for space-based operations. The required space-based operations and timelines are compared to the equivalent ground-based operations and an assessment is presented for some of the basic differences in the two operations. The results show that for an average mission, it takes approximately 40 hours of crew time and up to four crewmen for the space-based operations. For an average mission, it takes approximately 150 hours of hands-on-the-vehicle technicians time and 35 different crewmen for the ground-based operations. Basically, the ground-based operation requires more people because the vehicles, so far, have been designed primarily for performance optimization, and maintainability and accessibility has not been design drivers. Likewise, the vehicle has been constrained by cargo bay dimensions for accessibility. In contrast, the space-based OTV will be designed for maintainability/accessibility and will have more built-in test equipment for checkout because man hours are very expensive on orbit.

A list of technology needs to meet the space-based OTV goals is presented, and a recommendation that these technology needs be pursued vigorously in the near future and that further study is needed to define candidate OTV/Space Station accommodations is put forth.

OTV MISSIONS
1991 - 2000

- NASA, DoD & commercial ETR missions considered
- GEO satellite missions
  - 70% commercial & NASA market share — 5 to 7 missions per year (3 to 4 satellites manifested on each mission ≈ 10,000 lb)
  - Servicing — 2 missions per year
  - DoD — 6 missions per year
- Low thrust LSS missions
  - 10,000 to 16,000 lb payload
  - 2 to 4 missions per year (>1994)
- Manned GEO sortie missions
  - 1 per year (>1995)
  - 13,000 lb payload round-trip

Figure 1

Review of planned OTV missions indicates that most will be to deliver GEO satellites. LSS and manned missions are later and fewer.

REPRESENTATIVE OTV CONCEPTS

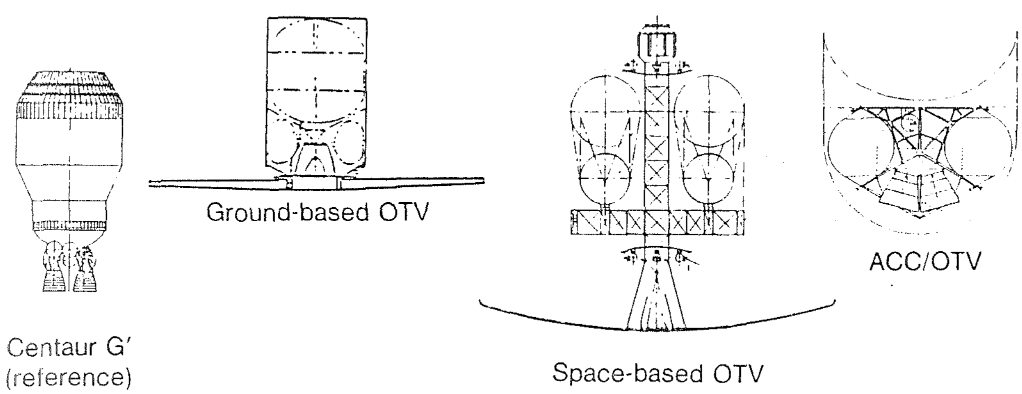

Figure 2

OTV concepts are compared. The space-based OTV illustrates the impact that removal of launch constraints has on design.

TRANSPORTATION COSTS TO GEOSYNCHRONOUS ORBIT
(85 Klb Payload STS)

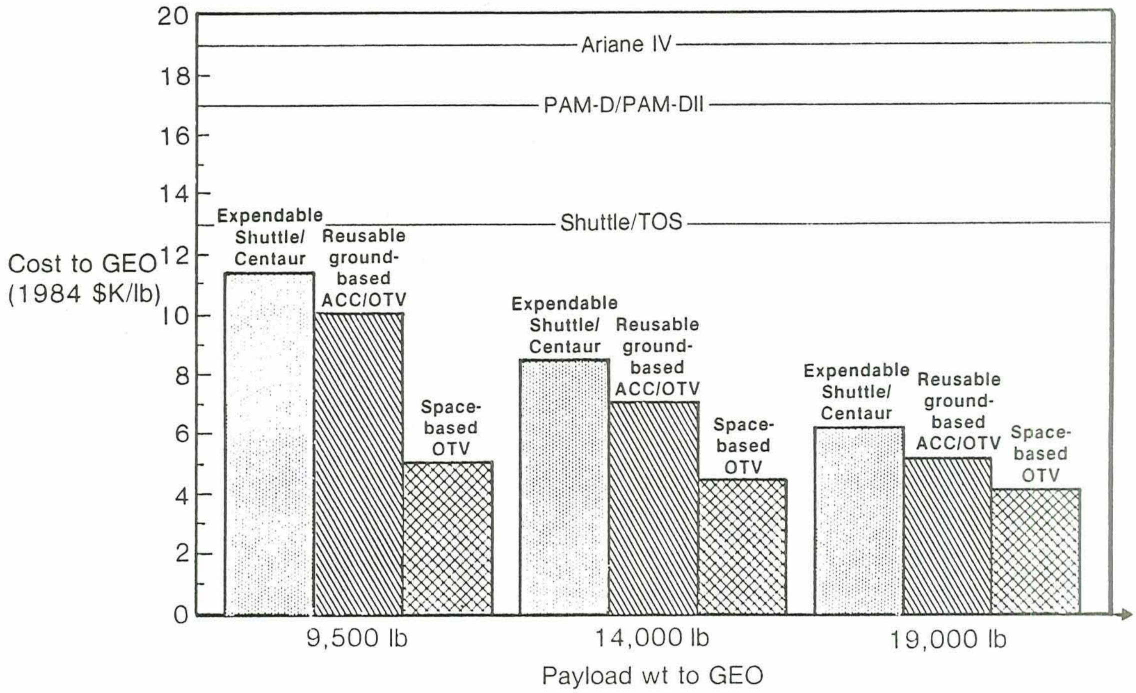

Figure 3

Effective cost per pound of payload delivered to GEO is shown for current expendables and planned reusable orbital transfer vehicles. Costs include STS, OTV, and operations. Low cost propellant delivery (tanker) is assumed for space-based OTV's.

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REPRESENTATIVE SPACE-BASED OTV SERVICING FACILITY

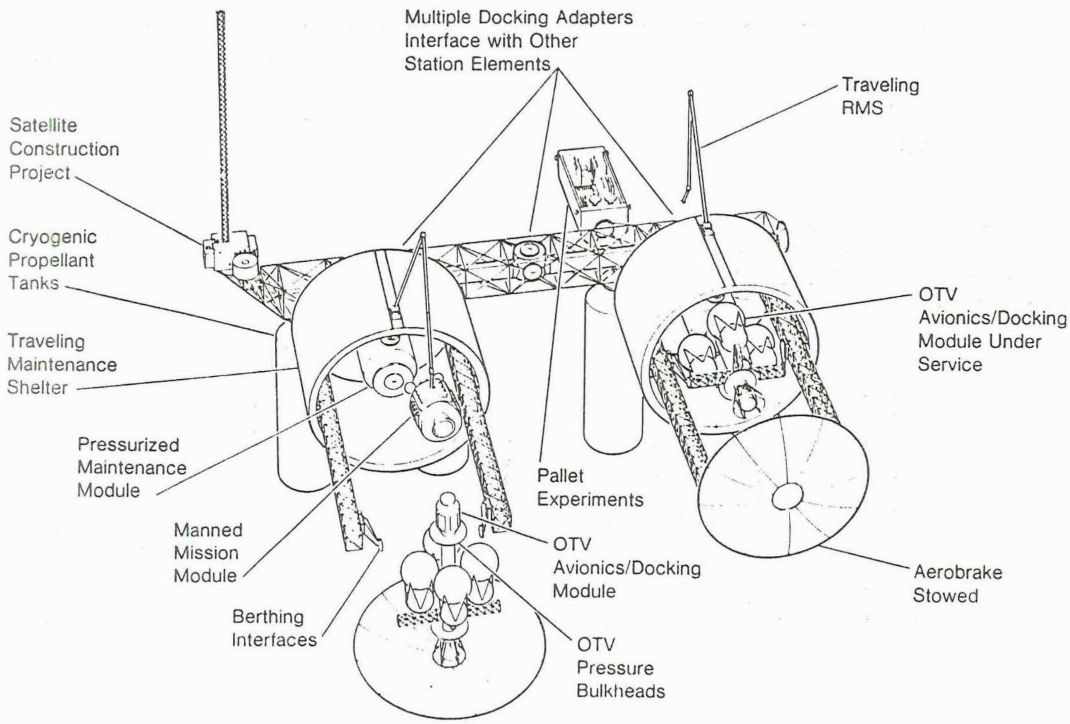


Figure 4

A space-based OTV servicing facility is shown identifying the operations and maintenance functions involved.

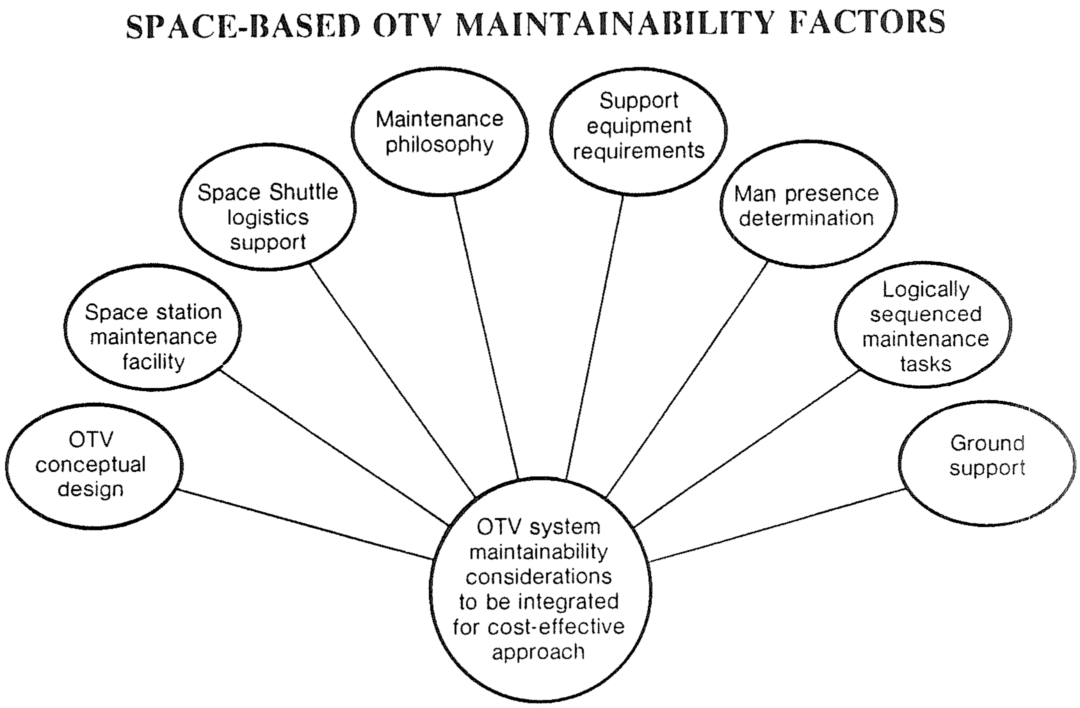


Figure 5

These space-based OTV system elements were identified as being the major factors contributing to and impacting the maintainability of an OTV in space. We constantly weighed each of these maintainability factors in formulating the Space-Based OTV concepts.

OTV MAINTENANCE PHILOSOPHY

Three-level maintenance — based on level-of-repair analyses

- I OTV local maintenance
- II Space station maintenance of replaceable units
- III Return-to-earth maintenance

Stock spare parts based on reliability, criticality & cost

- Station storage vs shuttle delivery

Stress modular construction for replacement capability

Provide operational flight instrumentation & built-in test

- Fault isolate to replaceable unit

Optimize EVA vehicle maintenance operations

- Consider safety in hazardous situations
- Tradeoff EVA vs support equipment
  - TV inspection
  - Robotic remove & replace

Figure 6

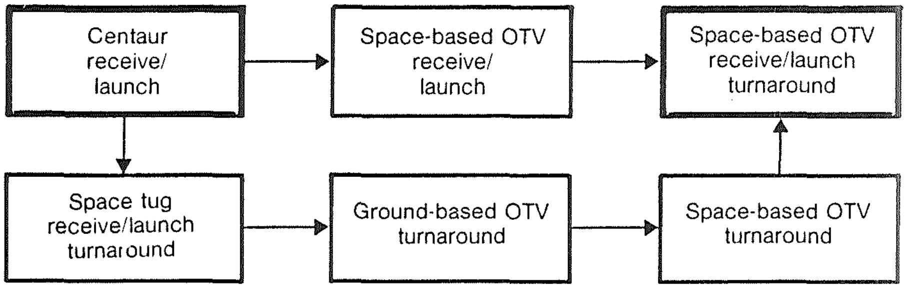
An OTV maintenance philosophy encompassing Space Station operations was developed to help us focus on the essential elements of maintenance support requirements. The maintenance philosophy is based on the three levels of maintenance described below.

Level I maintenance consists of the scheduled and unscheduled activities that occur on the vehicle while it is berthed in the Space Station maintenance dock.

Level II maintenance encompasses the off-vehicle repair of replaceable OTV components conducted at the Space Station. The OTV replaceable units will be dispositioned for return to earth or repaired at the station to the extent possible within the test equipment, spares availability and economic constraints.

Level III maintenance will involve normal earth oriented disposition for repair. An extensive analysis will ultimately provide the necessary repair or discard decision criteria.

EXTRAPOLATION OF CURRENT GROUND-BASED OPERATIONS TO SPACE-BASED OPERATIONS



- Functions
- Manpower/skills
- Function allocations between ground & space
- Implications to SB OTV design
- Space station support requirements

Figure 7

This chart is a road map showing how we have extrapolated our present experience with cryogenic upper stages to arrive at the tasks/manhours/number of men for a space-based operation. We are using our actual Centaur experience for receive and launch operations. We have used that experience in the past to come up with projected turnaround tasks for a ground-based vehicle. This was accomplished on the Space Tug Study in the early 70's. We also looked at the turnaround of a ground-based OTV in a study for MSFC in 1980. Using this information as a data base, we performed our operations analysis to identify the required space-based operations/timelines/manpower.

GROUND-BASED VEHICLE TURNAROUND ASSESSMENT

- Ship, integrate & launch status has not been attained
  - Tendency to ship short & assemble missing parts later
  - Requires some disassembly & component checkout
  - Assumes man can compensate for system shortcomings
- Vehicles designed primarily for performance optimization
  - Maintainability & accessability not a design driver beyond providing access panels
- Checkout accomplished with GSE external to vehicle
  - Requires multiple interfaces (manual connection)
- Personnel required to analyze data & write maintenance plan
- Preventive & corrective maintenance accomplished manually
- Inspection requires dismantling to verify vehicle integrity
- Operation requires download, upload & integration with shuttle
- Operation requires transport & interface with maintenance facility
- QA & safety support required because of dismantling process & personnel involvement

Figure 8

We have made an assessment of how we would turnaround a ground-based vehicle under today's conditions at our facility at the Eastern Test Range in Florida. This was done to identify the tasks to be performed, the timelines, and the number of different personnel involved. We used this as a data base for generating the turnaround tasks to be accomplished at the Space Station for a space-based OTV.

First of all we must characterize a present day reusable, ground-based vehicle and how it is processed on the ground so that a comparison with an improved state-of-the-art space-based vehicle can be meaningful. This chart characterizes a potential present day reusable, ground-based vehicle and how it would be handled at ETR. The background of how we have checked-out and launched upper stages in the past has a big impact on the approach used today. Present day vehicles were not designed using maintainability and accessibility as design drives. The types of operations required to be performed are fairly labor intensive. In addition, a ground-based vehicle must be downloaded, uploaded and integrated with the shuttle.

SPACE-BASED OTV TURNAROUND ASSESSMENT

- Vehicle is fully checked on ground with planned assembly at the space station
- Turnaround operations are optimized by restriction to Level I maintenance
- Maintainability is a primary vehicle/system design requirement
  - Accessibility for remote & EVA operations
  - Modular construction of space-based OTV simplifies & speeds up replacement process
- Checkout accomplished with vehicle built-in test capability
  - Vehicle computer system evaluates & registers fault during mission
  - Vehicle status relayed to station via RF datalink or through data bus interconnect after berthing
  - Interfaces automatically connected during berthing operations
- Computer system analyzes & displays vehicle status & presents basic maintenance plan
- Majority of maintenance tasks are accomplished by semiautomatic (or robotic) equipment
- Inspection by TV without tear down operation
- No shuttle interface operations required beyond initial delivery
- Vehicle is not subjected to space–Earth transition environment
- Vehicle berths at maintenance facility
- Operations philosophy assumes vehicle is operational after good flight with aid of instrumentation & computer assessment
- Vehicle does not need to be dismantled after each mission, which minimizes damage due to maintenance operations

Figure 9

An assessment of Space Station operations, maintenance philosophy and space-based OTV design features was conducted to determine what the differences were from a ground-based system that affect the turnaround times and crew requirements. The results of the assessment are presented here to provide some of the reasons why a space-based OTV turnaround operation can be accomplished in less time and with considerably fewer men than a ground-based operation.

COMPARISON OF GROUND-BASED vs SPACE-BASED TASKS

Time (hr)	MH	Task No.	Reusable Ground-Based Vehicle — Task	Space-Based OTV Equivalent Task	Time (hr)	MH
8:00	64:00	1.1.1	Analyze data & prepare maintenance plan	Query computer about fault status	:15	:30
5:00	365:00	1.1.3	Transfer stage from pallet to maintenance & test stand	OTV docks at maintenance facility (includes rendezvous & capture)	3:40	7:20
3:00	68:00	1.1.4	Remove stage access doors & connect GSE	Automatic connection through berthing interface	**	
11:00	288:00	1.1.5.1	Inspect structural elements & thermal control	Visual inspection (TV)	2:00	4:00
		1.1.5.2	Inspect tanks, supports & interior	Visual inspection (TV)		
		1.1.5.3	Inspect MLI & thrust structure	Visual inspection (TV)		
		1.1.5.4	Inspect docking mechanism	Visual inspection during capture before docking (TV)		
		1.1.5.5	Inspect avionics & flight control units	Visual inspection (TV)		
		1.1.5.6	Inspect engine flood & pressure lines	Visual inspection (TV)		
		1.1.5.7	Inspect fuel cells	Visual inspection (TV)		
4:00	32:00	1.1.6.1	*Perform scheduled checkout & fault isolate*	Initiate test routine & fault isolate	*	
16.00	79.00	1.1.6.2	Perform leak check on LH_2 & LO_2 tanks & engine	Monitor for propellant leakage	:15 → 30:45	.30
		1.1.6.3	Inspect stage/orbiter interface (post-flight fault ISO)			
8.00	80.00	1.1.7	Review inspection & checkout results & complete maintenance plan	Formulate integrated maintenance plan (partially automated function)	:30	1:00
8.00	160.00	1.1.8	Perform unscheduled maintenance	Perform unscheduled maintenance	Avg *(2.45)	Avg (8:30)
		1.1.9	Perform scheduled maintenance — structures	Perform scheduled maintenance	8:35	17 10
20:00	842.00	1.1.10	Perform scheduled maintenance — avionics			
		1.1.11	Perform scheduled maintenance — propulsion			
		1.1.12	Perform scheduled maintenance — thermal control			
1:00	8:30	1.2.1	Mate stage & stage/orbiter adapter			
5:00	50:00	1.2.2	Check out docking mechanism			
Not in time line		2.1.5	Prepare for storage	Deactivate & stow all systems	**	
		2.1.6	Monitor stage in storage			
		2.1.7	Remove from storage	Activate OTV & maintenance facility	**	
		2.1.8	Accomplish mission-peculiar	(Not defined at this time)		
16.00	320.00		preparations	Perform system operational testing	45	1:30
Not in time line		2.1.9	*Perform systems test*	Perform corrective maintenance	*	
Not in time line		2.1.10	Correct faults	Perform system operational testing after corrective maintenance	*	
		2.1.11	Reverify system after correction			
7:50	77:80	2.1.12	Secure from system test			
3:30	52:30	2.2.5	*Mate stage & spacecraft*	Mate payload to OTV	4:15	8.30
1:00	8.00	2.2.6	Verify stage/spacecraft interface	Verify OTV/payload interface	:15	:30
Not in time line		2.2.7	Perform integrated system test	Perform payload/OTV integration test	:30	1.00
3:30	21.00	2.3.1	*Transport payload (stage & spacecraft)*			
6:30	77:00		to orbiter			
2:00	24:00	2.3.2	Install in orbiter			
5.00	25.00	2.3.3	Verify orbiter/payload interface			
1:00	4:00	2.3.4	*Conduct integrated systems test (ORB/PL)*			
		2.3.5	Check status — stage/shuttle interface (after shuttle up-load)			
3:00	12:00	2.4.1	Conduct orbiter/payload integrated test			
1:00	4.00	2.4.3	Conduct launch readiness test (stage)	*Perform prelaunch operations*	4:00	8:00
4.00	20.00	2.4.4	Load propellants & pressurants	Transfer propellants from station to OTV	6:00	12:00
0.25	1:30	2.4.5	Conduct terminal countdown	Launch OTV/payload	1:45	3:30
.30	2:30	3.1.1	Safe stage	Transfer OTV propellants to station	1:45	3:30
6:00	48.00	3.1.2	Purge main propellant tanks	(Pertinent OTV data in computer memory)		
:30	1.00	3.1.3	Remove flight data recorder tapes			
8:30	110.30	3.2.1	Remove stage from orbiter			
2:00	18.00	3.3.1	Transfer stage to TMF			

* *Not in normal turnaround*
** Incorporated in other task

Figure 10

The tasks and times required to perform a turnaround operation on a reusable ground-based vehicle were identified, which provided the data base necessary to produce the space-based OTV turnaround timelines. This space-based OTV data is listed here to correspond with the summary ground-based tasks to show a comparison between the two different modes of operation.

OTV TURNAROUND
Shelter Configuration — No EVA

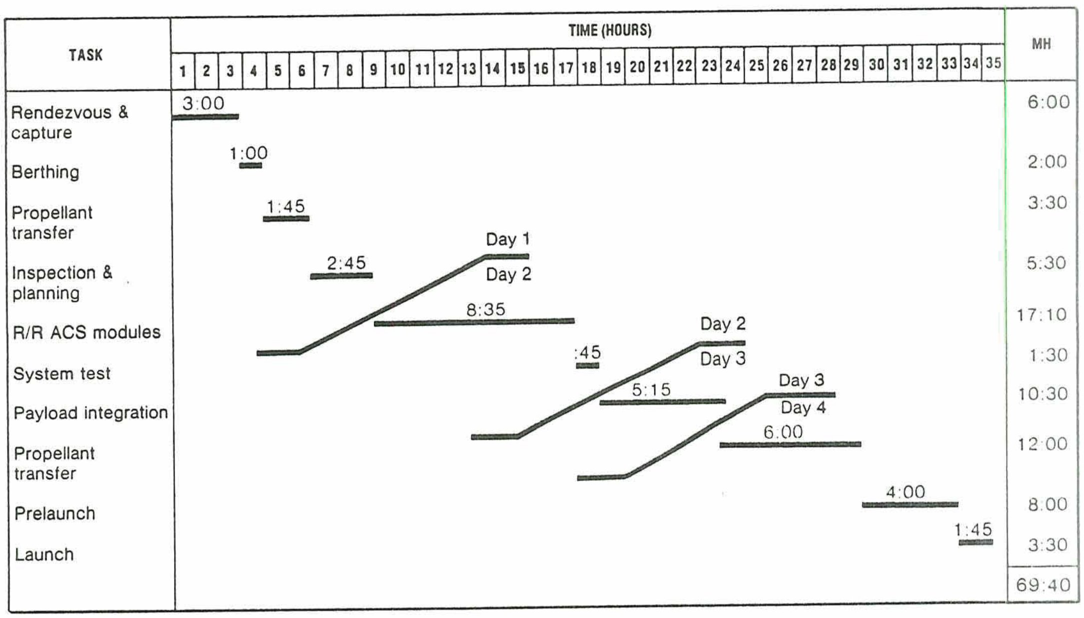


Figure 11

The normal space-based OTV turnaround timeline begins when the OTV approaches the Space Station within a specified distance (we are currently using one mile, as the distance) to allow OMV rendezvous and docking with the OTV. The OMV will maneuver the OTV for capture with the station RMS, which will be used to place the OTV in the berthing structure.

The complete OTV turnaround at the Space Station will require four working days and a dedicated IVA crew. EVA will not be required for normal turnaround operations. The total elapsed task time is 34 hours and 30 minutes.

The prelaunch and launch times were derived from the actual timelines established for the Shuttle/Centaur International Solar Polar Mission.

OTV PERIODIC & UNSCHEDULED MAINTENANCE
Shelter Configuration

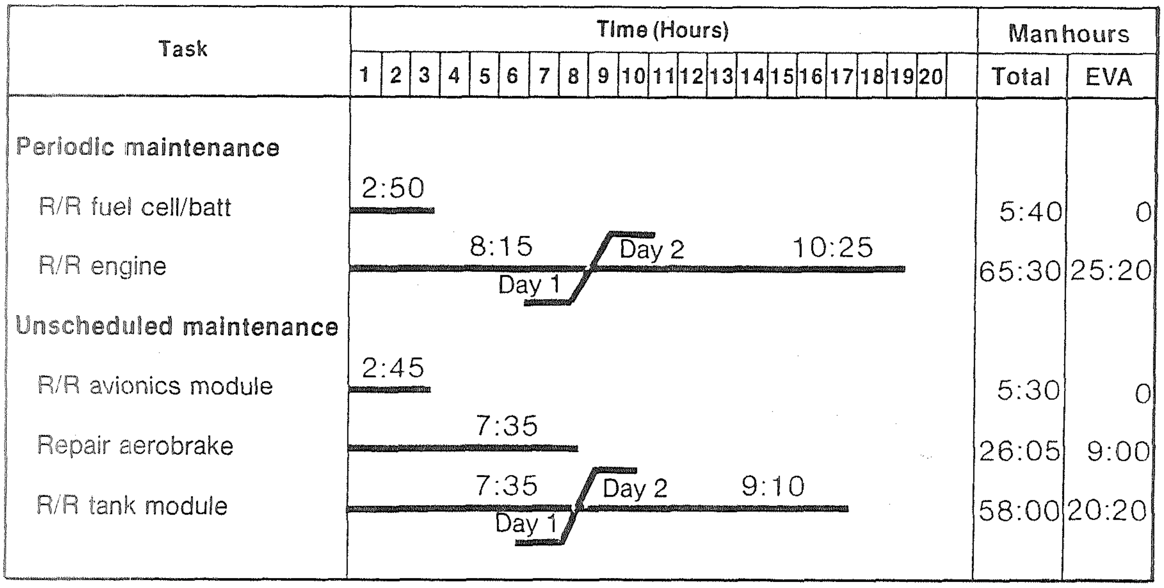


Figure 12

This chart presents the timelines for maintenance tasks that are not considered as part of the normal turnaround cycle, because they do not occur on each and every mission. Periodic maintenance will occur on a timely basis, e.g. engine changeout after ten missions or some predetermined time of operation; fuel cell servicing after five missions, etc. Unscheduled maintenance is performed as a result of equipment failure or damage and occurs generally on a random basis.

Our analysis reveals that engine and tank module changeouts will each require two days to accomplish the task with an EVA involvement. Aerobrake damage repair times are difficult to establish, however, we have determined that one day of EVA operations would provide for minor damage repair. In any case, the aerobrake could be removed and replaced within a one day operation. The fuel cells/batteries and avionic modules can be replaced utilizing IVA remotely controlled equipment and can each be accomplished in less than three hours.

MAN-HOUR COMPARISON FOR 20-MISSION YEAR

Predicted Task Requirements per Year	Shelter		Maint Module	
	Total	EVA	Total	EVA
20 turnarounds	1380:00	—	1513:20	—
4 fuel cell R/R	22:40	—	22:40	—
2 engine R/R	131:00	50:40	36:20	—
5 avionic R/R	27:30	—	16:40	—
1 aerobrake repair	26:05	9:00	26:05	9:00
2 tank R/R	116:00	40:40	116:00	40:40
Total man-hours per year	1703:15	100:20	1731:05	49:40
Percent EVA per year		5.89%		2.87%

Figure 13

A probable 20 mission year was formulated and summarized to provide a means for assessment of crew man-hour requirements per year at the Space Station to support the OTV. The information gained allows for man-hour comparisons between the shelter and maintenance module configurations. The chart also presents the expected amount of EVA envolvement, which amounts to less than 6% of the total Level I maintenance effort.

Preventive maintenance changeouts were selected on the basis of fuel cell replacement every fifth mission and engine replacement after ten missions. Unscheduled maintenance items and frequency of replacements were selected arbitrarily to show avionic, aerobrake and tank module activities throughout the year. Tank module changeout could also represent a configuration change for a manned mission.

COMPARISON OF GROUND-BASED vs SPACE-BASED
Total Task-Times

	Task-Time	Manhours	Average No. of Men per Task	Total No. of Men Required For All Tasks
All ground-based OTV task-times listed	152:45	2534:30	16.6	35
Space-based OTV avg task-times for nominal 20-mission year	38:45	85:10	2.2	4

Figure 14

The accumulative task times that are required to perform a turnaround operation on ground-based and space-based vehicles are presented here with the associated man-hours and average man-loading requirements. A column to show the total number of men required to perform all tasks has been included, which takes into account the peak loading and total skill level requirements. The task times include all tasks required for ground or space-based operations, as an average for the nominal 20 mission year. Note that an additional ground support crew of 35 are also required to support either the ground or space-based operations.

OTV TECHNOLOGY NEEDS

High mass fraction vehicle
- Low vapor pressure/lightweight tanks/meteoroid protection
- Composite structure
- Modular for maintenance

Efficient propellant management
- Thermal control
- Propellant acquisition
- Tank pressurization
- Propellant conditioning
- Propellant transfer
- Mass gauging
- Propellant utilization

High performance engine
- High Isp
- Throttling
- Low inlet pressure/NPSP
- Reusable/space maintainable

Lightweight aerobrake
- Materials
- Heating

Avionics
- Redundant
- Fault tolerant
- Aerobraking GN&C
- Fault detection/isolation
- Rendezvous/docking

Servicing
- Cryogenic propellant transfer, storage & reliquefication
- OTV payload handling & integration
- Maintenance
- Deployment/retrieval

Low-cost propellant delivery

Figure 15

A summary of OTV technology needs identifies the major areas requiring attention.

SUMMARY

- Considered ground- & space-based reusable OTVs
- Determined potential advantages of space-basing
- Defined operations, servicing & maintenance requirements for reusable cryogenic OTVs
- Established space-based OTV technology development areas
- Further study is needed to better define candidate OTV/SS accommodations

Figure 16

D23

A. T. Zachary
Rockwell International/Rocketdyne Division

The remoteness of space requires that future rocket engines operate with integrated health monitoring and control systems that ensure maximum flight reliability and indicate the need for maintenance only when required. In addition, maintenance concepts must be developed that are consistent with the remote and hostile environment of space.

INTRODUCTION

The effective use of space will require the advent of a low-earth orbit space station and creation of such a station has been initiated as directed by the President of the United States. To effectively utilize such a station will ultimately require the development of an Orbit Transfer Vehicle (OTV) for space transportation. The unique characteristics and requirements of a space-based OTV raise a number of major issues particularly those related to vehicle/engine servicing and operations. These issues will, to a major extent, be met by more fully utilizing recent and projected advances in control and diagnostic systems that will provide greater flight reliability and virtually eliminate scheduled maintenance.

CONTROLS AND DIAGNOSTICS

As indicated in table I, advances in computer technology have not been fully utilized in rocket engines. The capability of providing engines with Integrated

Table I. Controls and Diagnostics

- TECHNICAL ISSUES
  - COMPUTER TECHNOLOGY HAS OUTSTRIPPED ROCKET TECHNOLOGY
  - ADVANCED ENGINES REQUIRE MORE PRECISE CONTROL WITH:
    - LESS POWER DEMAND
    - LIGHTER WEIGHT
    - INCREASED RELIABILITY
  - REUSABILITY AND SPACE BASING REQUIRE COMPLETE AND IMPROVED DIAGNOSTICS CAPABILITY
- RECOMMENDATION
  - DEVELOP MICROPROCESSORS AND ADVANCED SOFTWARE FOR ENGINE USE
  - DEVELOP CONTROL VALVES AND SENSORS PROVIDING MULTI-VARIABLE CONTROL
  - DEVELOP IN-FLIGHT DIAGNOSTIC SYSTEMS

Control and Health Monitoring (ICHM) systems will require advancements in microprocessor, software, multivariable controls, and particularly sensors. The ideal functional capabilities of a rocket engine control system will require the type of technology growth illustrated in figure 1. The result will be a system capable of performing the control system functions listed in Table II. The ICHM presently envisioned consists of in-flight and between-flight sensors whose output is processed in a computer with the resulting control system actions, maintenance actions, and maintenance information storage. A simplified concept of the system is shown in figure 2.

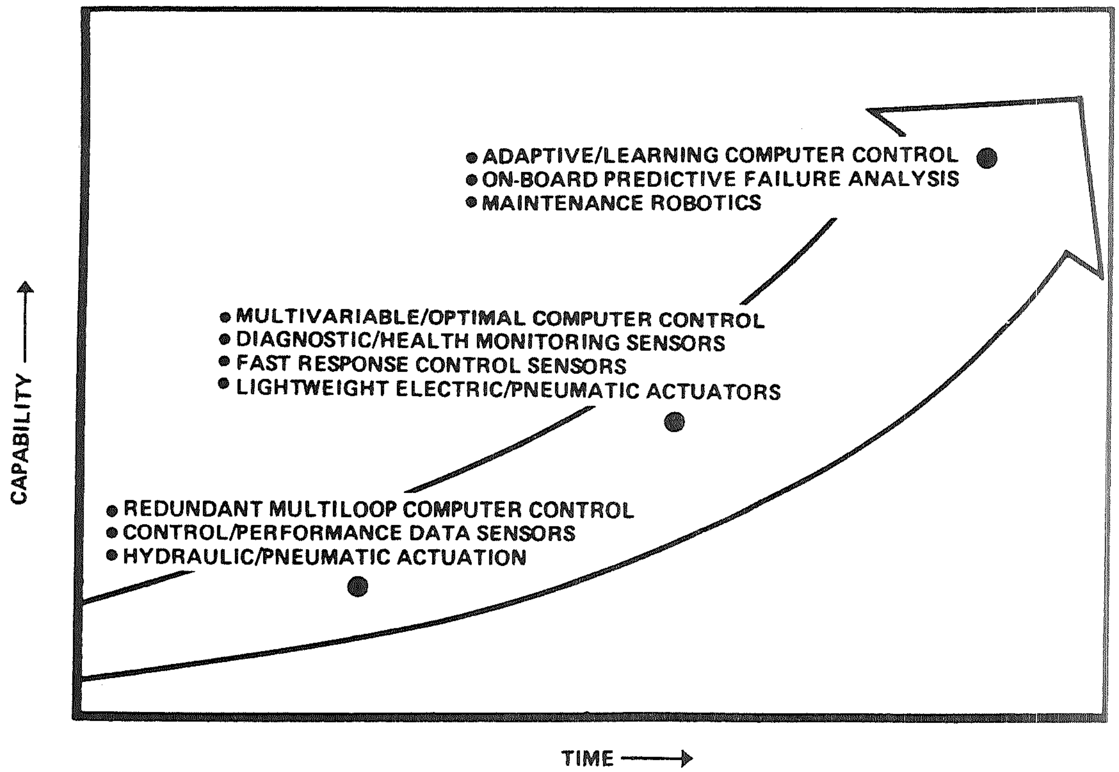


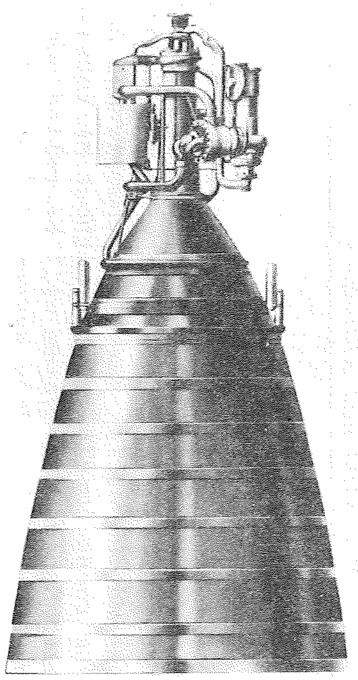
Figure 1. Controls and Diagnostics

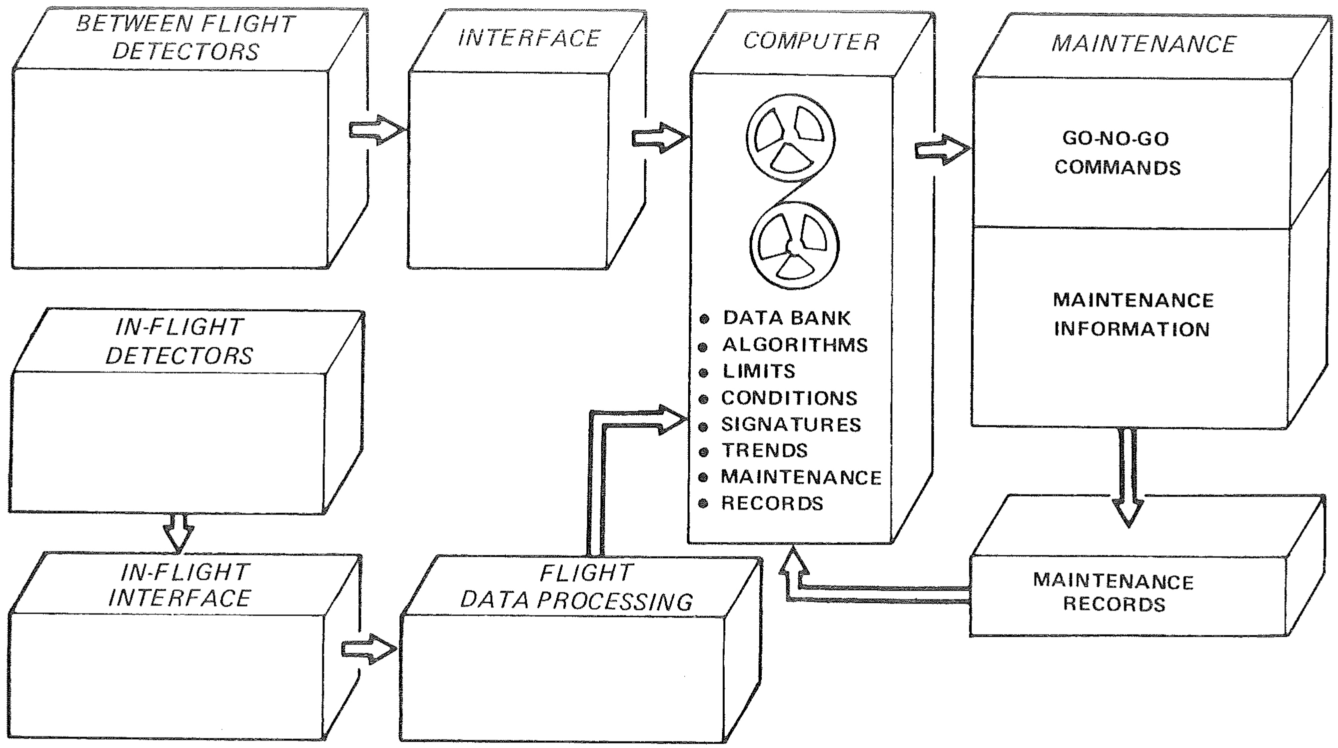
Table II. Projected Control and Diagnostic Functions

• CLOSED-LOOP ENGINE CONTROL
• CRITICAL COMPONENT WEAR DIAGNOSTICS AND LIFE PREDICTION
• BETWEEN FLIGHT SERVICING IDENTIFICATION
• IN-FLIGHT FAULT DETECTION AND AVOIDANCE

MAINTENANCE

With the relative remoteness of the engine and reliability such an important concern, maintenance actions (e.g., inspections, checkouts, and preventive replacement) must be minimized because of the difficulties of working in a vacuum environment as well as the cost of supplying and supporting parts, equipment, and personnel at a space station. To accommodate these considerations without seriously affecting performance, the Integrated Health Monitoring system will be relied upon to specify the timing and extent of maintenance actions. This system will provide a detailed assessment of engine health through the use of advanced sensor, inspection, and fault





ACHIEVED BY USING A BETWEEN-FLIGHT AND/OR IN-FLIGHT CONDITION-MONITORING SYSTEM CONSISTING OF STATE-OF-THE-ART AND/OR NOVEL AUTOMATED DETECTION TECHNOLOGIES AND TAILORED DATA PROCESSING AND COMPUTERS.

Figure 2. Diagnostics for Maintainability Approach

isolation technology and ameliorate the effects of a malfunction and reduced maintenance through control of failure propagation.

With the use of advanced interface concepts that provide high reliability and ease of connect/disconnect by robots or suited astronauts in an extra vehicular activity, the engine would be configured into several modules, each containing one or more related components. Any module, or the entire engine, could be removed and replaced quickly. Upon return of an OTV to the station, routine health monitoring inspections would be undertaken (perhaps automatically by robots). The results would be combined with the data from flight monitoring to isolate any faults or impending failures. The affected module(s) would be removed and replaced with new or repaired modules, which inventoried at the space station. After a functional checkout, the engine would be ready for the next mission. Initially, defective modules would be returned to earth for repair. The modules, because of their light weight and compact size, should be easily accommodated on regularly scheduled shuttle flights. As the space station evolves, a repair shop capable of doing many repairs could develop. The modules would be brought into a pressurized environment so disassembly could utilize the full dexterity of maintenance personnel. Modules would be returned to earth only for complex repairs or overhaul.

CONCLUDING REMARKS

Servicing requirements and space operations will directly impact the design of future rocket engines. Checkout and launch requirements, maintenance philosophy, manned involvement, and space versus ground rehabilitation of systems or components are key considerations in maintaining a rocket engine system in space. The nerve center of an engine is the control system and with continuing advances in control and diagnostic technology particularly in the areas of computers and sensors, it will make space-based operations practical.

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Grumman Aerospace Corporation

People can perform work in the airless environment of space by enclosing their bodies in space suites (Extra Vehicular Activity - EVA) and doing the work themselves or, by remaining inside a pressurized volume (Intra Vehicular Activity - IVA) and controlling external work which is performed by mechanical arms.

The latest state of the art mechanical arms utilize Bilateral Force Reflection (BFR) to provide operators with some "feeling" of what is happening at the worksite. BFR is a requirement for task efficiency, providing an order of magnitude improvement in task time when compared with all other currently known control modes.

IVA can be conducted from an Orbiter cabin or Space Station control room utilizing SAM - Surrogate Astronaut Machine - to perform tasks. SAM has 2 dextrous arms, a rigid arm (stabilizer) to maintain (and alter) his location relative to a worksite, illumination, a TV camera and a small tool chest. SAM is transported to and from worksites by the Shuttle Remote Manipulator System (RMS).

Another method of conducting IVA is to transport "Man-in-a-can" to a worksite with the dextrous arms attached to the can. A closed cabin cherry picker can perform the tasks of transporting materials as well as the dextrous tasks of installation and removal.

EVA can be conducted in a variety of ways. We believe that there will not be extensive use of a Manned Maneuvering Unit around a Space Station for work activities because of the risk of human error and the potential damage that may result to the Space Station. Rather, we believe that most EVA work will be conducted on the RMS/MFR (Manipulator Foot Restraint, formerly known as the Open Cherry Picker - OCP). This system offers the advantage of enhancing Space Station safety by using collision avoidance software.

Earth based tests of space tasks have been conducted on 6 degree of freedom motion simulators and underwater neutral buoyancy facilities. A one to one correspondence has been demonstrated between the task times required on these two types of facilities. However, no correspondence has been demonstrated yet between these earth based test facilities and the actual task time required in the zero gravity of space.

Two timelines are shown for performing the same task in two different ways EVA and IVA. The EVA task requires more time, principally for preparation (although this MOTV scenario assumes that pre-breathing is <u>not</u> required), than the IVA task. The EVA work is more efficient (when current state of art dextrous arms are used for IVA), but the total job time is significantly less for IVA because it is easier to get started and finish.

Two charts compare EVA with IVA on several different measures. For most of these, IVA offers a significant advantage. Our preference is for IVA utilizing man-in-a-can, which brings direct binocular vision (and depth perception) to the worksite. The advantages of human adaptability and versatility can be maximized by having man in close proximity to the work he is performing.

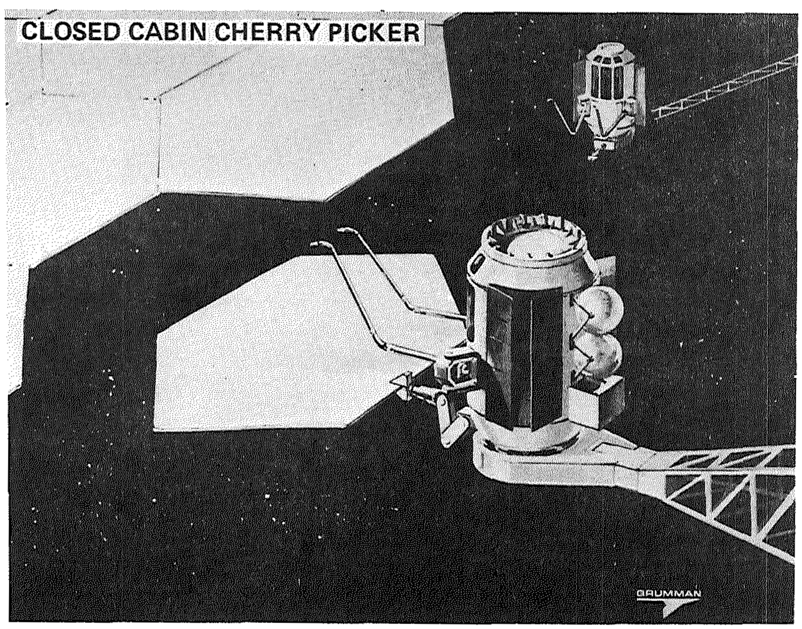


Figure 1

GRUMMAN/MARTIN/NASA – JSC WORK RESTRAINT SYSTEM
THERMAL PROTECTION SYSTEM REPAIR

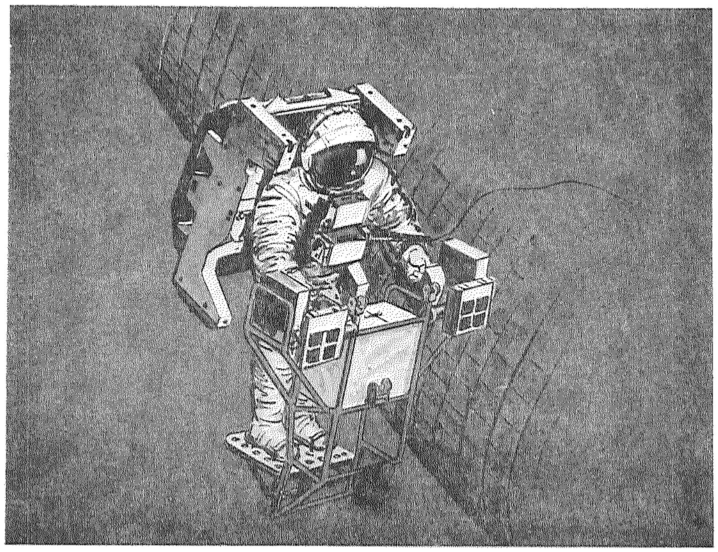


Figure 2

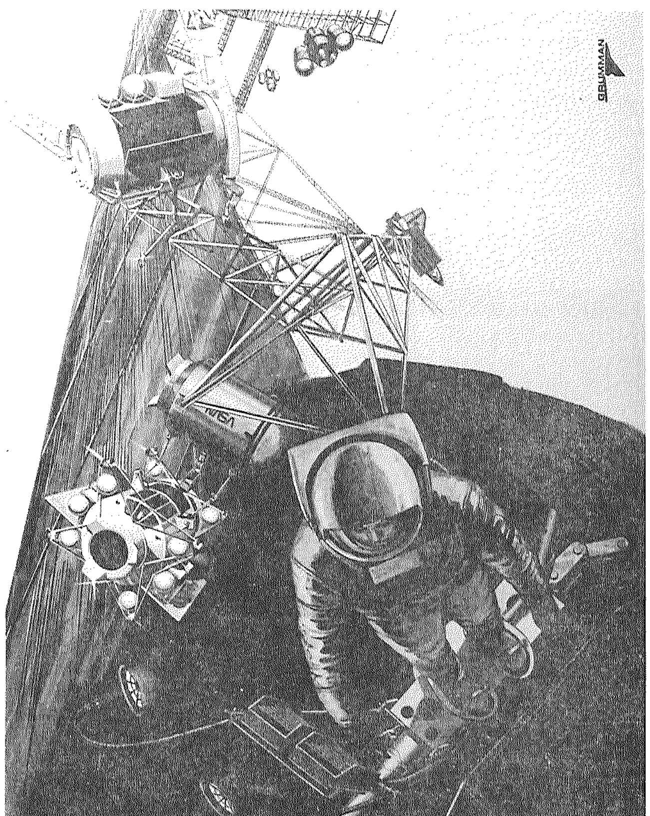


Figure 3

MANIPULATOR TECHNOLOGY LABORATORY

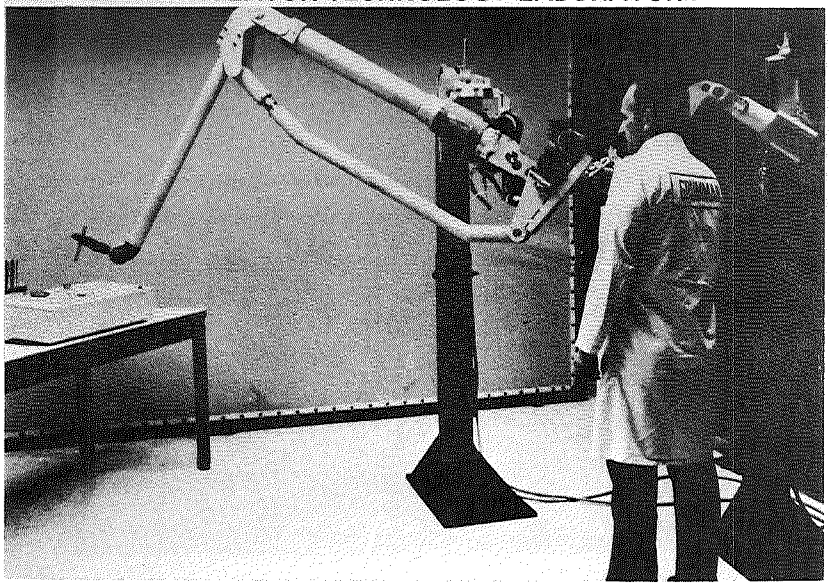


Figure 4

TELEPRESENCE: DEXTEROUS MANIPULATOR AND RMS PERFORM MODULE EXCHANGE ON MMS ATTACHED TO HPA

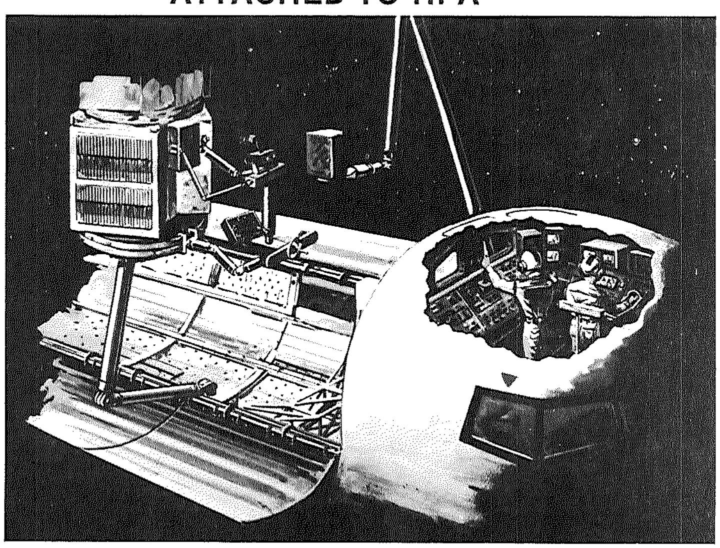


Figure 5

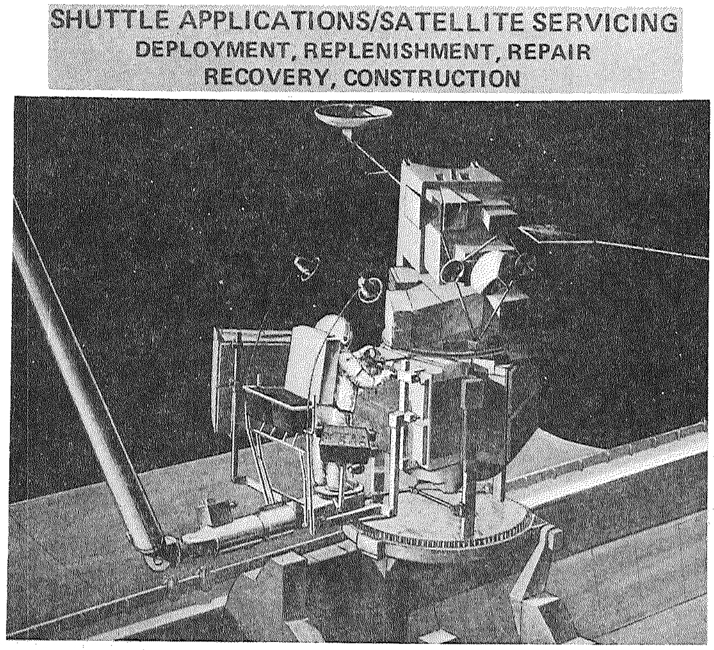


Figure 6

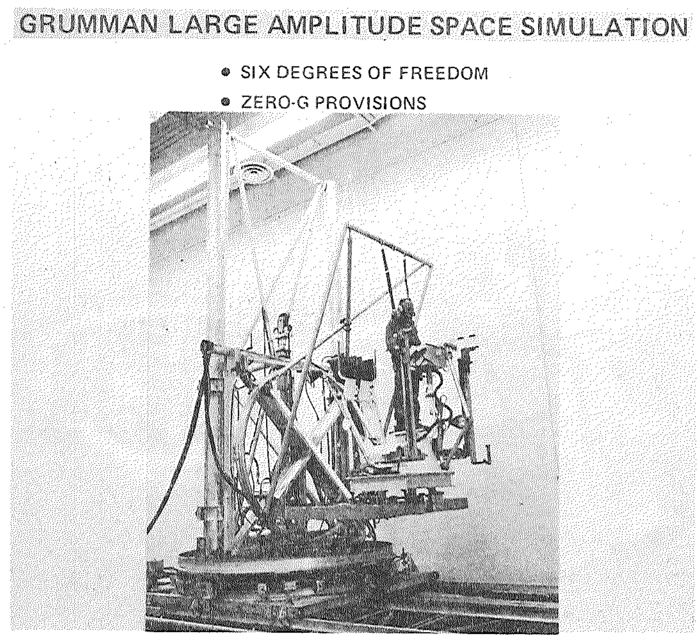
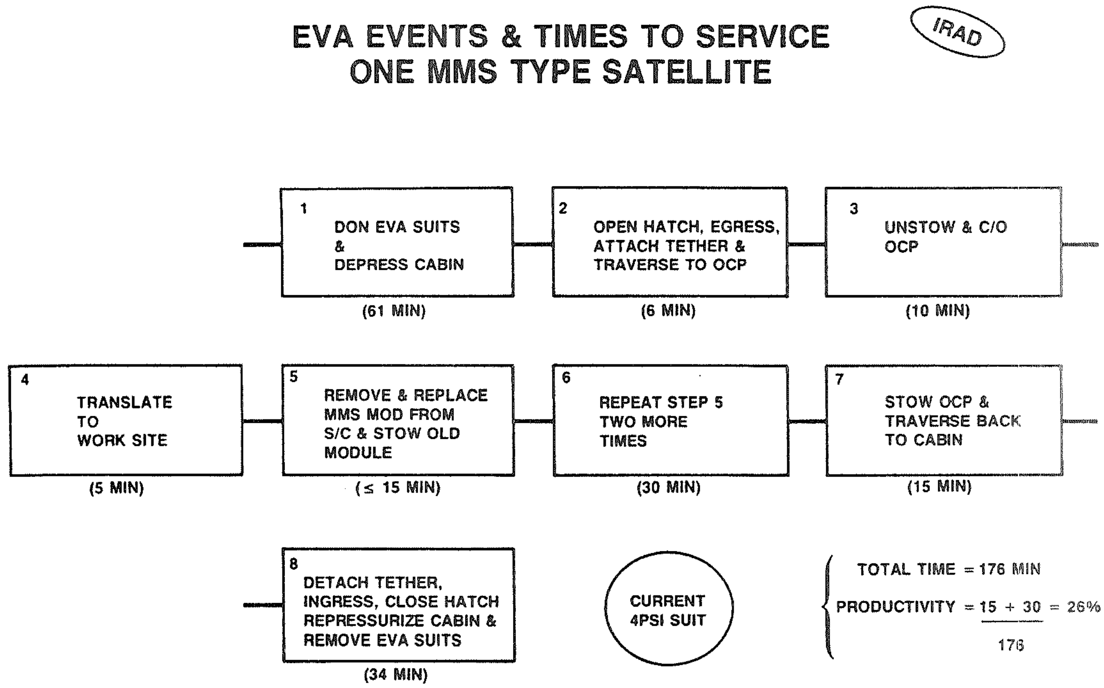


Figure 7

EVA EVENTS & TIMES TO SERVICE ONE MMS TYPE SATELLITE



IVA EVENTS & TIMES TO SERVICE ONE MMS TYPE SATELLITE

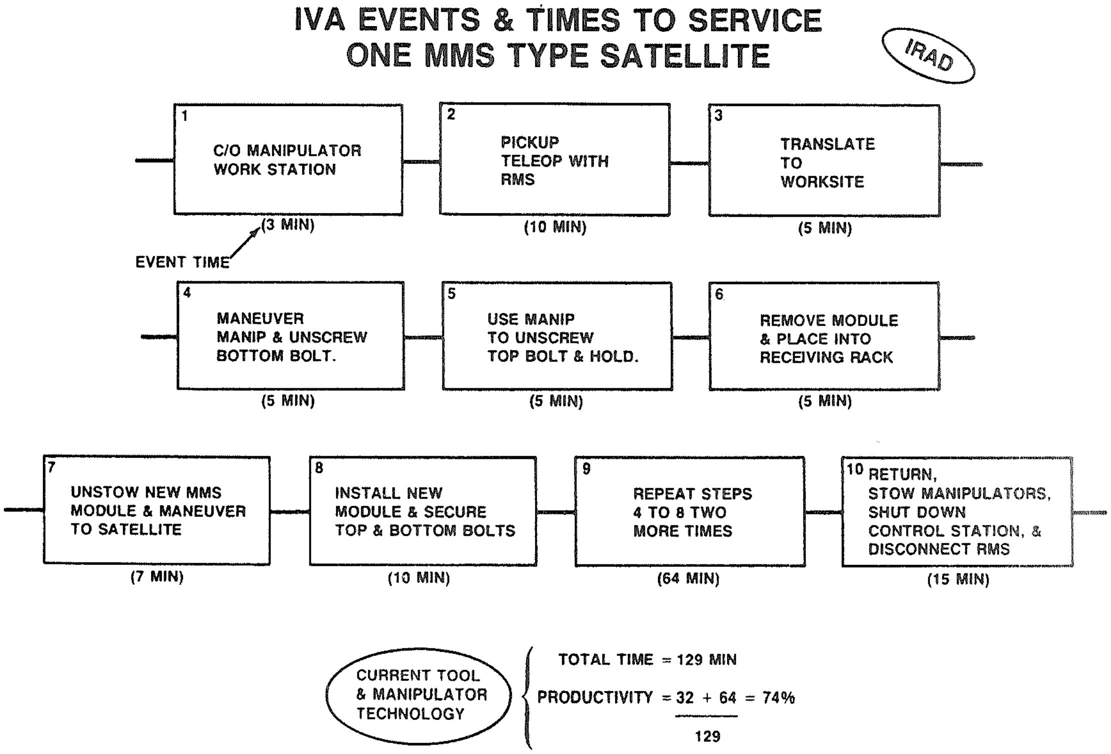


Figure 8

EVA vs IVA: SOME COMPARISONS

ISSUE	EVA	IVA	ADVANTAGE
• NUMBER OF WORKERS:			
– OUTSIDE VEHICLE	2 (BUDDY SYSTEM (REQ'D TO DATE)	0	
– INSIDE VEHICLE	1 (FOR RMS)	2 (RMS & TELEOP) OR 1 (MAN IN CAN)	
– TOTAL WORKERS	3	2 OR 1	IVA
• WORKER FATIGUE	STRENUOUS, NO RELIEF	ALLEVIATE BY SWITCHING RMS & TELEOP	IVA
• WORK CONTINUITY	STOP FOR SOUTH ATLANTIC ANOMALY (EVERY 1 1/2 HOURS)	CONTINUOUS	IVA
• TASK EFFICIENCY (SUBJECTIVE ESTIMATES W/O DETAILED STUDY)	MOST EFFICIENT	CURRENT SOA: 2X EVA FUTURE SOA: 1X EVA	EVA NEITHER

EVA vs IVA: COMPARISONS (CONT'D)

ISSUE	EVA	IVA	ADVANTAGE
• JOB EFFICIENCY:			
– GENERAL	LONG PREP TIME WITHOUT 4 HOUR PREBREATHING	RAPID CHECKOUT & PREPARATION, EASY LUNCH BREAK	TBD
– SERVICE MMS	2.9 HOURS	2.1 HOURS	IVA WITH CURRENT TECH.
• SAFETY			
– PEOPLE	UNHEALTHY, HAZARDOUS	SAFE	IVA
– FACILITIES & EQUIP	RISKY	SAFEST-ADV TECH INCREASES SAFETY	IVA
• DEVELOPMENT COSTS	~$150M FOR NEW 8PSI SUIT	$ 8M - MANIPULATORS ~$130M - MAN IN CAN	IVA STANDOFF
• RECURRING COSTS	> $10M/YEAR FOR 160 EVAs	TBD, BUT SMALL	IVA

IVA APPEARS TO OFFER SUBSTANTIAL ADVANTAGES

Figure 9

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