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PHASE I FINAL REPORT

VOLUME IB - PART I
STUDY RESULTS

SYSTEM TECHNOLOGY ANALYSIS OF AEROASSISTED ORBITAL TRANSFER VEHICLES: MODERATE LIFT/DRAG (0.75-1.5)

AUGUST 1985

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FORWARD

This final report of the "System Technology Analysis of Aeroassisted Orbital Transfer Vehicles: Moderate Lift/Drag (0.75-1.5)" was prepared by the General Electric Company, Space Systems Division for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center (MSFC) in accordance with Contract NAS8-35096. The General Electric Company, Space Systems Division was supported by the Grumman Aerospace Corporation as a subcontractor during the conduct of this study. This study was conducted under the direction of the NASA Study Manager, Mr. Robert E. Austin, during the period from October 1982 through June 1985.

The first phase of this program focused on a ground based AOTV and was completed in September 1983. The second phase was directed towards a space based AOTV and the cryofueled propulsion subsystem-configuration interactions and was completed in March of 1985. The second phase was jointly sponsored by NASA-MSFC and the NASA Lewis Research Center (LeRC). Dr. Larry Cooper was the LeRC study manager.

This final report is organized into the following three documents:

Volume IA	Executive Summary - Parts I & II
Volume IB	Study Results - Parts I & II
Volume II	Supporting Research and Technology Report
Volume III	Cost and Work Breakdown Structure/ Dictionary

Part I of these volumes covers Phase 1 results, while Part II covers Phase 2 results.

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VOLUME I - PART B

1.0 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 sensitivites 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 screenings 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).

2.0 SYSTEM ANALYSIS

2.1 Flight Mechanics

2.1.1 General Mission Model

The generalized mission model addressed is summarized in Figure 2.1-1. The initial ground based AOTV's will be deployed from a 150 nmi circular orbit, launched from ETR at an orbital inclination of 28.5^o. Eventually, the space based AOTV's will be in a 200 nmi circular orbit at 28.5^o inclination. Launch vehicles considered include the standard STS, an improved STS, the aft cargo compartment (ACC) and the shuttle derived cargo vehicle.

Operating scenarios were established for the several reference missions and ΔV budgets determined for use in the performance compuations. Effective use of aeromaneuver for return from Molniya is not possible due to large ΔV required for apsis rotation of the Molniya orbit.

Initial payload capbility has been evaluated for a baseline of delivery to GEO, six hour polar, and Molniya (12 hours x 63.4^o 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).

Table 2.1-1. **AOTV GENERAL MISSION MODEL DEFINITION**

LEO - LOW EARTH INITIAL PARKING ORBIT

- GROUND BASED 150 NM CIRCULAR, $i = 28.5°$
- SPACE BASED 220 NM CIRCULAR, $i = 28.5°$

HEO - HIGHER EARTH ORBIT DESTINATIONS

BASELINE {
- GEO - GEOSYNCHRONOUS EQUATORIAL ORBIT, $i = 0°$
- MOLNIYA - 12 HR PERIOD, $i = 63.4°$, $h_a = 21500$ NM, $h_p = 400$ NM

- SIX HOUR POLAR - ETR LAUNCH ~ 5600 NM CIRCULAR
- 5X GEO

TIME CONSTRAINED MISSION - SINGLE ORBIT DEPLOYMENT, LOITER, AND RECOVERY

2.1.2 Propulsive ΔV Budgets

The characteristic velocity for the baseline missions were determined parametrically with Shuttle parking orbit inclination and transfer method, i.e., two-impulse, three impulse, or transfer through infinity. It was assumed that all of the inclination change for the outbound leg was performed propulsively. The inbound leg was treated parametrically with propulsive inclination change and apses rotation.

The impulsive transfer from some initial parking orbit to a desired final orbit can be performed in several ways; a one, two, or three-impulse transfer, and a transfer through infinity. One-impulse transfers are not common since the two orbits must intersect or be tangent. The more common type of transfer is the two-impulse, of which the most familiar (for coplanar transfers) is the Hohmann transfer. Three impulse, time-open transfers are rare but have been shown to be optimal for transfers to highly ellliptical orbits where large inclination changes are required (16). The transfer through infinity assumes that the initial impulse places the vehicle on an asymptotic escape orbit from which a small impulse at a large radius (approaching infinity) can be made to complete the transfer. The second impulse is small due to the small orbital velocity at a large radius. While constraining the transfer to finite radii the optimality of either two or three-impulse transfers are dictated by required plane change and the initial and final orbit apogee and perigee radii.

The equatorial, circular geosynchronous mission is straight forward in that the logical choice for the shuttle parking orbit is 28.5 degrees so as to minimize the required plane change. It can be shown from Gobetz and Doll (19) that the transfer can be made optimally using a two-impulse transfer. The first impulse of 7985 ft/sec inserts the vehicle into a transfer orbit with perigee at 150 nm (Shuttle parking orbit altitude) and apogee at 19,365 nm (circular geosynchronous altitude). The second impulse of 6009 ft/sec, approximately 5.1 hours later, circularizes the vehicle at GEO and changes the orbital inclination from 28.5 degrees to equatorial. The return leg consists of a transfer to an orbit with the perigee within the atmosphere (60 nm altitude) and requires an impulse from 5400 ft/sec if no propulsive inclination change is needed to 6420 ft/sec if all inclination change is performed propulsively. Upon exit from the atmosphere following aerodynamic braking and maneuvering an additional impulse of approximately 200 ft/sec is required to circularize in low earth orbit. Another 300 ft/sec is budgeted for LEO phasing maneuvers. The above budget is summarized in Figure 2.1-2.

FIGURE 2.1-2 AOTV INITIAL ΔV BUDGET - TYPICAL

L/D = 1.5 GEO DELIVERY & RETURN, i = 28.5°

ΔV_1	TRANSFER TO h_a	7985
ΔV_2	CIRCULARIZATION & INCLINATION CHANGE @ GEO	6009
ΔV_3	TRANSFER TO h_p = 60 NM ONE PASS CAPTURE BOUND	5414
ΔV_4	CIRCULARIZATION AT LEO	200
ΔV_5	Σ LEO PHASING AND ORBITAL MANEUVER	295
ΔV_T		19903

The six-hour polar mission, due to the large inclination change, requires a considerable characteristic velocity. The outbound transfer from LEO made with either two or three impulse maneuvers, depending on necessary plane change, is shown in Figure 2.1-3. Basically for Shuttle inclinations greater than 40° the transfer can be optimally with only two impulses, for inclinations less than 45° the three-impulse transfer is optimal. For transfer from a 28.5° shuttle orbit the outbound delta V is approximately 17,300 ft/sec. The return from the six-hour orbit to entry is shown in Figure 2.1-4 as a function of plane change performed propulsively. As much as 13,000 ft/sec impulse may be required if 60° of plane change is needed. The total outbound and inbound delta V is shown in Figure 2.1-5 as a function of both inbound plane change (that not accomplished aerodynamically) and Shuttle orbit inclination. As can be seen a transfer from a 30° shuttle orbit and return to a 30° plane (60° plane change would require 30,700 ft/sec total delta V. In addition to these impulses another 300 ft/sec for phasing and 200 ft/sec for LEO circularization following atmospheric exit should be budgeted.

The Molniya mission provided additional complexity in that the transfer is no longer circular to circular but circular to elliptical which requires placing (or rotating) the line of apses to the desired position. The Molniya orbit is defined to have a period of 12 hours, inclination of 63.4° and perigee altitude of 400 nm. The apogee is usually constrained to be located over the northern hemisphere implying that the argument of perigee will lie between 180° and 270°. In performing a non-coplanar circular-to-elliptical two-impulse transfer the transfe angle should not exceed 180° for time-open transfers. The transfer eliipse is usally entered just before its perigee, and perigee passage on this ellipse occurs from 0° to 90° before perigee of the terminal ellipse. In addition the final orbit is always entered near a node. A schematic of a typical transfer from LEO to Molniya is shown in Figure 2.1-6. As would be expected the delta V increases with inclination change. Also since the orbital velocity at the ascending node of the transfer and final orbit increases as the argument of perigee increases (from 180° to 270° the delta V will also increase. These trends are illustrated in Figure 2.1-7 which shows the outbound delta V as a function of shuttle orbit inclination for two arguments of perigee; 220° (apogee 40° from ascending node) and 270° (apogee 90° from ascending node). The solid lines indicate a two-impulse transfer while the dashed lines indicate three-impulse maneuvers. As can be seen a delta V ranging from 8400 ft/sec to 15,300 ft/sec may be required for the outbound leg. The return or inbound leg may again require some or all of the inclination change and some rotation of the line of apses (so that atmospheric entry can be made near a node - thereby maximizing

FIGURE 2.1-3 DELTA V REQUIREMENTS FOR ASCENT TRANSFER FROM LOW EARTH ORBIT

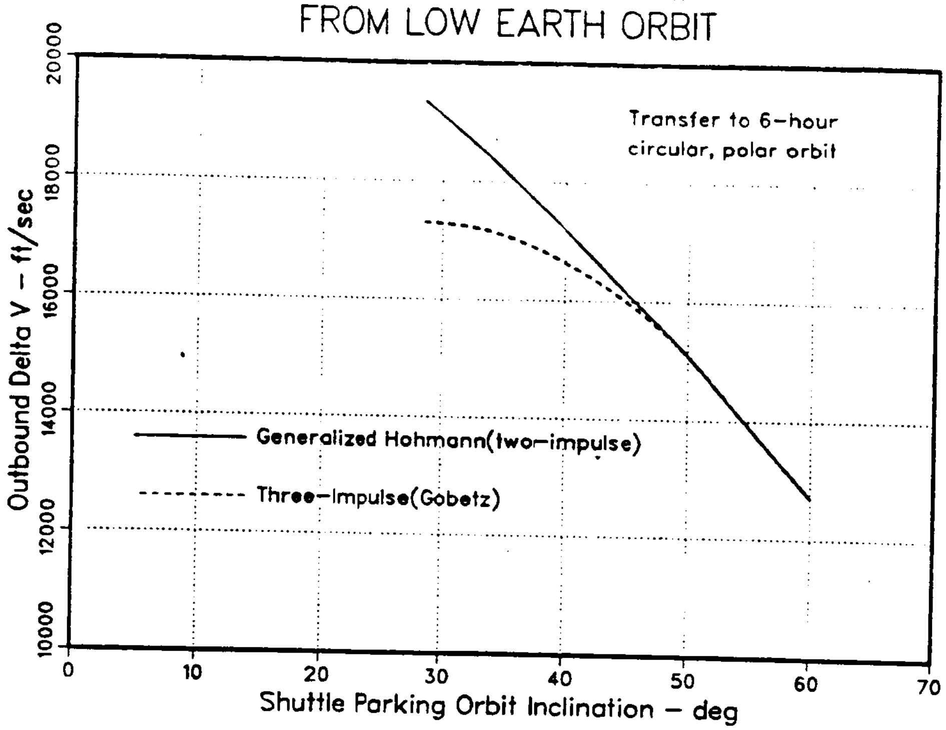


FIGURE 2.1-4. DELTA V REQUIREMENTS FOR RETURN TRANSFER TO LOW EARTH ORBIT

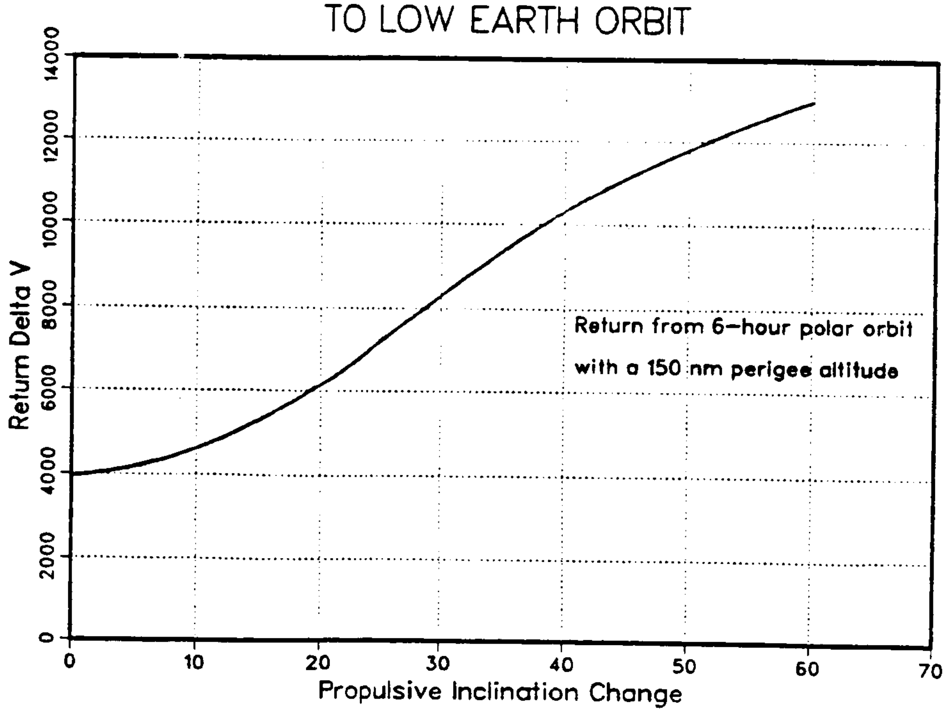
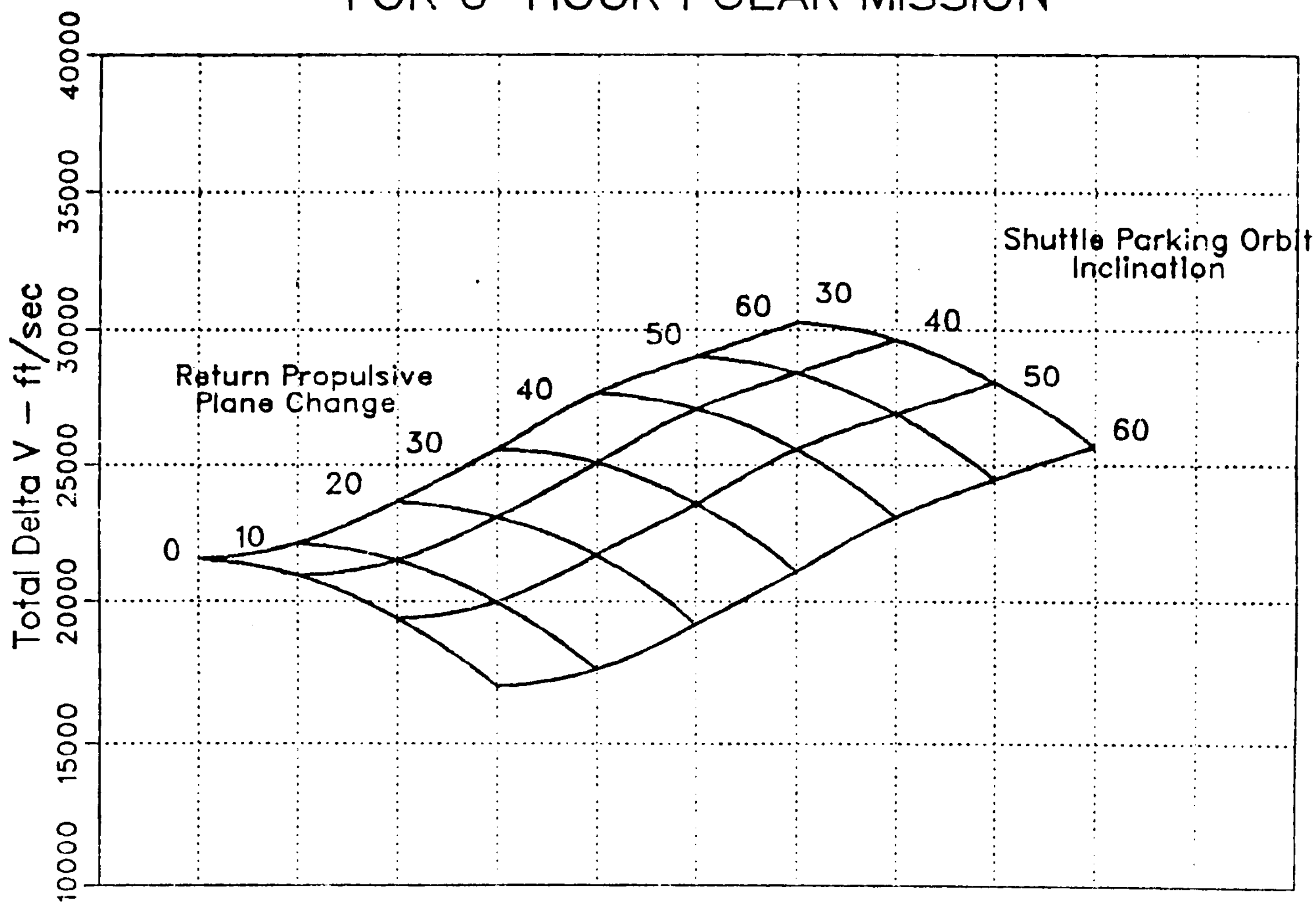


FIGURE 2.1-5 DELTA V REQUIREMENTS
FOR 6–HOUR POLAR MISSION



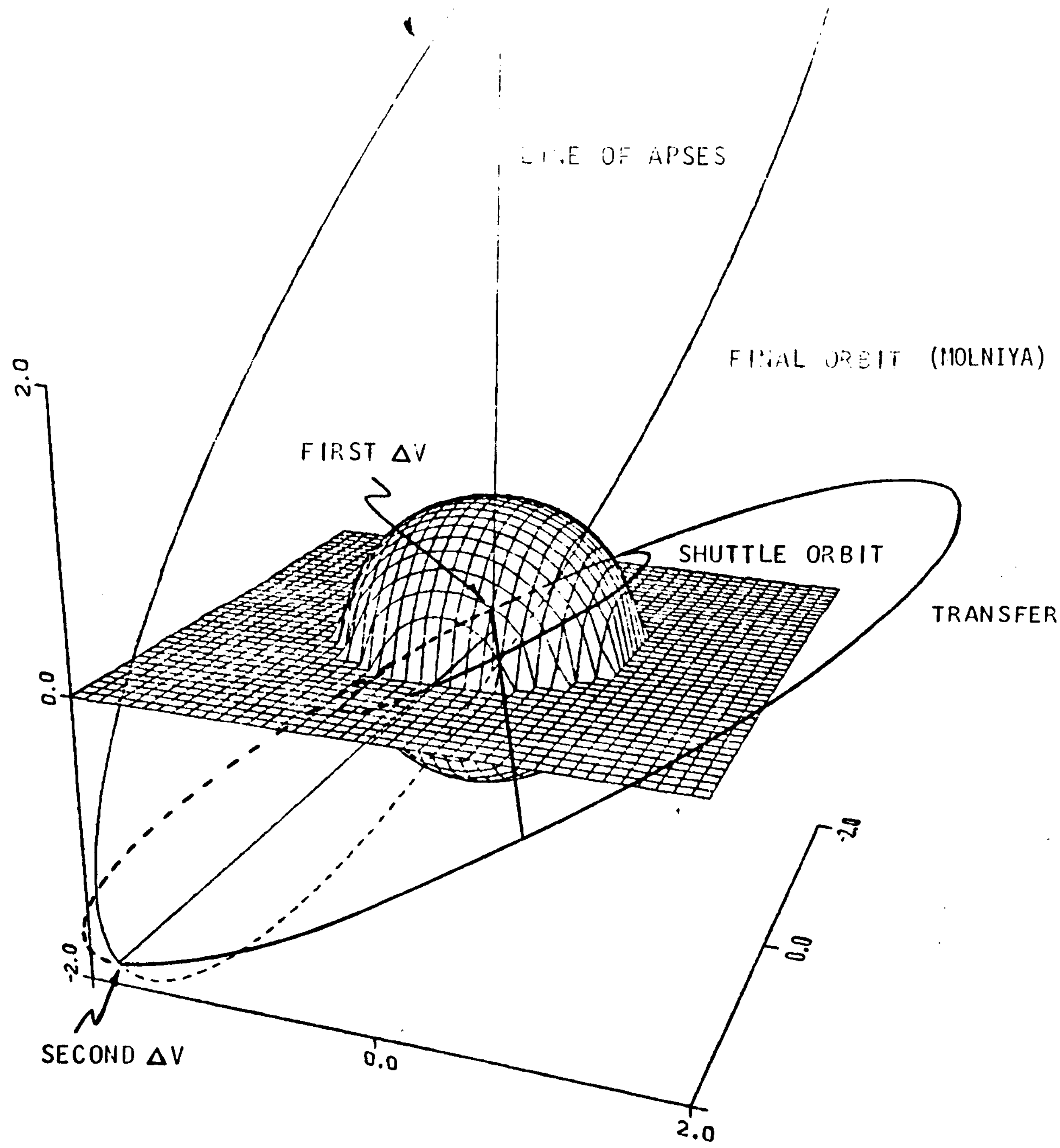

LINE OF APSES
FINAL ORBIT (MOLNIYA)
FIRST ΔV
SHUTTLE ORBIT
TRANSFER ELLIPSE
ASCENDING NODE
SECOND ΔV
2.0
0.0
-2.0
0.0
2.0
-2.0
0.0

FIGURE 2.1-6

plane change via aerodynamic lift). A return from Molniya with a 220° argument of perigee would require approximately 35° of apses rotation to put entry at or near a node. The inbound delta V as a function of inclination change and apses rotation is shown in Figure 2.1-8 and indicates that between 8500 and 9500 ft/sec may be required with a 35° apses rotation.

2.1.3 Aerodynamic Plane Change

Earlier studies (16) reported results for plane change assuming that the flight is performed following entry at the overshoot bound (minimum entry path angle to ensure aerodynamic capture) and using a constant drag control law (17) used extensively in aerocapture studies. However it is demonstrated below that operation at the overshoot bound and use of the constant drag control law is not necessarily compatible with maximizing aerodynamic turning for plane change.

The hypersonic L/D varies significantly with altitude, due to the viscous drag effects, but has been treated as a constant here since it has been demonstrated (6,22) that inclusion of the high altitude effects has a trivial effect upon the plane change obtained. The AOTV mission ends at a Mach number of about 25; therefore, the usual low hypersonic and supersonic variation of the aerodynamic characteristics has no effect here either.

A series of computations were performed for a representative AOTV assumed to weigh 10,000 pounds, with a lift to drag ratio (L/D) of 1.0 and a $W/C_L A$ of 68 psf. Two methods were used to determine the maximum plane change; first, entry is made at a path angle greater than or equal to the overshoot bound. The lift vector is modulated so as to maintain a reference drag that has been pre-selected to correspond to that deceleration sensed at the minimum altitude. The vehicle is commanded to rotate the lift vector full up at the proper time so that atmospheric exit is made at the optimum velocity (optimum in terms of minimal delta velocity for orbit circularization). In an alternate approach, the vehicle flys a constant bank angle throughout the flight and entry is made at the path angle which will result in an atmospheric exit at the optimum velocity. In addition to plane change reference values of hypersonic convective heat transfer (normalized to a one foot nose radius) is given for both methods.

FIGURE 2.1-7. DELTA V REQUIREMENTS FOR ASCENT TRANSFER FROM LOW EARTH ORBIT TO MOLYNIYA

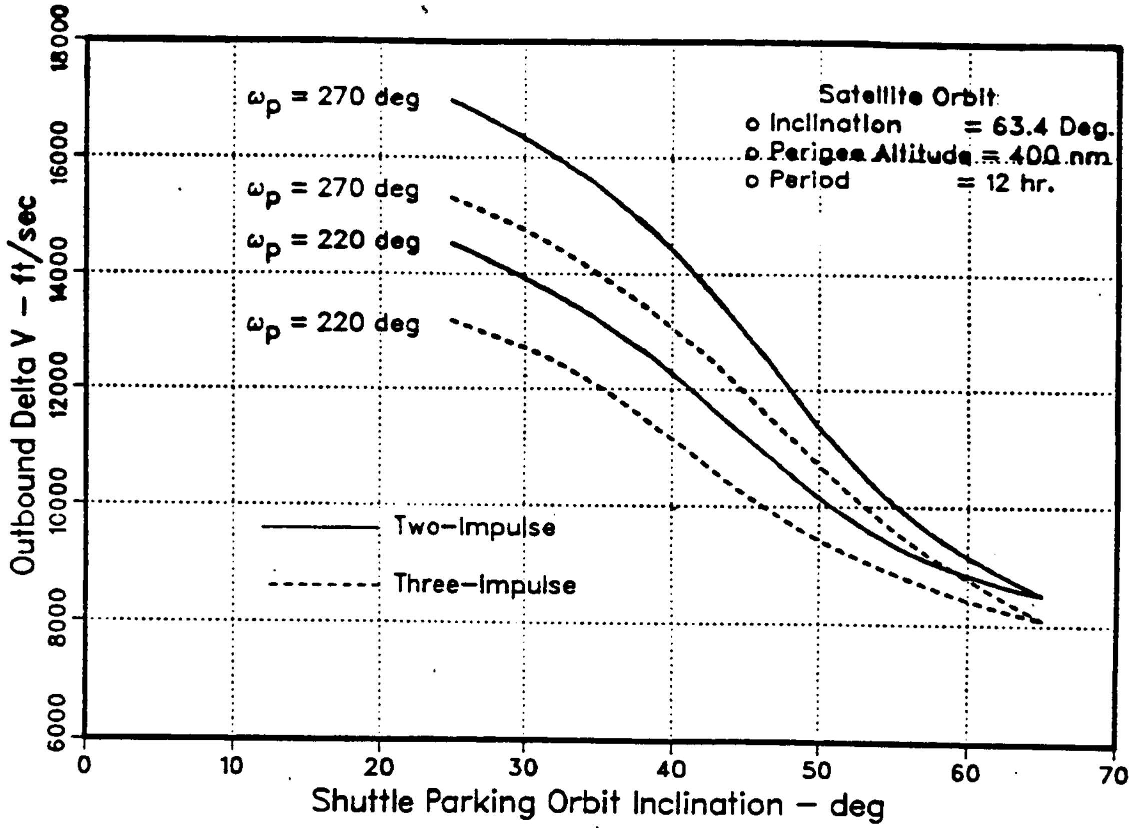
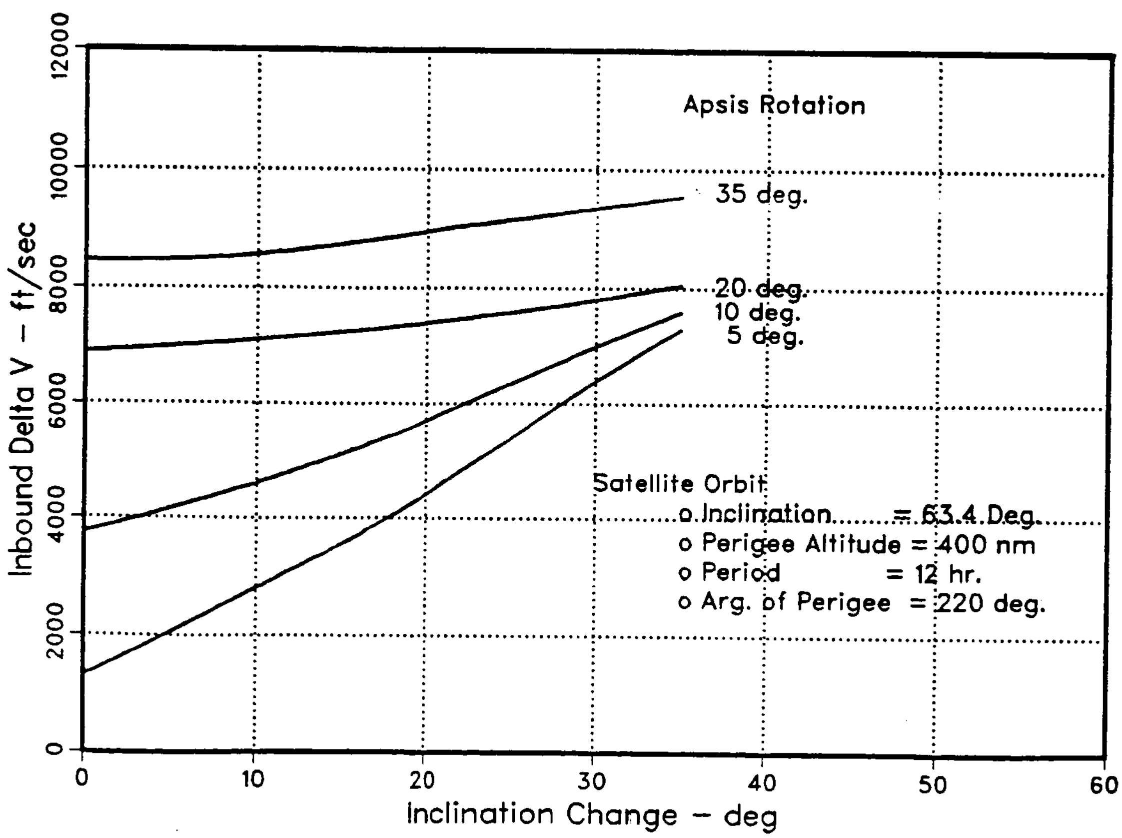


FIGURE 2.1-8. DELTA V REQUIREMENTS FOR RETURN TRANSFER FROM MOLNIYA



The results for flight using the constant reference drag control law are illustrated in Figure 2.1-9. As can be seen a 33% increase in plane change capability over operation at the overshoot bound can be realized (from 8.5^o at the overshoot bound to a maximum of 11.5^o at an entry path angle of 4.8^o and reference drag of 1.0 G's). The stagnation point heat flux increases from 170 BTU/sq.ft.sec to 205 BTU/sq.ft.sec for a one foot nose radius. Figure 2.1-10 shows results for flight at a constant lift vector bank angle. It can be seen that the plane change is maximized at a 90^o bank angle (90^o corresponds to all lift vectored normal to the flight path plane and 0^o indicates full lift up). In essence when flying at a constant 90^o bank angle the vehicle is operating similar to a full aerobraking device, depleting the proper amount of velocity before centrifugal acceleration pulls the vehicle out of the atmosphere, while using all available lift to turn. This maximum plane change of 14.6^o is approximately 70% greater than the plane change at the overshoot bound using the reference drag control law. The normalized stagnation point heat flux is approximately 274 BTU/sq.ft.sec for the constant 90^o bank angle maneuver.

2.1.4 Typical Trajectories and Performance

Time histories of stagnation point heat flux, free stream Reynolds number, and dynamic pressure are shown in Figures 2.1-11, 12 and 13, respectively, for the overshoot bound/ reference drag flight, the maximum plane change/reference drag flight, and the maximum plane change/constant bank angle (90^o) flight trajectories. The primary point to be derived from these figures is the fact that although the stagnation point heat flux is higher for the maximum plane change trajectories the soak or heating time tends to be significantly shorter. Therefore, these lower integrated heat loads although yielding higher skin temperatures may require less thermal protection. The respective velocity, altitude, and Mach number histories are illustrated in Figures 2.1-14, 15 and 16.

EFFECT OF STEERING SCHEME ON AERODYNAMIC PLANE CHANGE CAPABILITY

FIGURE 2.1-9

$L/D = 1.04$ $\quad W/C_L A = 68$ PSF

D_{REF} CONTROL LAW

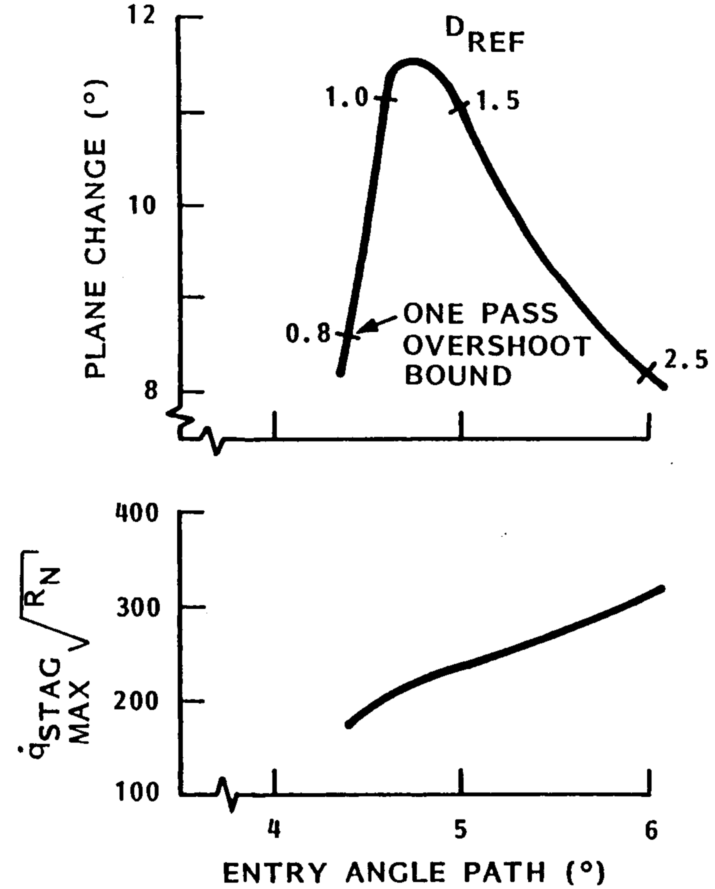


FIGURE 2.1-10

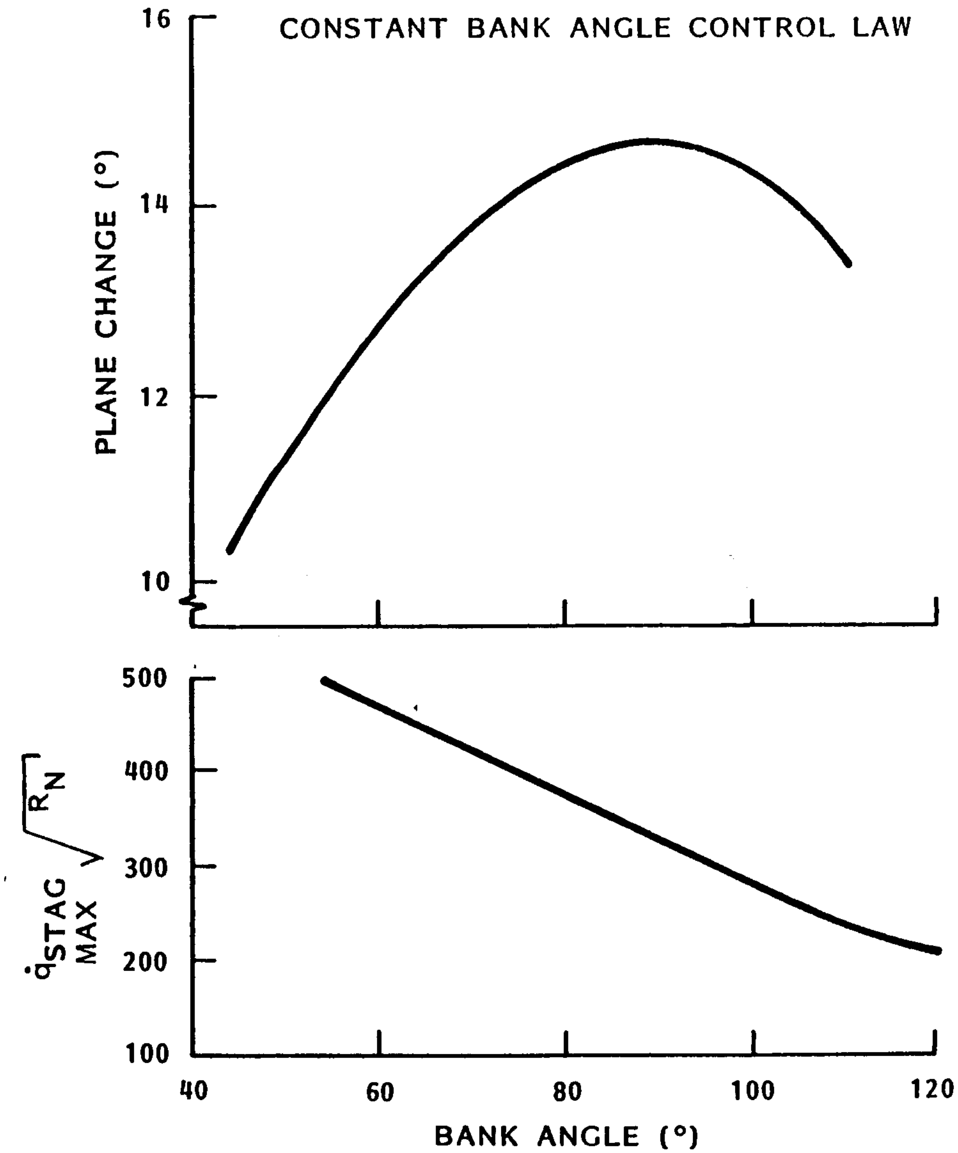


FIGURE 2.1-11 EFFECT OF STEERING SCHEME ON LAMINAR HEAT RATE

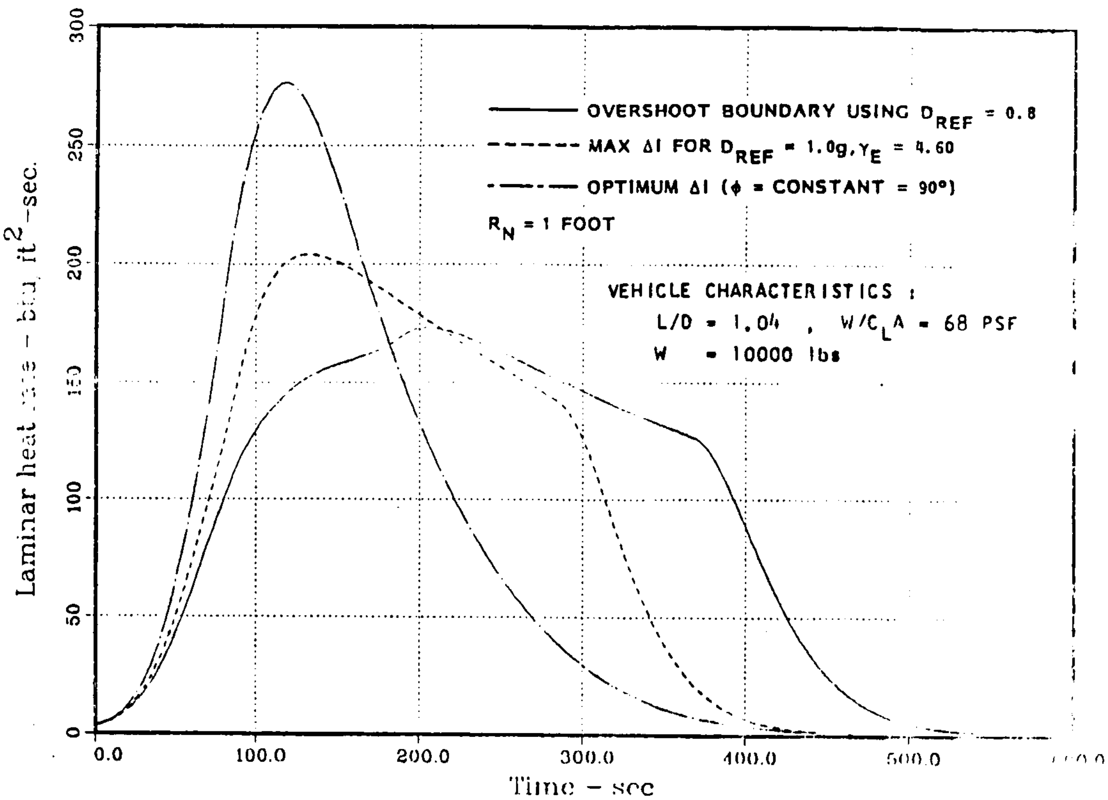


FIGURE 2.1-12 EFFECT OF STEERING SCHEME ON REYNOLDS NUMBER

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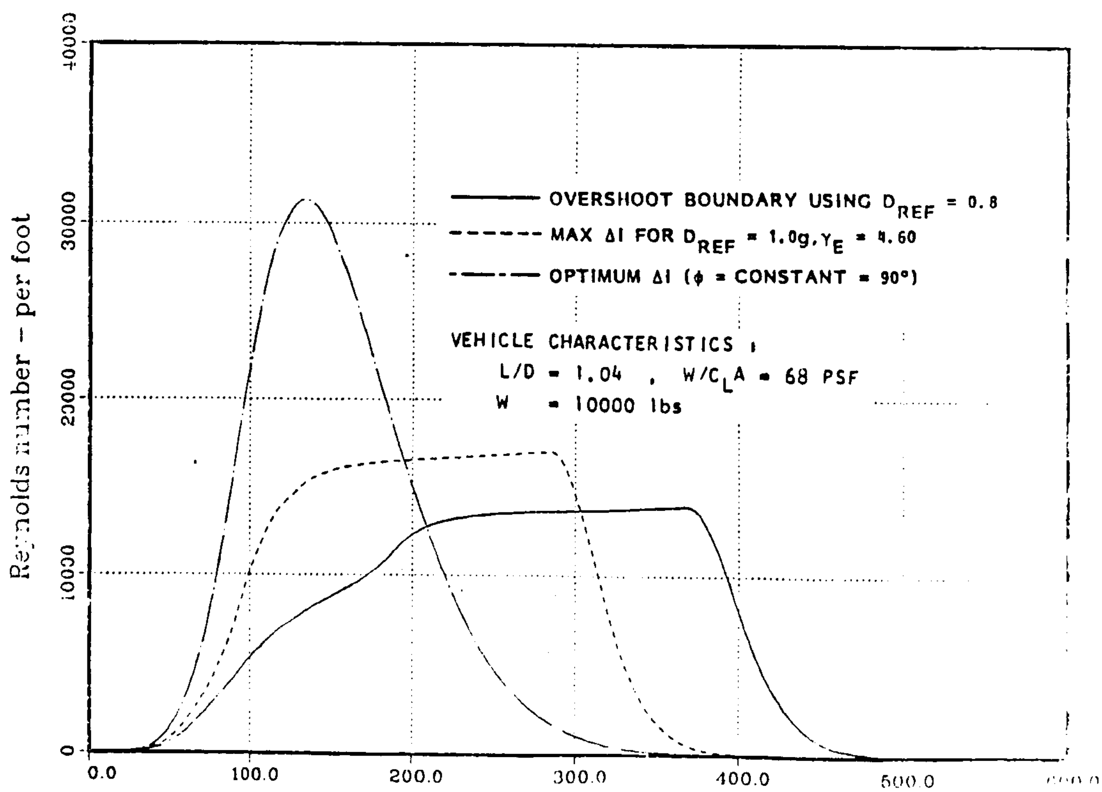


FIGURE 2.1-13 EFFECT OF STEERING SCHEME ON DYNAMIC PRESSURE

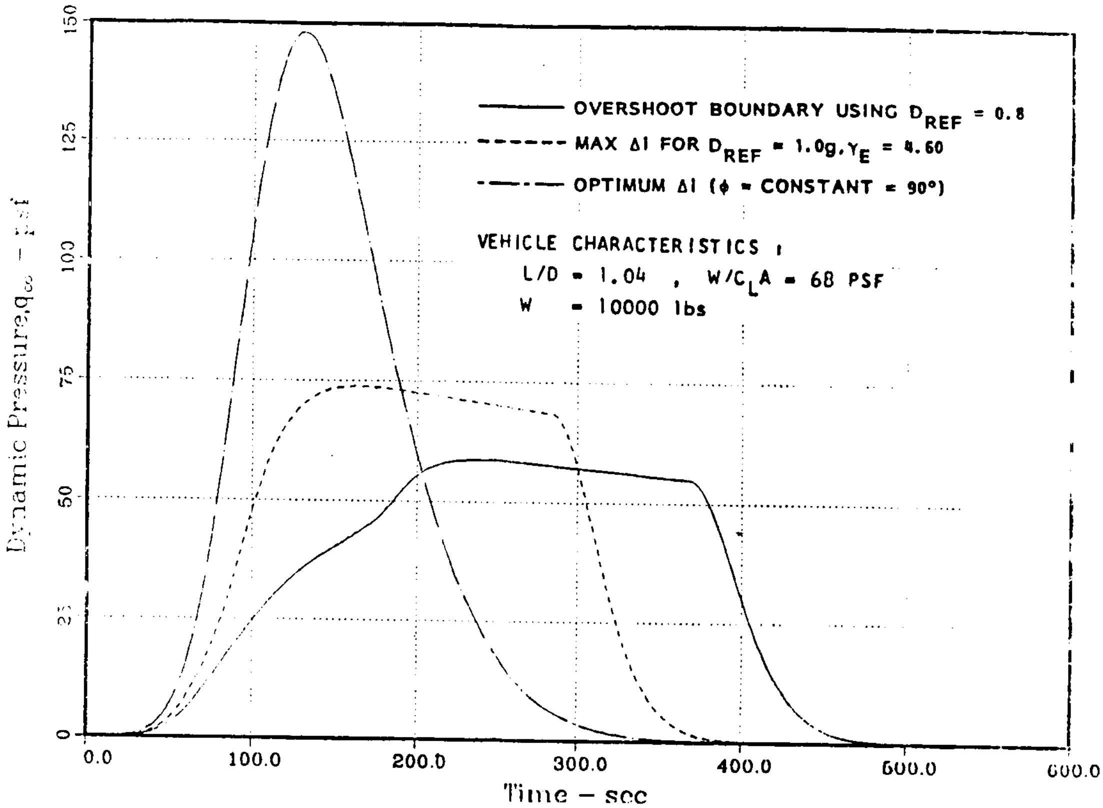


FIGURE 2.1-14 EFFECT OF STEERING SCHEME ON RELATIVE VELOCITY

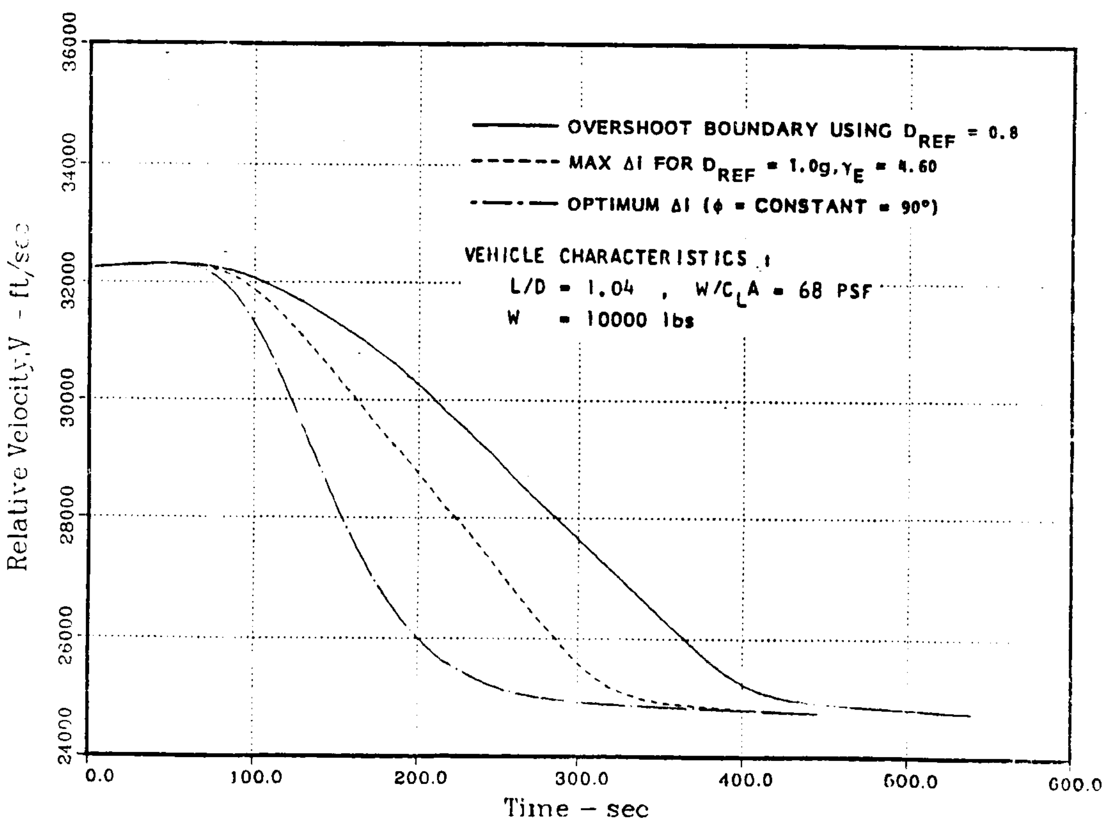


FIGURE 2.1-15 EFFECT OF STEERING SCHEME ON ALTITUDE TIME HISTORY

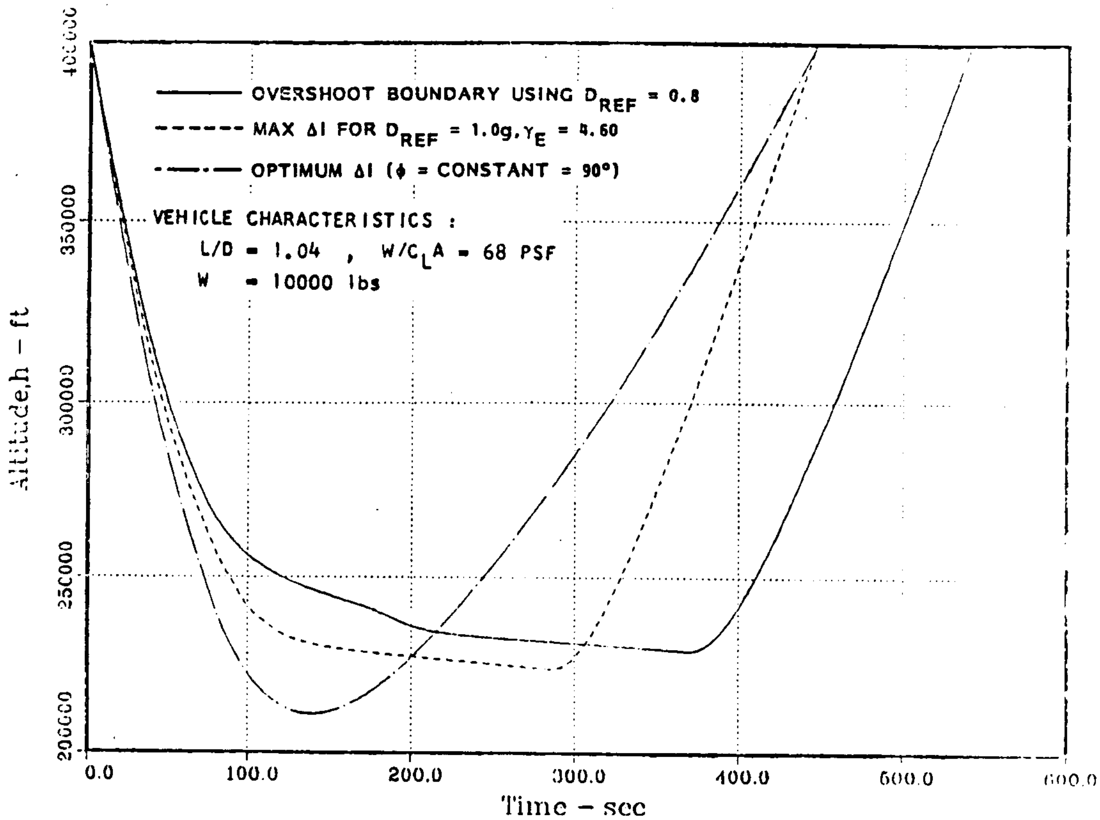
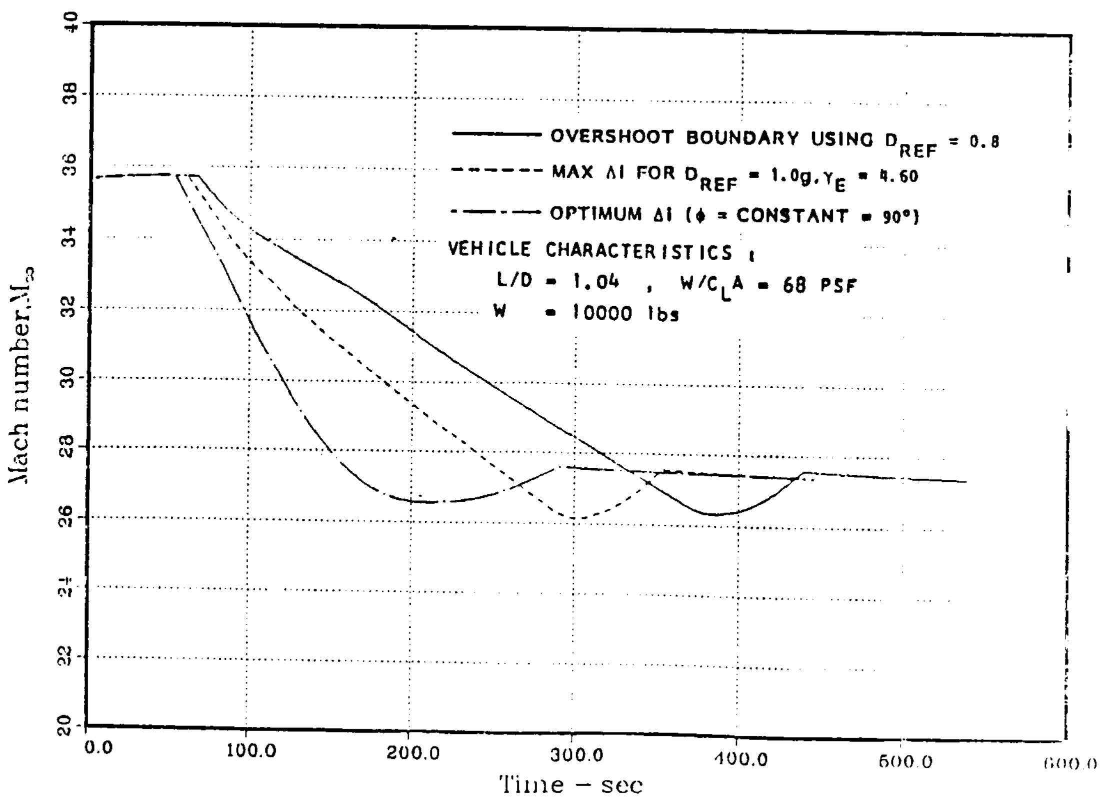


FIGURE 2.1-16 EFFECT OF STEERING SCHEME ON MACH NUMBER



Performance studies have been conducted for return of mid L/D vehicles from GEO, 5 x GEO, and 6-hour polar circular orbits employing steering laws that include constant deceleration cruise at the overshoot and undershoot bounds, and constant bank angle cruise. Orbital plane change obtained is summarized in Figure 2.1-17, 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.

Use of the various steering laws results in different minimum altitudes and thus different maximum heating rates, Figures 2.1-18 and 19. It can be noted that maximum heat transfer rate increases with vehicle ballistic coefficient, W/C_DA, with increasing entry velocity (5 x GEO results in maximum entry velocity) and with decreasing minimum flight altitudes (constant 90° bank angle results in minimum flight altitudes).

Minimum flight altitudes for these vehicles extend down to 160 kft for return from 5 x GEO of a 24K lbs, L/D = 1.5 vehicle. At altitudes below 190 kft, local boundary layer transition to turbulent flow may occur with a resulting increase in convective heat transfer by a factor of two to five. This must be avoided in order to minimize peak temperatures on the aft frustum area. A possible simple strategy involves employing an out-of-plane propulsive burn to obtain part or all of the desired plane change during the deorbit burn. This provides a significant increase in minimum flight altitude, laminar flow over the aft frustum, and reduced local heat transfer rates, Figure 2.1-18. At 5 x GEO altitude there is a very small propellant weight penalty for this burn. This technique allows flying a reference drag deceleration trajectory at the overshoot boundary.

FIGURE 2.1-17 **AOTV PLANE CHANGE CAPABILITY**

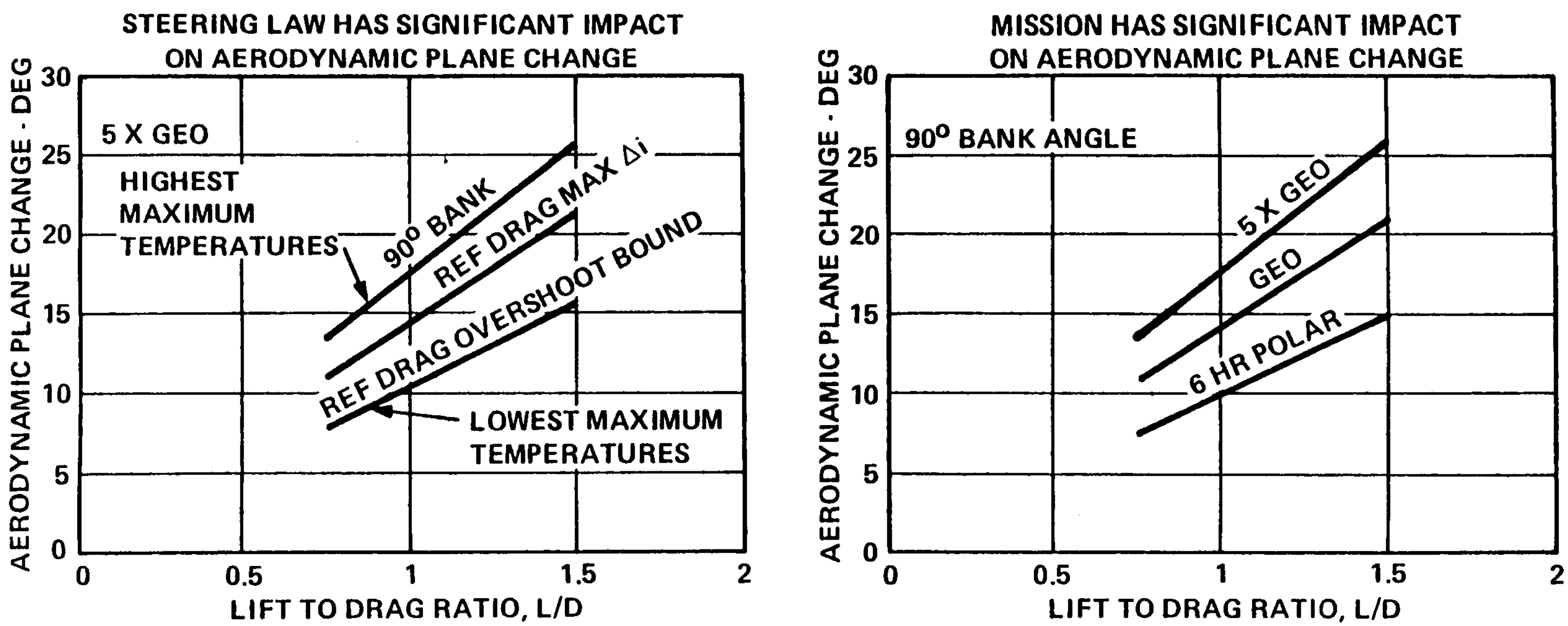


FIGURE 2.1-18

EFFECT OF BALLISTIC COEFFICIENT, STEERING LAW, AND MISSION ON MINIMUM FLIGHT ALTITUDE

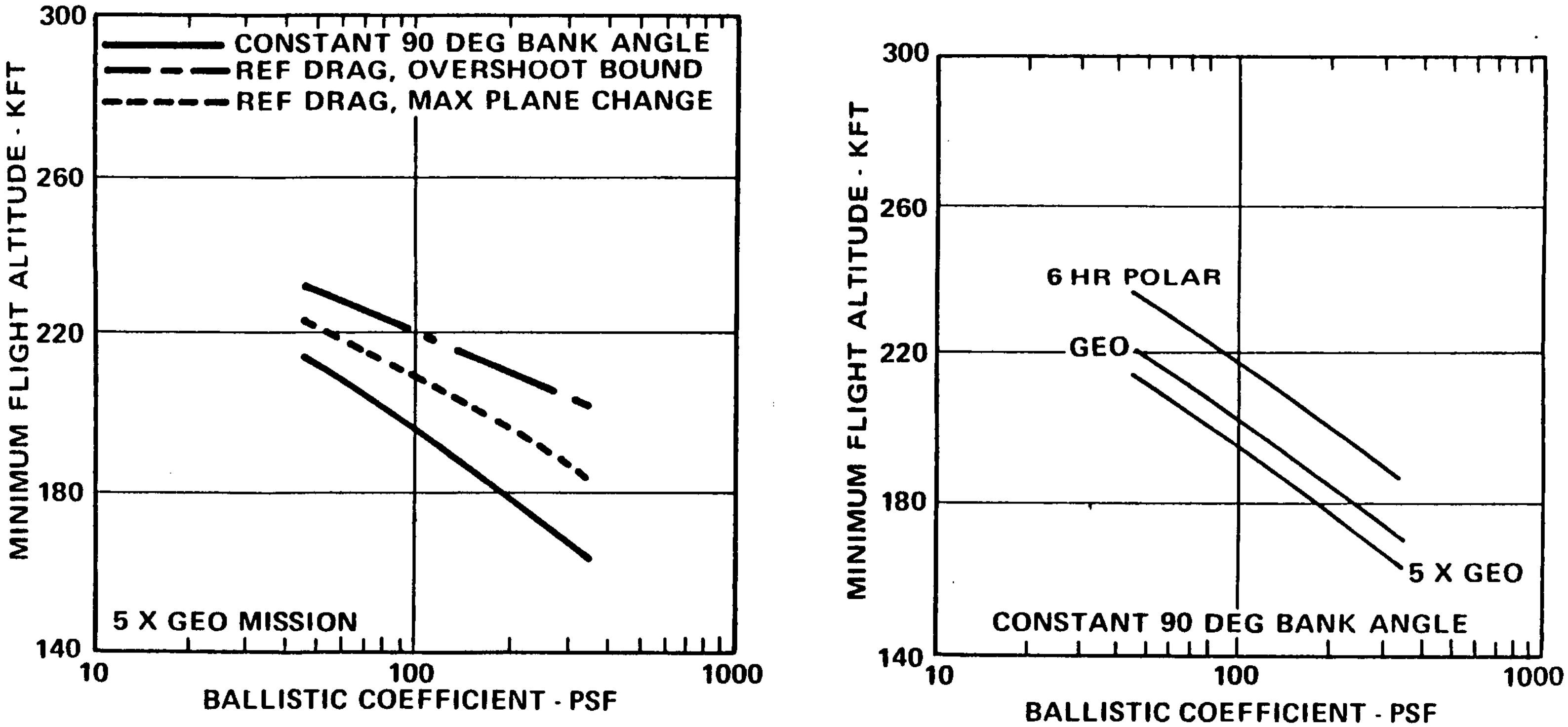
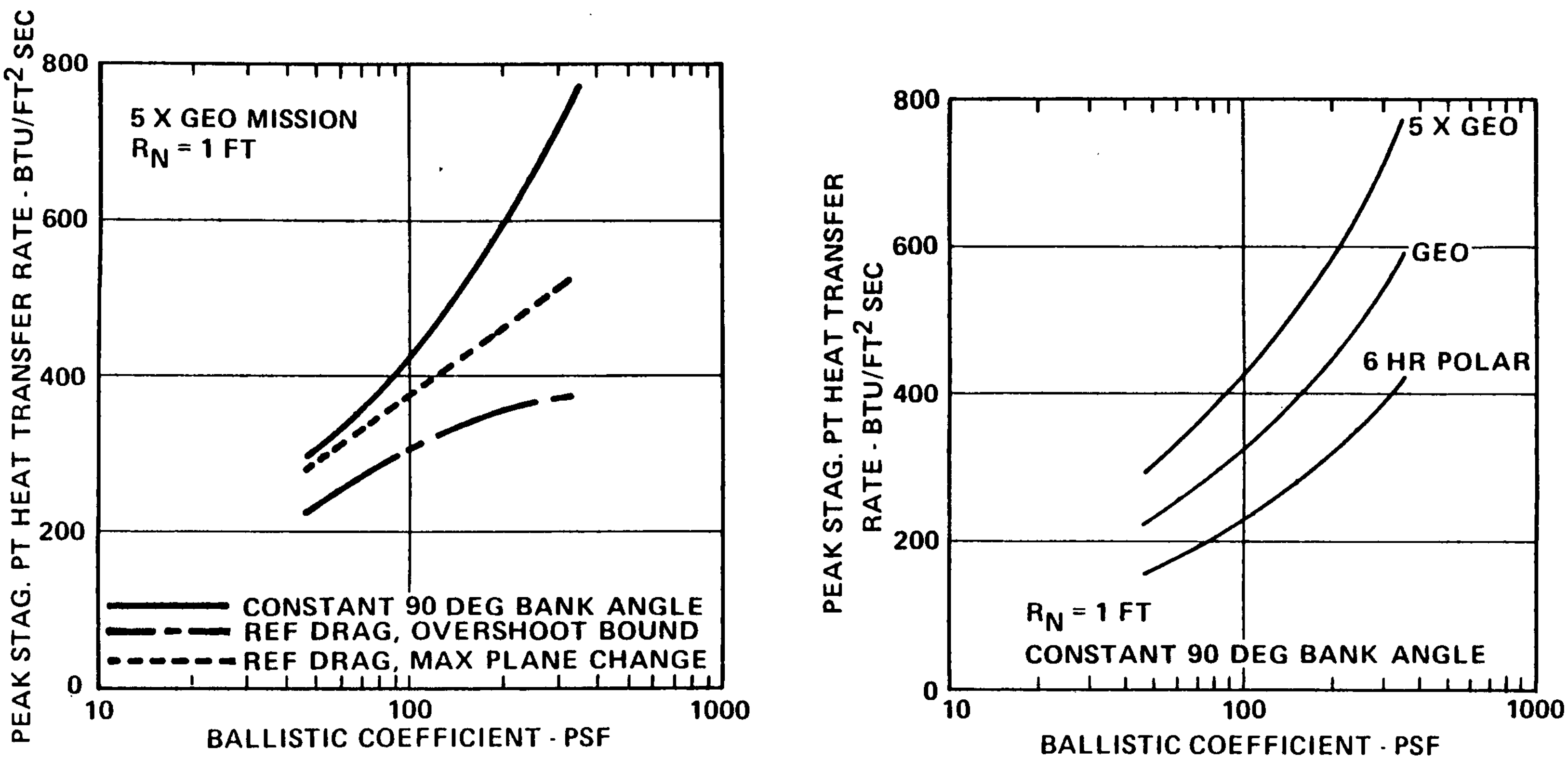


FIGURE 2.1-19

EFFECT OF BALLISTIC COEFFICIENT, STEERING LAW, AND MISSION ON MAXIMUM CONVECTIVE HEAT TRANSFER RATE



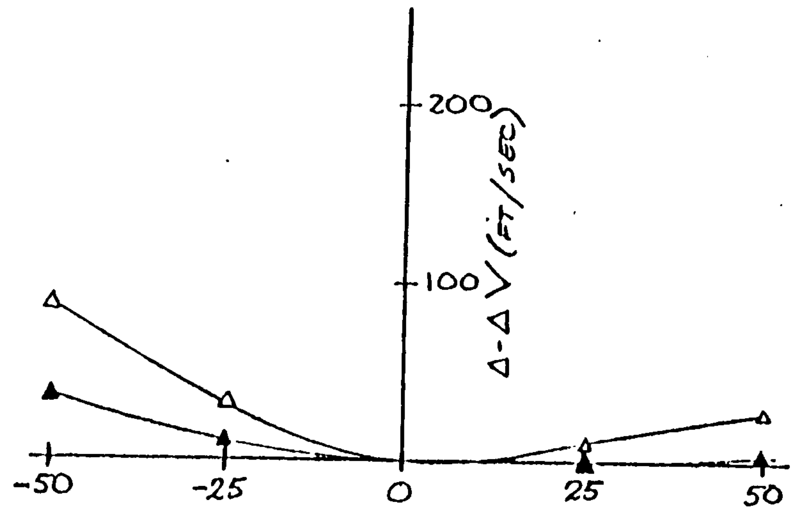
2.1.5 Some GN&C Results

Numerous steering law evaluations have been conducted (5, 22, 29, 30, 31, 32 and 33) to determine the magnitude of LEO circularization burn ΔV uncertainties resulting from an off-nominal atmosphere, errors in entry interface state conditions (VE, γE), and uncertainties in AOTV aerodynamics.

The insensitivity of an L/D = 1.5 AOTV to variations from the nominal in the atmopsphere 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 2.1-20. Note that the mid L/D is relatively insensitive to atmospheric and drag coefficient uncertainties.

FIGURE 2.1-20. MID L/D AOTV IS RELATIVELY INSENSITIVE TO ATMOSPHERIC DENSITY AND DRAG COEFFICIENT UNCERTAINTIES

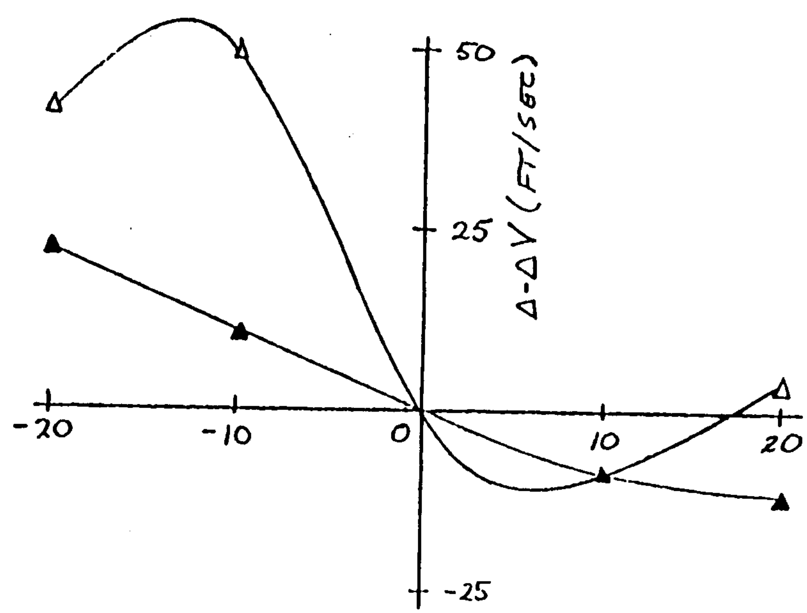
L/D = 1.5 $W/C_L S$ = 97 PSF



DENSITY VARIATION, %

SYMBOL	GUIDANCE	ΔV NOMINAL
OPEN	MPAD	202
FILLED	ET	239

SOURCE: L. SKALECKI
NASA JSC/ET3



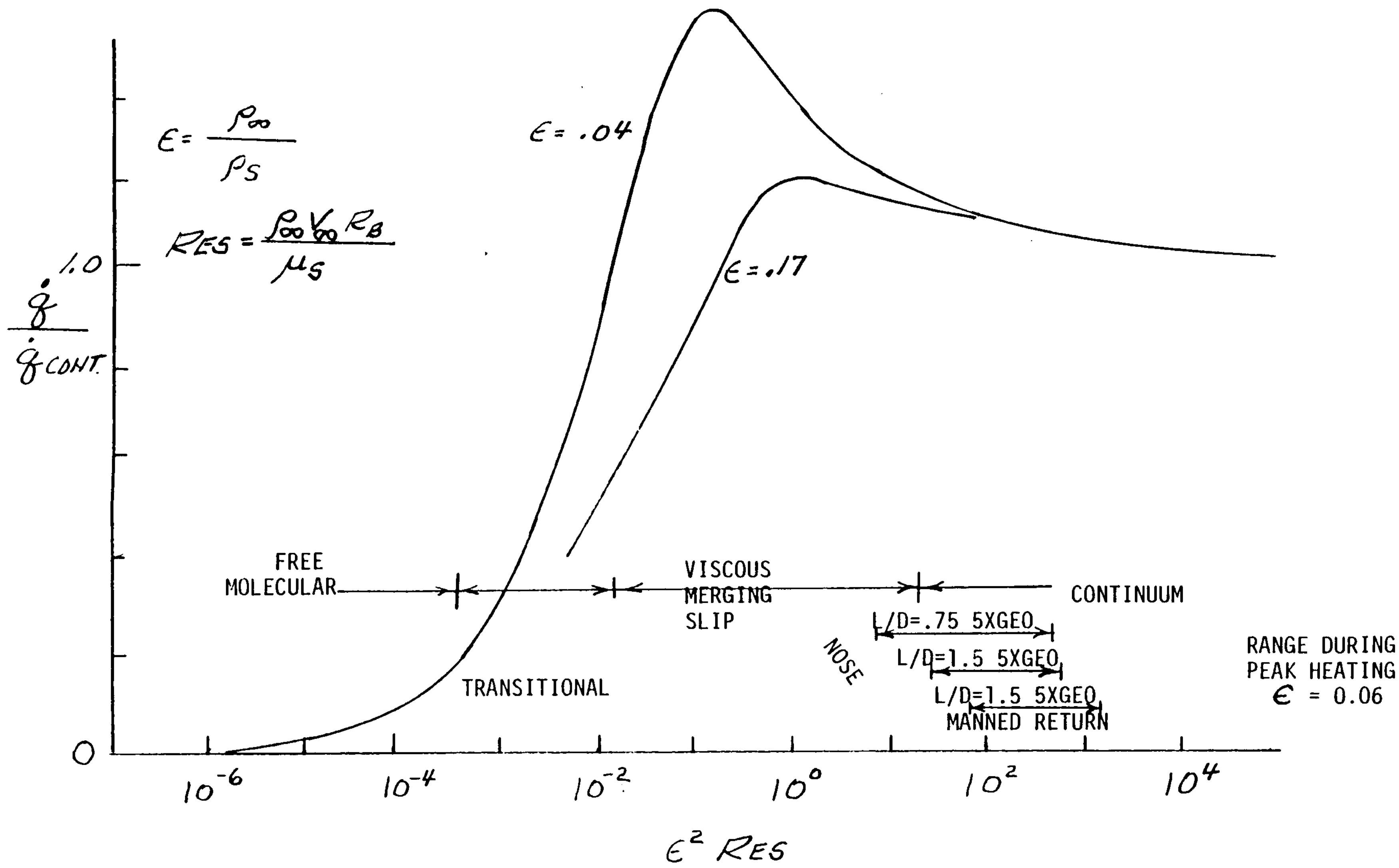
DRAG COEFFICIENT VARIATION, %

SYMBOL	GUIDANCE	ΔV NOMINAL
OPEN	MPAD	233
FILLED	ET	239

2.2 Aerothermodynamics

Several hypersonic flow regimes are traversed as the AOTV enters the atmosphere and executes the pull-up maneuver. Regimes encountered begin with free molecular flow and proceed through the transitional and viscous merging slip flows, finally entering full continuum flow during the constant "g" or minimum altitude portion of the mission and then proceeding back through the rarefield flow regime to free molecular as the exit maneuver is performed. Free molecular and continuum drag and heat-transfer characteristics are well defined from both theory and ground and flight test data. Characteristics in the transition and slip flow regimes are less well defined and predictions are based on a shock Reynolds number semi-empirical bridging technique developed by Gilbert and Goldberg (25). This bridging relationship represents a correlation of the numerical solutions of a theoretical model of the hypersonic viscous shock layer of air in chemical equilibrium over a range of shock Reynolds numbers of $50\text{-}10^4$ and is illustrated in Figure 2.2-1. The range of shock Reynolds numbers computed at peak heating for the various AOTV trajectories is illustrated in Figure 2.2-1 and indicates that the vehicle frustum is operating in full continuum regime during this time. For purposes of this evaluation, only continuum heating will be considered.

FIGURE 2.2-1. EFFECT OF FLOW REGIMES ON HEAT TRANSFER TO BLUNT BODIES



Numerous vehicle configuration and entry trajectory parameters effect the magnitude of the hypersonic convective and radiative heat transfer rates experienced by the AOTV. These parameters include entry velocity (mission dependent), entry path angle (determined apriori depending on steering law selected), vehicle configuration and size, ballistic coefficient (W/CDA), angle of attack, steering law and number of atmospheric passes. The combined impact of W/C_DA, entry trajectory/steering law and mission have been presented in Figure 2.1-19 for a single pass mission. In this study vehicles with significantly different configurations and size have been evaluated. The nominal cases evaluated have been flown at an angle of attack consistent with maximum L/D (to maximize payload delivered). Typical values of angle of attack range from 25° for L/D = 0.75 to 15° for L/D = 1.5, Figure 2.3-11.

The convective hypersonic heat transfer computed during this study was based on equilibrium flow and a fully catalytic thermal protection material surface. Space shuttle flight experience as well as numerous investigators (26, 28) have indicated that the shuttle is experiencing convective heat flux 20% less than the fully catalytic values, but more than the 50% reduction that would be expected with a fully non-catalytic surface. Computations for a range of AOTV's has indicated (20, 21) up to a 67% reduction potential for a fully non-catalytic material. Using the shuttle experience of a partially non-catalytic surface, for the AOTV flight conditions, results in an expected peak surface temperature reduction of up to 200°F on the aft frustum and up to 400°F on the nose.

Estimates have been made of equilibrium hot gas radiation using the results of Page(13) for the range of initial vehicles selected, Table 2.2-1. It can be seen that the hot gas radiation for the nose area is approximately 10% of the convective heat transfer for all cases except the manned vehicle (heavy AOTV) return from 5 x GEO where it increases to about 20%. Aft on the vehicle, the hot gas radiation drops significantly and is expected to be an even small fraction of the convective heat transfer. Hence, for the purposes of this study, hot gas radiation will be neglected, since it is a second order effect compared to all other parameters.

The coupling involved between steering law employed, aerodynamic plane change obtained, resulting hypersonic convective heat load and maximum heat transfer rate, heating time and thermal protection system weight required is illustrated in Table 2.2-2 for an L/D = 1.04 and W/C_DA = 68 psf. Note that as the plane change is increase from the overshoot bound value, the heat load and heating time decrease significantly while the maximum heat transfer rate increases. Thus, higher allowable local TPS surface temperatures, permit a lighter thermal protection subsystem.

Table 2.2-1. EQUILIBRIUM HOT GAS RADIATION IS SMALL REALTIVE TO CONVECTIVE HEAT TRANSFER

R_N = 1 FT

L/D	MISSION	$\dot{q}_{R_{STAG}}$ (BTU/FT^2 SEC)	$\dot{q}_{STAG}$ (BTU/FT^2 SEC)
0.75	5XGEO	22	294
1.04	5XGEO	31	362
1.5	5XGEO	60	511
1.5	5XGEO MANNED RETURN	155	776

Table 2.2-2. EFFECT OF STEERING LAW ON TPS WEIGHT FOR GEO RETURN

STEERING LAW	Δi (o)	$\int \dot{q}_s \sqrt{R_N}$	HEATING TIME (SEC)	$\dot{q}_{S_{MAX}}$ R_N=2FT	TPS WEIGHT / TPS WEIGHT @ OVERSHOOT BOUND
D_{REF}-OVERSHOOT BOUND	8.6	53,430	541	120	1.0
D_{REF}-UNDERSHOOT OPTIMUM	11.1	46,890	446	145	0.86
CONSTANT BANK ANGLE	14.6	40,310	447	194	0.85

CONCLUSION:

ALTERNATE STEERING APPROACH PROVIDES REDUCTION IN HEAT LOAD AND HEATING TIME -

THUS POSSIBLE REDUCTION IN TPS WEIGHT AT THE EXPENSE OF HIGHER LOCAL TPS SURFACE TEMPERATURES WHILE INCREASING AERODYNAMIC PLANE CHANGE

2.3 Aerodynamic Configuration Development

The AOTV configuration must provide a high packaging efficiency for the propellant tanks and various subsystems as well as meet the external constraints of the launch vehicle. The length and weight constraints to be used in this study for the various contemporary and advanced launch vehicles have been specified by NASA and are summarized in Table 2.3-1. The principal aerodynamic configuration drivers and assumptions identified at the initiation of this study are summarized in Table 2.3-2.

Table 2.3-1

LAUNCH VEHICLE OPTION EFFECT ON AOTV CONSTRAINTS

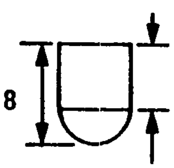
LAUNCH VEHICLE	MAXIMUM AOTV + ASE		
	LENGTH (FT)	DIAM (FT)	WEIGHT (K LBS)
STS	60	15	65
IMPROVED SHUTTLE	60	15	100
AFT CARGO COMPARTMENT	20.8  13'	25	65
SHUTTLE DERIVED CARGO VEHICLE	90	25	148

Table 2.3-2. AERODYNAMIC CONFIGURATION DRIVERS & ASSUMPTIONS

STS AND SDV LENGTH AND DIAMETER CONSTRAINTS PRODUCE AOTV $\ell/d \leq 4$

VELOCITY RANGE OF INTEREST: 25000 TO 35000 FT/SEC

ALTITUDE RANGE OF INTEREST: 180 TO 250 KFT

INITIALLY WILL CONSIDER FIXED TRIM VEHICLE

MAGNITUDE OF RADIAL AND AXIAL C.G. OFFSETS WILL PROBABLY REQUIRE SPLIT FLAP CONTROL

The hypersonic L/D varies significantly with altitude, due to the viscous drag effects, (6,23) but has been treated as a constant here since it has been demonstrated that inclusion of the high altitude effects has a 3rd order effect on the plane change obtained and payload delivered (6,22). Some insight into these results is obtained by noting that the flight regime where 90% of the velocity loss is experienced (20) is the continuum flow regime.

To provide initial direction to the mid L/D AOTV packaging studies, it was necessary to have an axial center-of-mass location requirement for both launch (propellant tanks loaded, AOTV in the shuttle orbiter) and entry (propellant tanks empty and in some cases staged).

Xcm requirement at launch is clearly defined in "Shuttle Orbiter/Cargo Standard Interfaces" ICD2-19001, JSC07700, Vol. 14, Attachment 1, Rev. G (24) for the STD 65,000 lb. STS. The entry Xcm requirement for both the AMOOS type configurations and the higher L/D biconics that are 60 ft. long is in the vicinity of 55% of the vehicle length (aft of the nose) to provide a trim angle consistent with (L/D) max, Table 2.3-3. This value was used in the initial configuration screening process.

Table 2.3-3 INITIAL AERODYNAMIC CHARACTERISTICS

AEROMANEUVER CENTER OF MASS REQUIREMENT, X_{CP}/L_V

CONFIGURATION	LENGTH / DIAMETER	(L/D) MAX	(L/D)	$\alpha(°)$	$\delta_F(°)$	X_{CP}/L_V
LOCKHEED - AMOOS 5	~4	1.5	1.4	20	–	0.55
LOCKHEED - AMOOS 5	~	1.5	0.75	52	–	0.55
LOCKHEED - AMOOS 5B	~4	1.04	1.04	-	-	0.43-0.53
BOEING - AMOTV	~4	.85	.4	60	0	0.57
GE AEROCAPTURE 5-7	3.9	1.7	1.7	20	0	0.50
GE AEROCAPTURE 7-6	2.4	1.25	1.25	20	0	0.44
BOEING-AMOTV DELIVERY	2	0.8	0.8	35	-40	0.44

SELECTED X_{CG}/L_V = 0.55 AS INITIAL REQUIREMENT FOR CONFIGURATION SCREENING OF ℓ/d = 4 VEHICLE

To conduct the initial 3-DOF flight mechanics evaluations, where in atmosphere flight control steering laws were evaluated, primary operating altitudes, hypersonic flow regimes, and reference heat transfer determined, it was necessary to specify some typical aerodynamic characteristics. A survey of previous mid L/D AOTV type vehicles was conducted, Table 2.3-4. For the initial evaluations an AOTV return weight of 10,000 lbs was assumed. Using the aerodynamic characteristics of AMOOS 5B, operating at (L/D)max, results in a W/CLA of 68 psf.

TABLE 2.3-4. **INITIAL AERODYNAMIC CHARACTERISTICS**

AEROMANEUVER LIFT LOADING, $W/C_L A$

FOR W = 10K LBS

CONFIGURATION	α(°)	L/D	$W/C_L A$, PSF
LOCKHEED AMOOS 5B	<30°	1.04*	68
	45°	0.6	30
BOEING - AMOTV	25	0.85*	53
	50	0.55	30
GE - AEROCAPTURE 7-6	20	1.7*	88
	35	1.07	42
	45	0.75	39

SELECTED FOR INITIAL FLIGHT PERFORMANCE EVALUATIONS

* ~ L/D MAX

The mid L/D configuration evolution was aided by the large aerodynamic coefficient analytical and experimental data bank that already existed for conic and biconic bodies with various control surfaces and aft frustum angles down to 4°. Additional characteristics are available from the AMOOS studies, Reference 6 for cylindrical aft bodies and an aft frustum angle of 0.5°. To supplement these available results, additional new computations were performed for aft frustum angles of 1° and 2° employing HABP, Reference 8, various nose and vehicle length combinations, and various nose bend angles. This chronological sequence of events is outlined in Figure 2.3-1.

The initial configuration class defined for performing the additional computations is illustrated in Figure 2.3-2. The results of these computations are summarized in Figures 2.3-3, 4, 5, and 6 for a vehicle with a full nose bend.

The dramatic effect that the nose length has on hypersonic L/D is illustrated in Figure 2.3-3; the change of nose radius from one to two feet is shown to have a negligible effect on maximum L/D in Figure 2.3-4. The aft movement of center of pressure location with angle of attack and increased nose length is illustrated in Figure 2.3-5; the effect of increased volume, non-circular cross section nose on hypersonic L/D is illustrated in Figure 2.3-6.

FIGURE 2.3-1

AOTV MID L/D AERODYNAMIC CONFIGURATION EVOLUTION

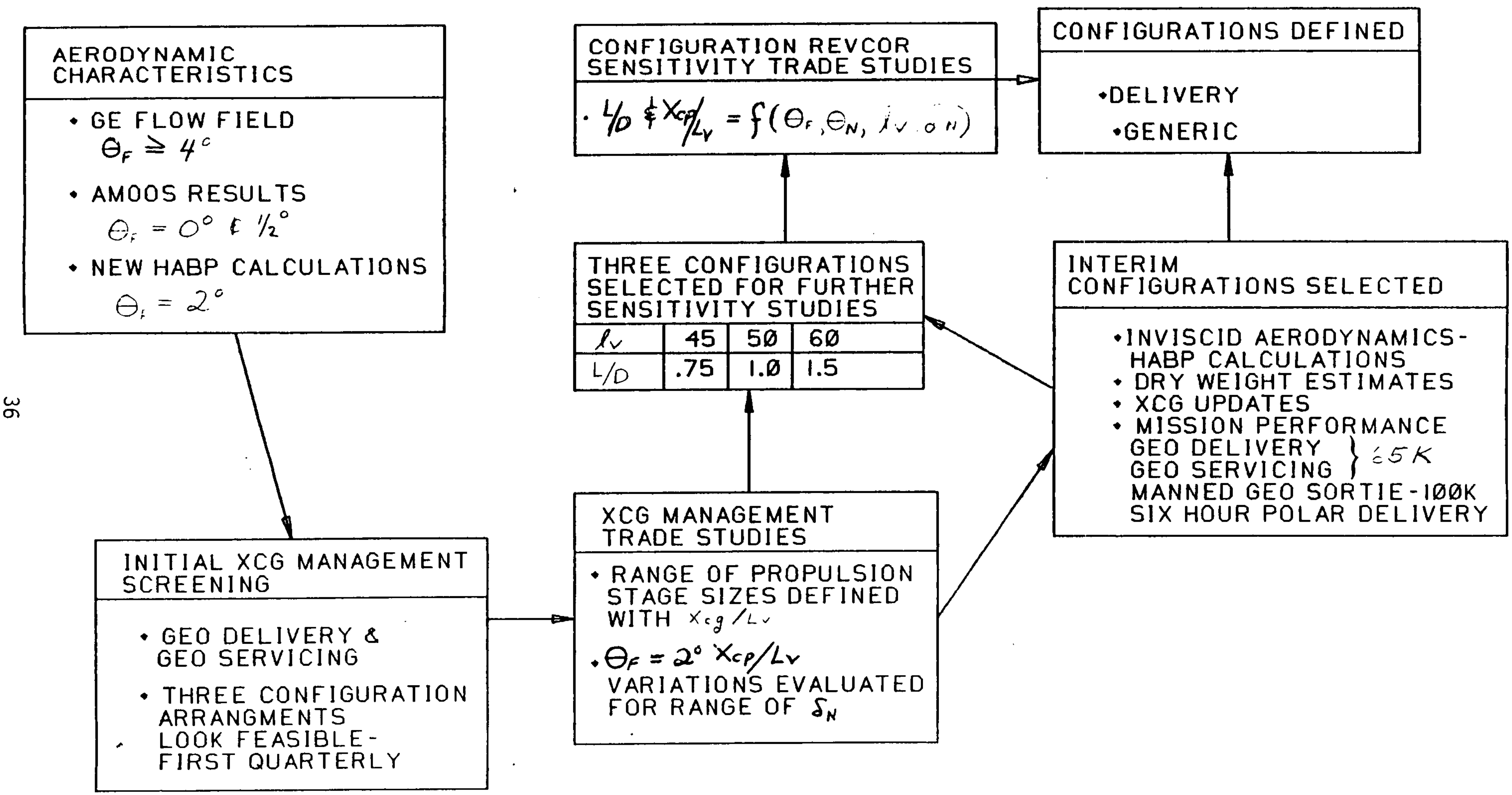


FIGURE 2.3-2. AERODYNAMIC CONFIGURATIONS EVALUATED

R_N = 1.0 - 2.0 FT

ℓ_N = 5 TO 35 FT

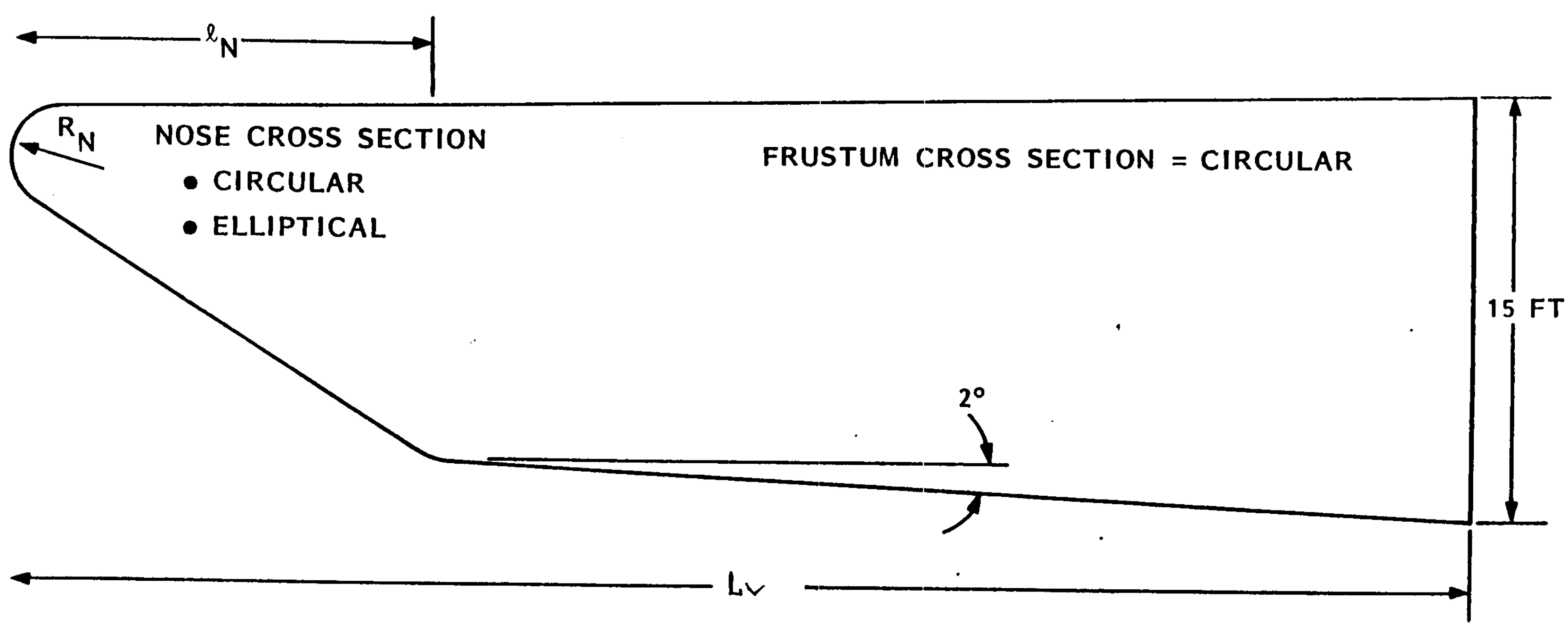


Figure 2.3-3. EFFECT OF NOSE LENGTH ON HYPERSONIC LIFT/DRAG RATIO

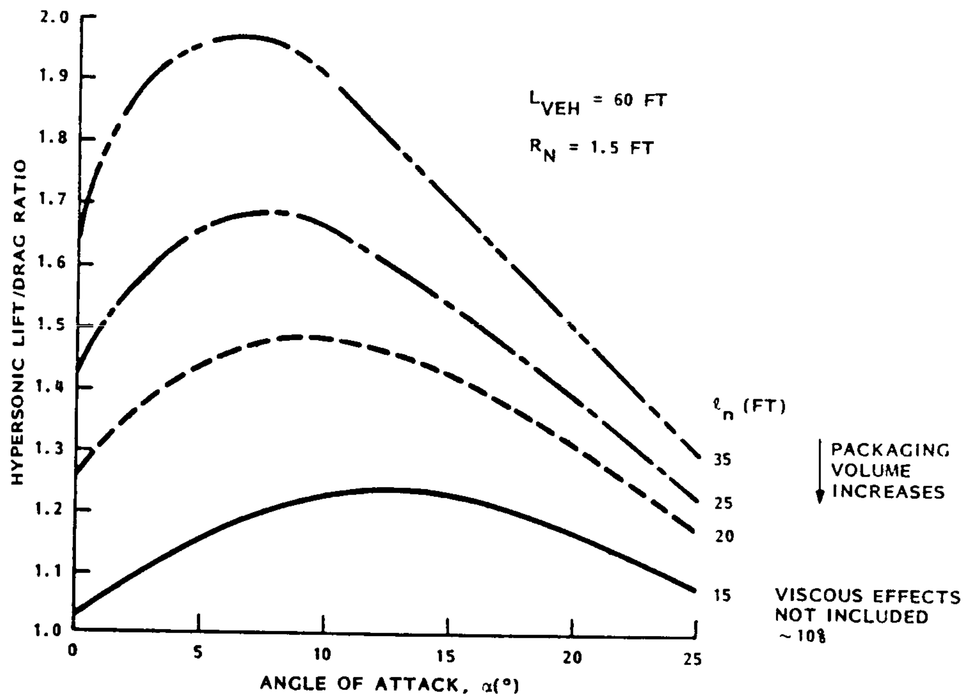


Figure 2.3-4. EFFECT OF ANGLE OF ATTACK AND NOSE RADIUS ON HYPERSONIC LIFT/DRAG RATIO

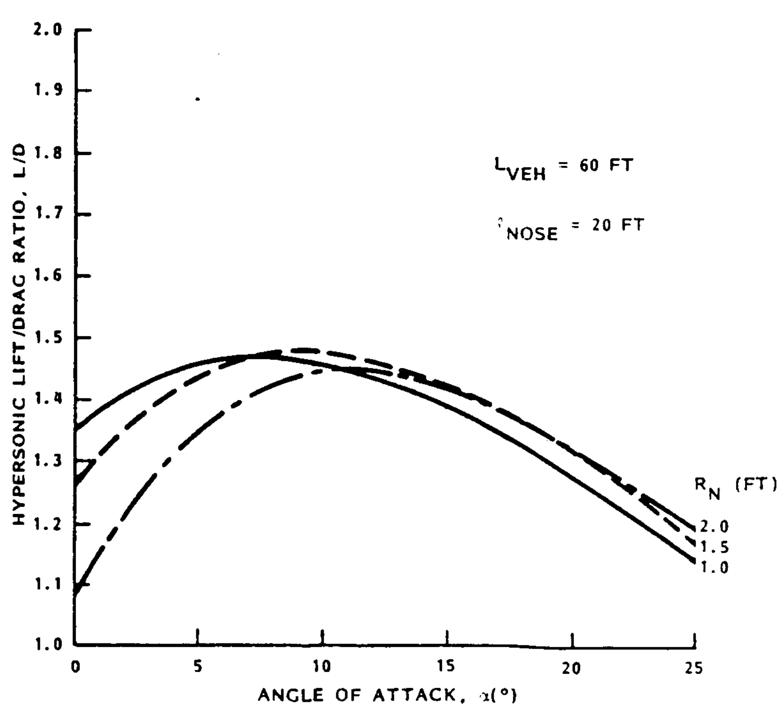


Figure 2.3-5. EFFECT OF ANGLE OF ATTACK AND NOSE LENGTH ON CENTER OF PRESSURE LOCATION

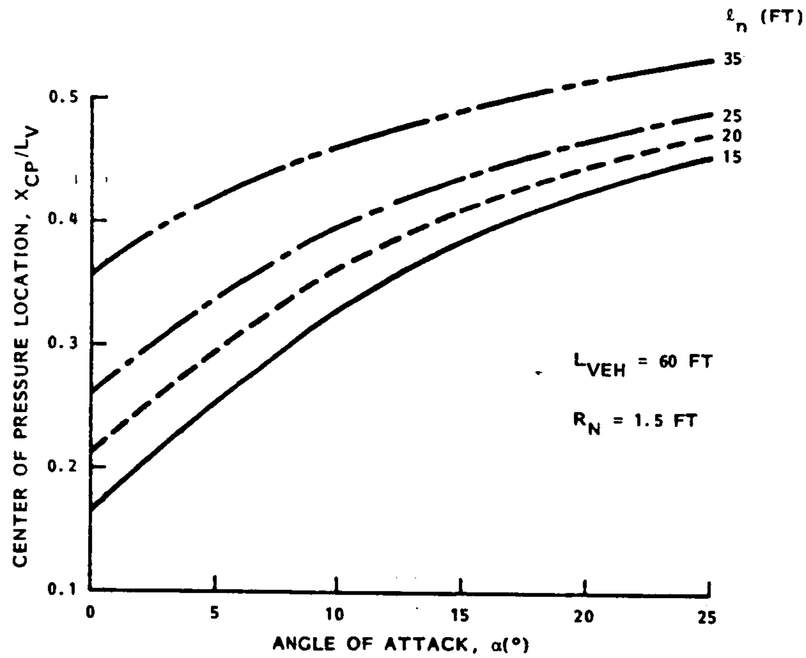
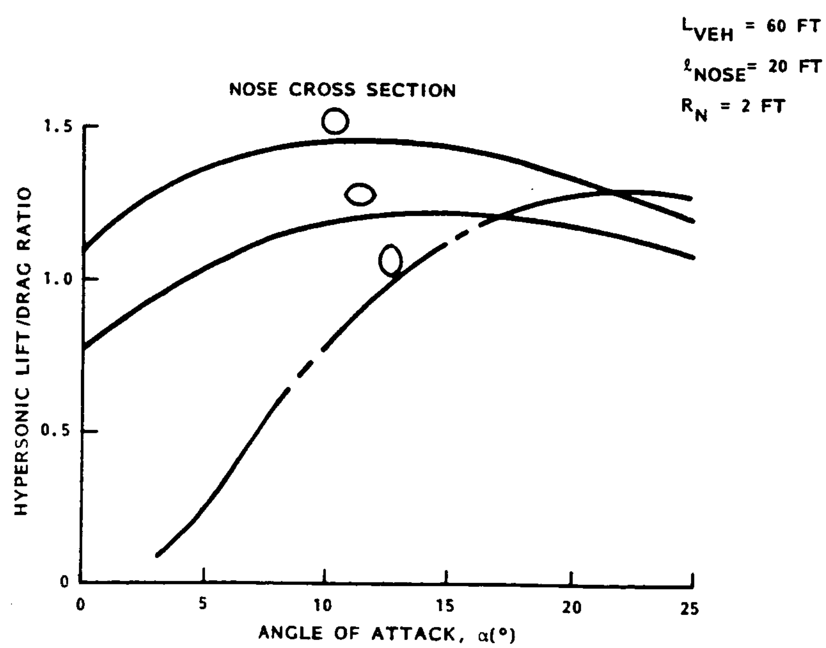


Figure 2.3-6. EFFECT OF NOSE CROSS SECTION ON HYPERSONIC LIFT/DRAG RATIO



CONCLUSION: NON CIRCULAR CROSS SECTION NOSE OFFERS INCREASED PACKAGING VOLUME AT SMALL LOSS OF L/D

For a single stage vehicle, propulsion stage packaging trends were evaluated to determine vehicle center-of-mass possibilities for combinations of total vehicle length, LV, and nose length, Ln, Figure 2.3-7. Additional aerodynamic computations were performed for shorter vehicles and for vehicles with less than a full nose bend, Figures 2.3-8. These results, in combination with the parametric center-of-pressure locations, Figure 2.3-9, were used to define three configurations, Figure 2.3-10, that span the range of L/D from 0.75 to 1.5 for further evaluation.

FIGURE 2.3-7. AOTV Propulsion Stage Packaging Trends

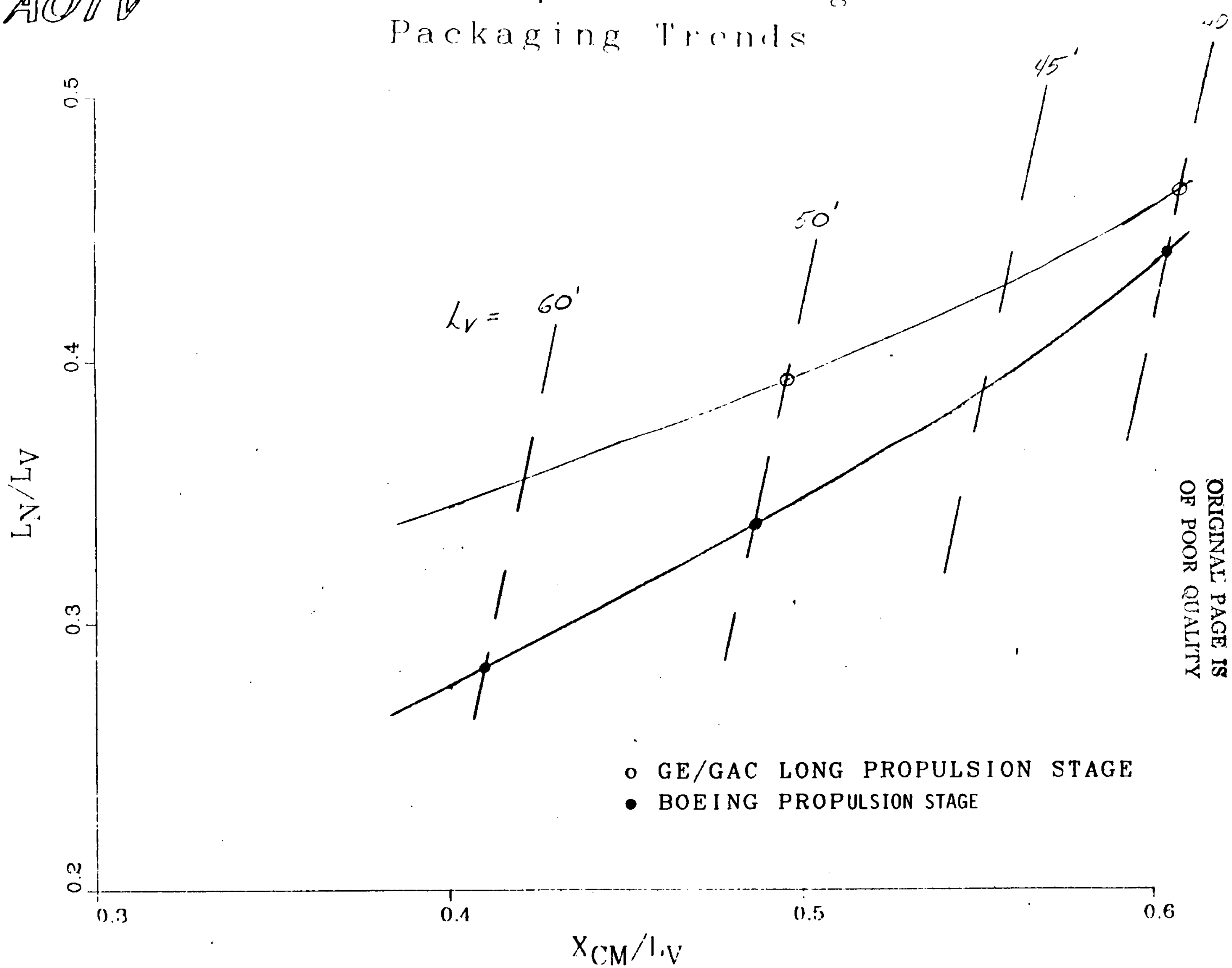


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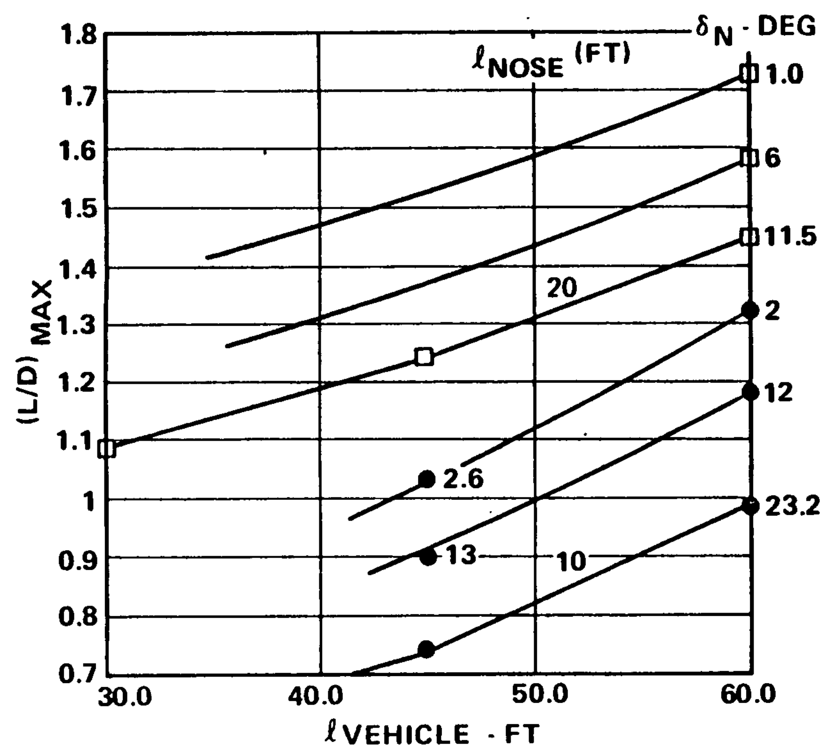
FIGURE 2.3-8. **EFFECT OF NOSE BEND ON MAXIMUM L/D**
• HABP NEWTONIAN CALCULATIONS

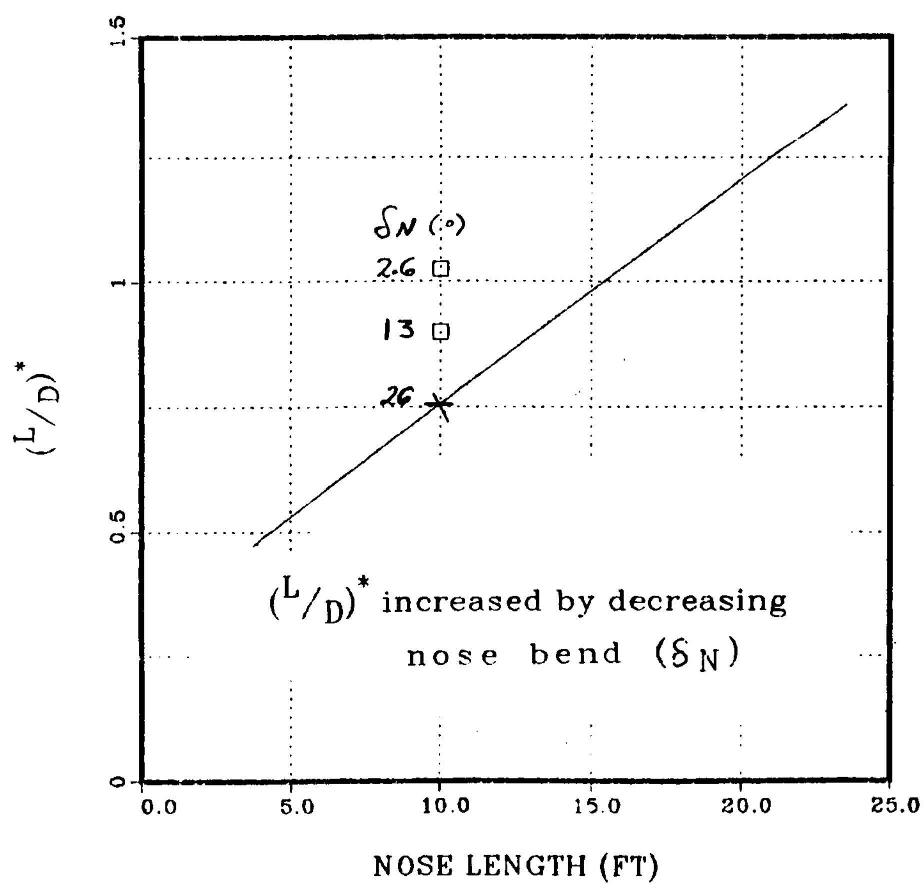


SYMBOL	ℓ_{NOSE} (FT)
OPEN	20
SOLID	10

FIGURE 2.3-9

Effect of Nose Length & Nose Bend on Aero Characteristics

$l_V = 45$ FT



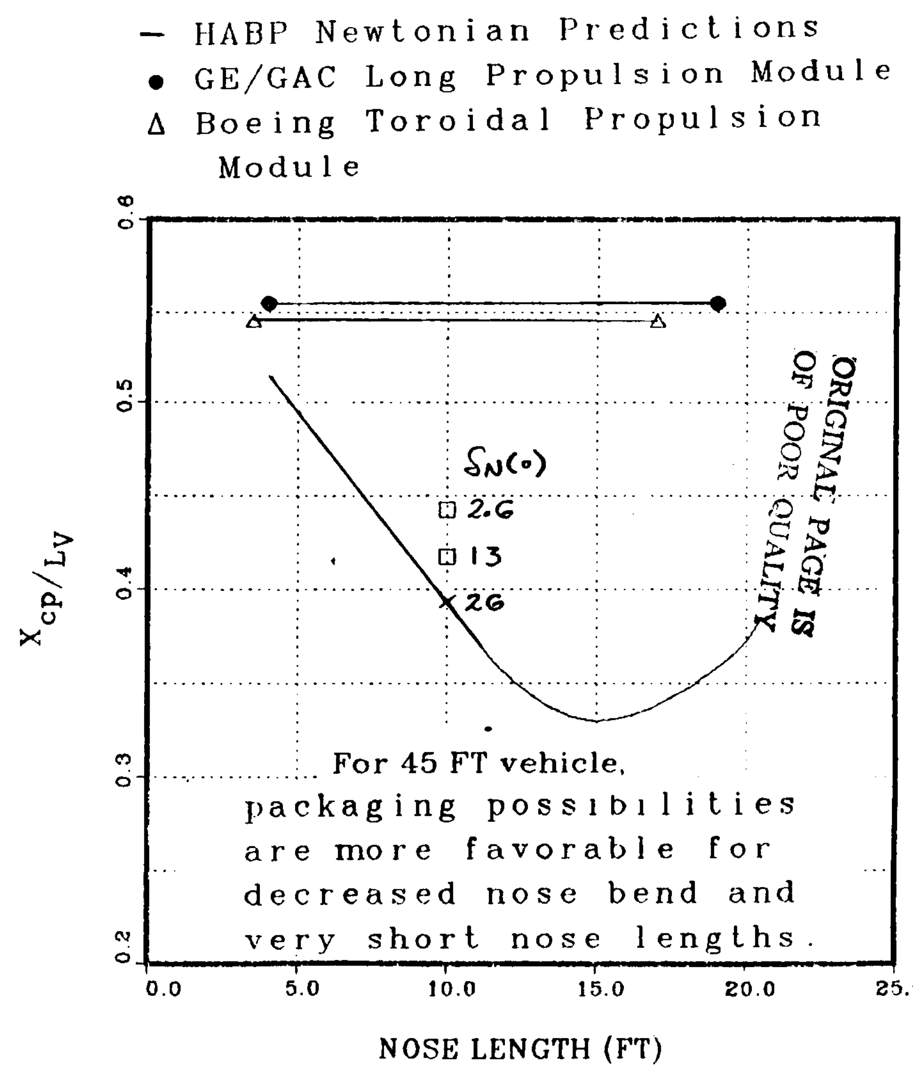


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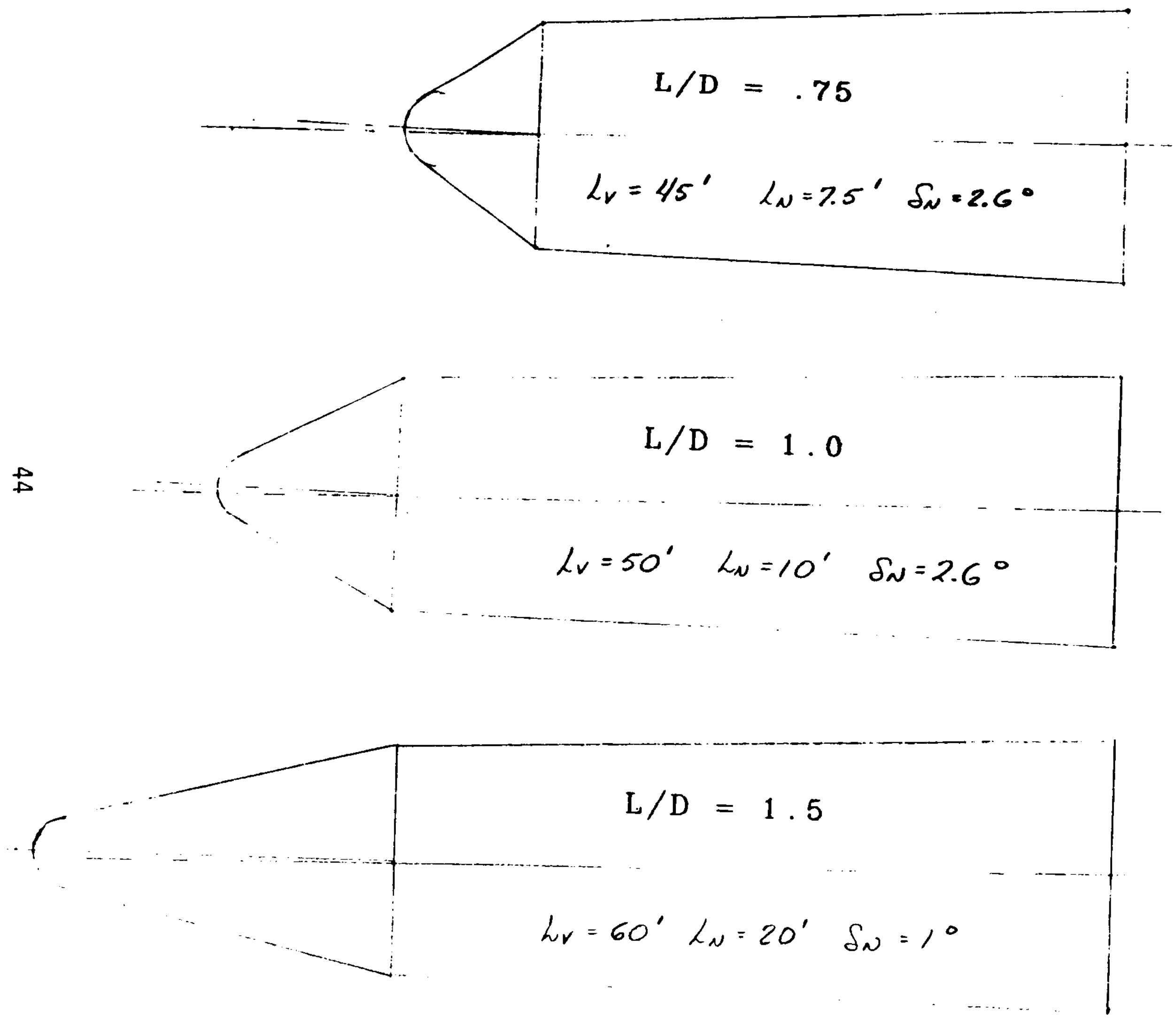
FIGURE 2.3-10

AOTV Configurations Selected for Further Sensitivity Studies with Attractive X_{cg} Requirements

$D_{BASE} = 15'$
$R_N = 2'$
$\theta_F = 2^{o}$



Specific aerodynamic characteristics of these three configurations have been generated and are summarized in Figures 2.3-11, 12 and 13. Note that the angles of attack required to obtain (L/D) max range from 15° to 25°, Figure 2.3-11. The effect of decreased nose bend on hypersonic L/D and static pitch stability, $\partial Cm/\partial Cn$ is illustrated in Figure 2.3-14.

Figure 2.3-11 Variation of L/D with Angle of Attack for AOTV Interim Configurations

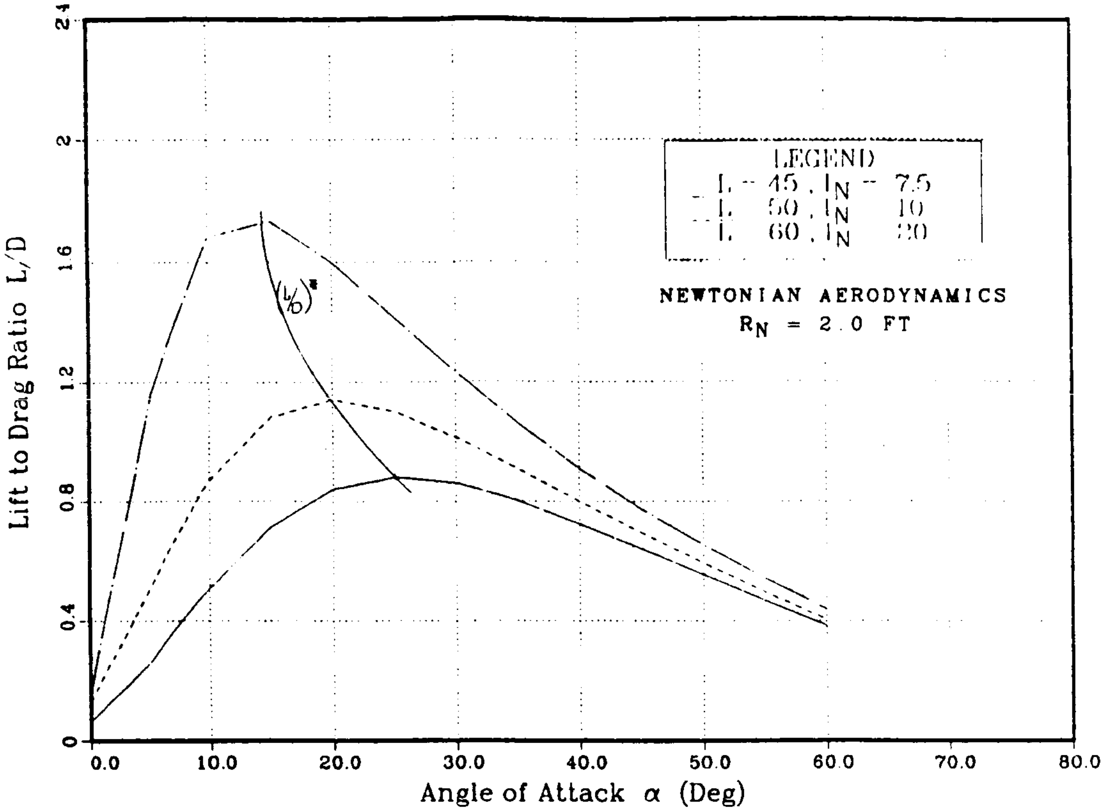
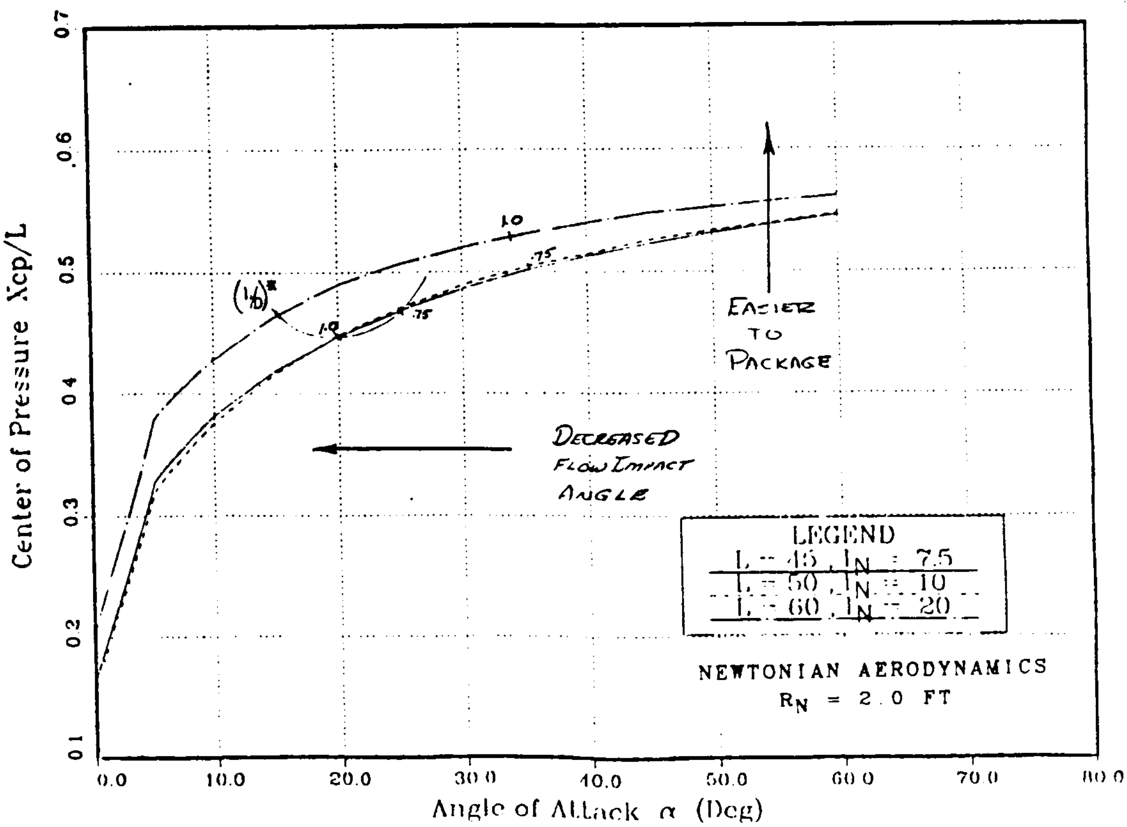


Figure 2.3-12 Variation of Center of Pressure with Angle of Attack for AOTV Interim Configurations



AOTV

FIGURE 2.3-13

Variations of Lift Coefficient with Angle of Attack for AOTV Interim Configurations

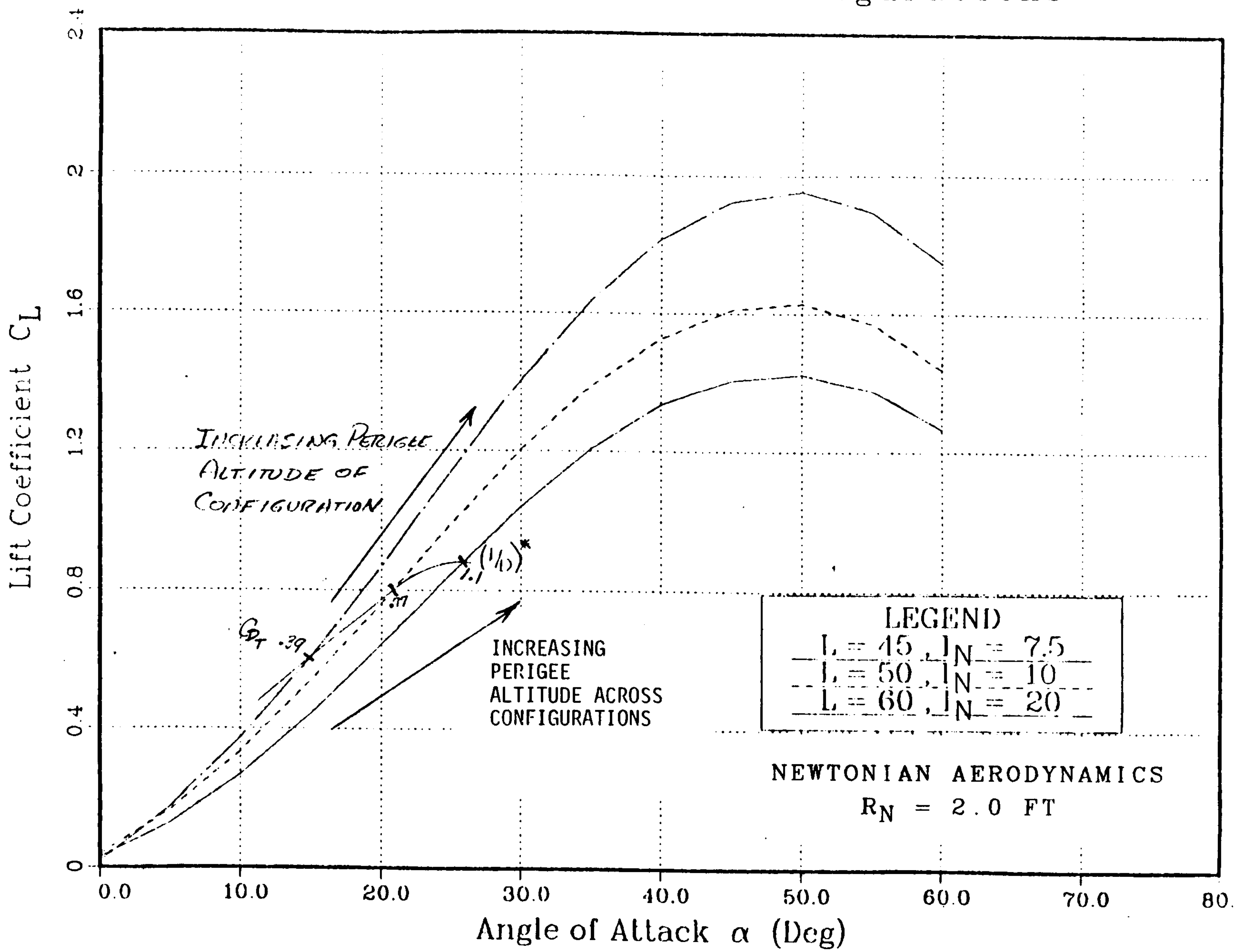
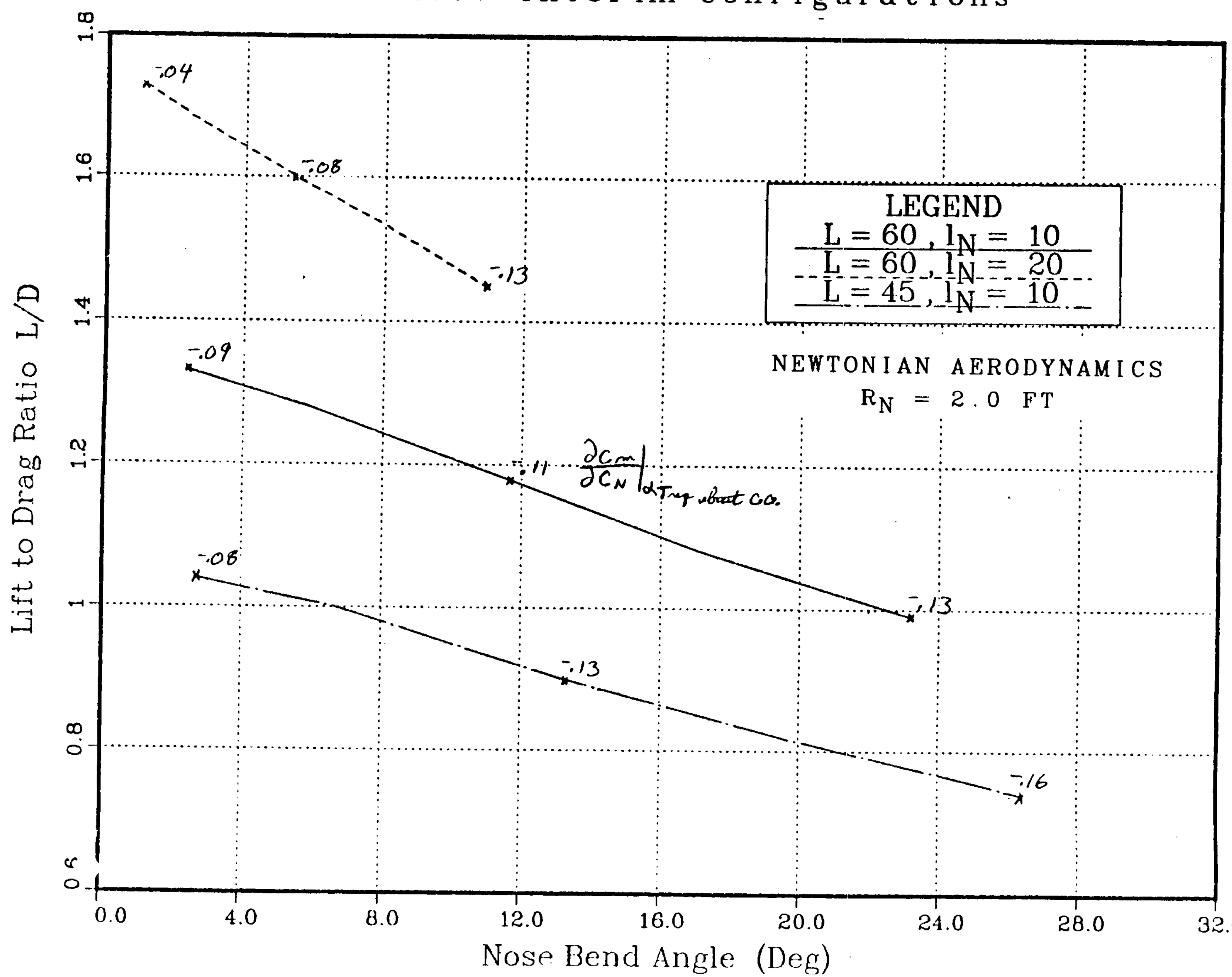


Figure 2.3-14

Effects of Nose Bend Angle on L/D_{MAX} & Pitch Stability for AOTV Interim Configurations



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Estimates have been made of the aft frustum angle and vehicle length effect on (L/D)max and Xcp/Lv using the RVCOR code. The RVCOR code is a high speed engineering design computer code based on algorithms developed from exact flow field computations, and flight and ground test data. These results can be employed to generate the incremental effect on (L/D)max of increased aft frustum angle and nose length, Figure 2.3-15, 16 and 17.

FIGURE 2.3-15

RVCOR ''Estimates'' of AOTV Configuration Trends

$R_B = 7.5'; X_F = 40'$

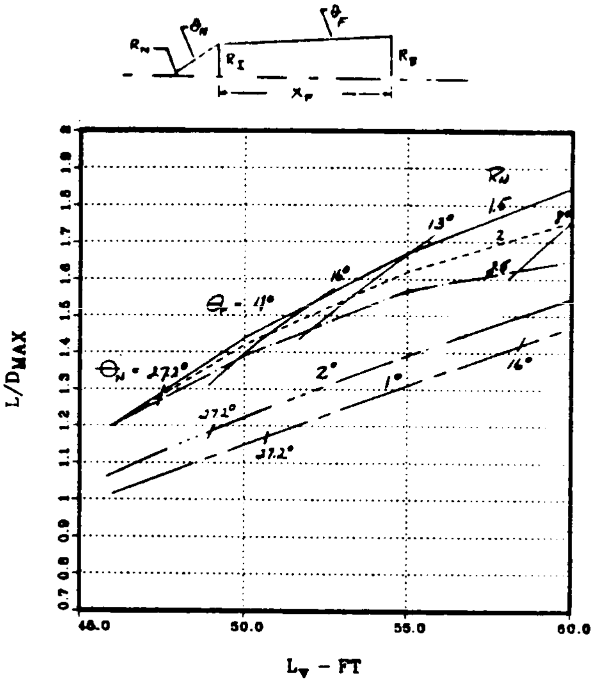


FIGURE 2.3-16 RVCOR ''Estimates'' of AOTV Configuration Trends

$R_B = 7.5'; R_N = 2'; X_F = 40'$

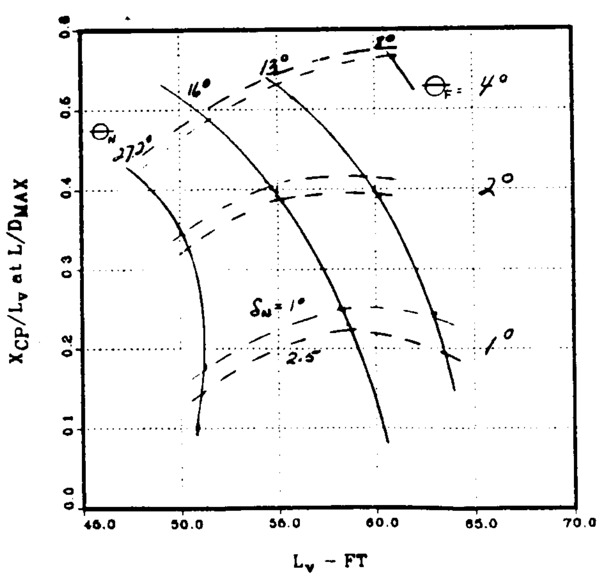
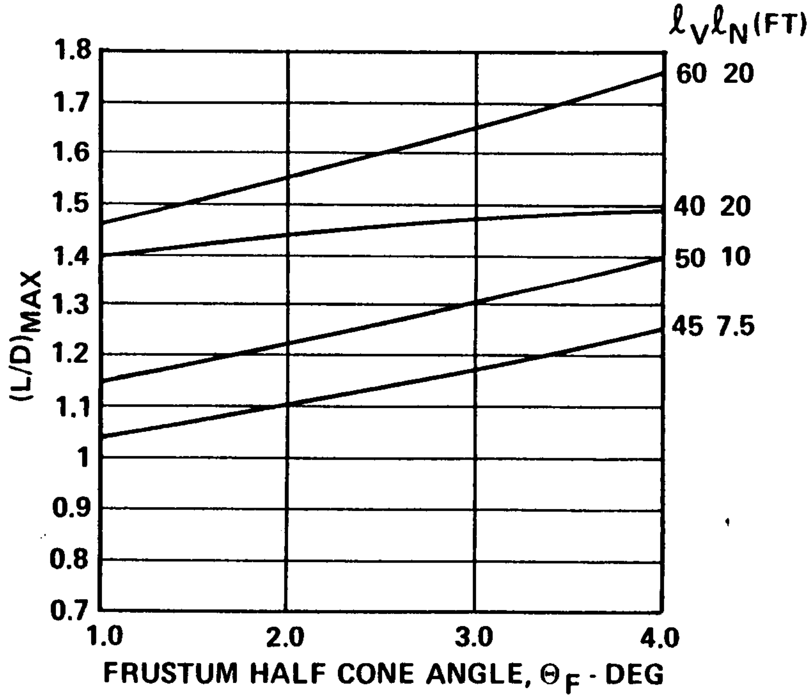


FIGURE 2.3-17 **EFFECT OF AFT FRUSTUM ANGLE ON MAXIMUM L/D**
— RVCOR ESTIMATES —
$\delta_N = 0^o$

INCREASED $(L/D)_{MAX}$ RESULTS FROM INCREASED AFT FRUSTUM ANGLE & INCREASED VEHICLE/ NOSE LENGTH



2.4 Configuration/Concept Development

2.4.1 Mass Estimates

To start the AMOTV mission analysis it is necessary to establish a set of vehicle reference mass properties. This was done by reviewing the OTV mass properties developed by the various contractors who have evaluated the APOTV and AOTV missions (3, 4, 6, 9). Since the vehicles varied in size, weight of propellant, configuration and subsystem requirements, the reference weights can be considered only in trends.

For parts that are directly affected by the vehicle configuration, such as shell length or diameter and tank volume, a unit weight per variable was selected. For subsystems and parts, like avionics and engine, a unit weight is used that is assumed constant for any AOTV vehicle. The structure is assumed for initial sizing to be not critical for the aeromaneuver loading since it now appears that the g level for the atmosphere deceleration will not exceed 1 to 1.5. Therefore, the all propulsive vehicle structure was reviewed as well as the AOTV for unit weight. It was assumed that the vehicle is roughly cylindrical, about 15 feet in diameter.The cryogenic tank material usually selected is 2219-TB7 aluminum. The final tank shape used affects the efficiency of the structure and hence its unit weight, i.e., the toroidal fuel tank is less structurally efficient than a spherical one.

A summary of the mass properties used in preceeding OTV studies is given in Table 2.4-1 and 2. Table 2.4-3 contains the unit AOTV masses employed during this study for configuration and mission evaluation.

The evaluation described above has been made for a basic 65,000 lbs shuttle payload with an AOTV vehicle diameter of 15 feet. For increased shuttle payload masses the following assumptions are made:

- The propulsion related components and subsystem weights are based on total propellant mass.
- The EPS and avionics subsystem masses are assumed constant.
- The structure unit mass increases as the square root of the propellant mass ratio for a range of two-to-one as shown in the MSFC AMOOS study (9).
- The TPS mass will increase by 10% for each doubling of the re-entry weight.

Table 2.4-1. OTV HISTORICAL MASS PROPERTY DATA

SUBSYSTEM	SOURCE	MASS PROPERTIES
STRUCTURE	LMSC	42.5 to 80 lbs/ft of vehicle length; Be-Al to Ti material
	GDC	43.3 lbs/ft Gr/Ep material
	BAC	45 lbs/ft Gr/Ep 25 lbs/ft skin panels 20 lbs/ft rings, bulkheads, and equipment supp.
TANKS-FUEL	BAC	W = 73.6 lbs/k lb fuel @ 6889 lbs of fuel
	GDC	W = 67.3 lbs/k lb fuel @ 7680 lbs of fuel
	MSFC	W = 57 lbs/k lt fuel @ 7458 lbs of fuel
	MSFC	W = 60.1 lbs/k lb fuel @ 11,894 lbs of fuel
	MSFC	W = 62.9 lbs/k lb fuel @ 14,179 lbs of fuel
- OXIDIZER	BAC	W = 10.5 lbs/k lb oxidizer @ 39,611 lb of oxidizer
	GDC	W = 14.95 lbs/k oxidizer @ 46,083 lb of oxidizer
	MSFC	W = 5.43 lbs/k lb oxidizer @ 44,742 lb of oxidizer
	MSFC	W = 7.3 lbs/k lb oxidizer @ 71,361 lb of oxidizer
	MSFC	W = 8.43 lbs/k lb oxidizer @ 85,069 lb of oxidizer
FLAPS	LMSC	W = 224 lb/4.6 kb = 49 lbs/k lbs
PROPULSION (TVC, PLUMBING PRESS, ETC)	MSFC	W = 603 lb/50,200 lb = 12.lb/k lb prop.
	GDC	W = 785 lb/53,763 lb = 14.6 lbs/k lb prop.
	BAC	W = 533 lb/46,247 lb = 11.5 lbs/ lb prop.
PROPULSION TPS	GDC	W = 8.05 lbs/k lb prop @ 53,763 lb prop.
	BAC	W = 6.77 lbs/k lb prop @ 46,500 lb prop.
	MSFC	W = 8.45 lbs/k lb prop @ 52,200 lb prop.
	MSFC	W = 8.25 lbs/k lb prop @ 83,260 lb prop.
	MSFC	W = 8.73 lbs/k lb prop @ 99,200 lb prop.
ACS	GDC	359 lb
	BAC	138 lb Average 391 lb. 391/10300 lb = 38 lb/k lb of aeromaneuver mass
	MSFC	500 lb
	LHMC	566 lb
AVIONICS	GDC	500 lbs
	BAC	649 lbs 607 lbs average
	MSFC	672 lbs
	LHMC	839 lbs
INFLIGHT LOSSES	GDC	492 lbs 337 lbs average 337/40k lbs = 8.4 lbs/1000 lbs
	BAC	181 lbs

Table 2.4-2. SUMMARY OF ELECTRICAL POWER SUBSYSTEM MASS

	POWER REQ.	UNIT COMPONENTS	MASS	NOTES
GD	1370 to 1490W	2-3.5 KW orbiter cell	160#	0.25 ft^2 modified orbiter cell
	wo manned	2-0.4 KW-hr, 25A Battery	30#	3.5 kW/40# = 87.5 W/#
	module, rendezvous	Support	100	= 40 W/kg
	and docking	Power Control	15	Used main propellant LH_2 & LO_2
		Harness	95	No kWhr or reactant words
			400#	
GRUMMAN	20 missions	3 fuel cells	119 kg	3 fuel cells
	150 to 2796 KWhr nom	AgZn Batteries		
	MANNED	3 tanks	372 kg	Sep.Tanks
		Reactant reg.slope	491 kg(1080#)	
		1086-486/800kWhr		
		= 0.75 kg/kWhr		
BOEING	1170 to 5450W	Fuel cells	160#	Fig. 3.4.2-2 shows schematic
	= 86, 107 Whr	Tanks and Plumbing	76	
	UNMANNED	Battery	28	25Ahr NiH
		Conditioning	156	
		Harness	214	
		Reactants w.reserve,etc	100	
			734#	
				HAC Control Equipment
ELECTRICAL PWR SUBSYSTEM SELECTED		2-3.5kW Orbiter Fuel cells	160#	2.27kg/KW each
		Batteries	30	6.81
		Conditioning	100 2 cu.ft.	6.81(2.2)7 = 105#
		Harness	100	
		Reactants	100	
		Tanks & Plumbing	70 24"D H_z 17" DO_2	
		Supports	70	
			600#	

Table 2.4-3. UNIT AOTV MASSES EMPLOYED

SUBSYSTEM	UNIT MASS	
STRUCTURE		
SHELL	25#/ft of length	A combined 1.04 psf of vehicle surface,
SUPPORT	20#/ft of length	which includes a 10% contingency used after 1st qtr
FUEL TANK	60#/k# H_2	
OXID. TANK	10#/k# O_2	
FLAPS	49#/k#AM	
TPS		
TANK INSUL.	8#k# prop	
SHELL	28#/ft of length	
PROPULSION		
ENGINE	433#	
TVS, PLUMBING	12#/k# prop	
ACS	38#/k#AM	
EPS	600#	
AVIONICS	607#	
INFLT. LOSSES	8.4#/k# prop	
RES & RESIDS.	0.02# prop	
CONTINGENCY	10% is used after the first quarter	

For use in performance calculations during the first and second quarters, mass estimates are made for several vehicles, Tables 2.4-4, 5 and 6. The structural shell and frame supports and the external thermal protection system were estimated based on AOTV length. During the remainder of the study, these weights were estimated based on total AOTV surface area, Table 2.4-7 As the total AOTV surface area decreased with increasing aft frustum angle and longer noses, this approach provided a more representative estimate.

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Table 2.4-4 INITIAL BASELINE AOTV MASS PROPERTIES
GEO DELIVERY 65K STS Lv=60ft

SUBSYSTEM	MASS (LBS)	
STRUCTURE		3682
SHELL	1500	
SUPPORT	1200	
FUEL TANK	384	
OXIDIZER TANK	380	
FLAPS	218	
TPS		2035
TANK INSULATION	355	
SHELL INSULATION	1680	
PROPULSION		1188
ENGINE	433	
TVS, PLUMBING	755	
ACS		376
EPS		600
AVIONICS		607
DRY WEIGHT		8488

Table 2.4-5 INITIAL BASELINE AOTV MASS PROPERTIES
GEO SERVICING 6K UP 2K BACK 65K STS
Lv=60ft

SUBSYSTEM	MASS (LBS)	
STRUCTURE		3756
SHELL	1500	
SUPPORT	1200	
FUEL TANK	396	
OXIDIZER TANK	396	
FLAPS	264	
TPS		2049
TANK INSULATION	369	
SHELL INSULATION	1680	
PROPULSION		1217
ENGINE	433	
TVS, PLUMBING	784	
ACS		456
EPS		600
AVIONICS		607
DRY WEIGHT		8685

Table 2.4-6 INITIAL BASELINE AOTV MASS PROPERTIES
GEO 14K UP & 14K BACK ~100K STS

SUBSYSTEM	MASS (LBS)	
STRUCTURE		5952
SHELL	2308	
SUPPORT	1846	
FUEL TANK	624	
OXIDIZER TANK	624	
FLAPS	550	
TPS		2262
TANK INSULATION	582	
SHELL INSULATION	1680	
PROPULSION		1671
ENGINE	433	
TVS, PLUMBING	1238	
ACS		950
EPS		600
AVIONICS		607
DRY WEIGHT		12042

Table 2.4-7. BASELINE AMOTV MASS PROPERTIES
GEO DELIVERY - 65K STS - Lv = 38 FT

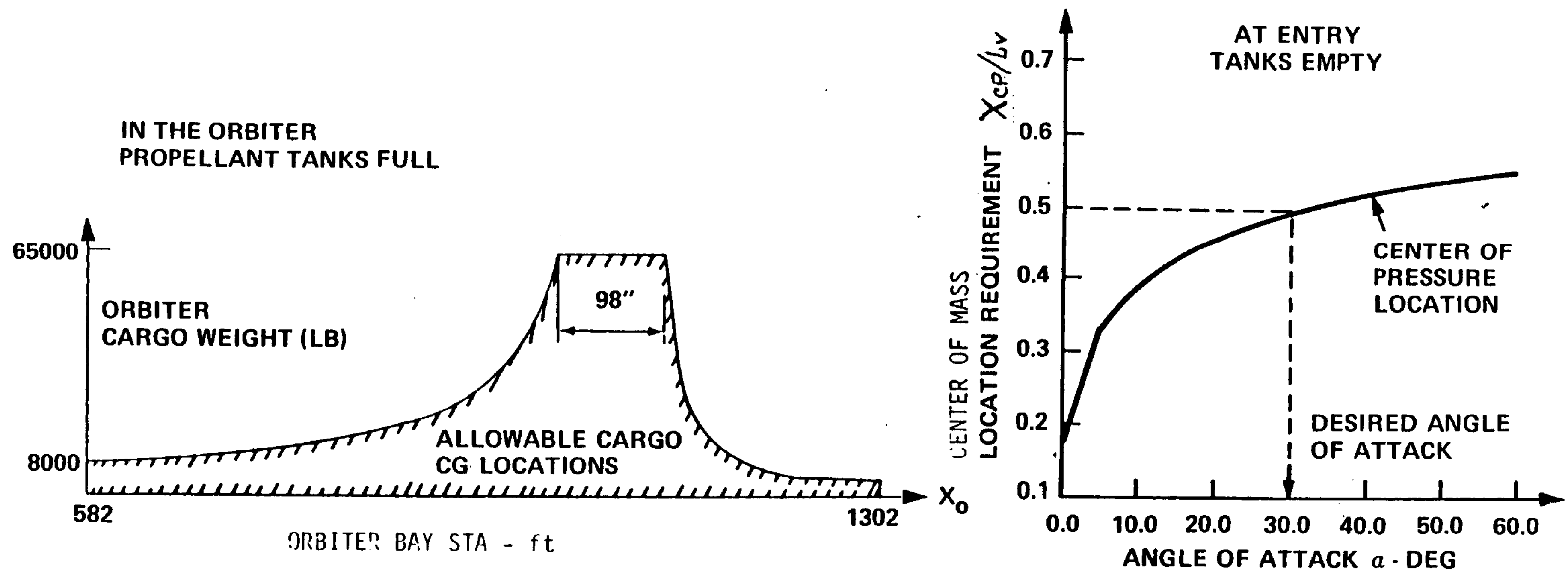
SUBSYSTEM	MASS (LBS)
STRUCTURE	
SHELL & FRAMES	1355
FLAPS	385
FUEL TANK	338
OXIDIZER TANK	395
THERMAL PROTECTION	
TANK INSULATION	361
EXTERNAL TPS	844
PROPULSION	
ENGINE	476
PLUMBING, TVS	542
ATTITUDE CONTROL	305
ELECTRICAL POWER	660
AVIONICS	668
TOTAL DRY MASS (LBS)	6330*

* INCLUDES
10% CONTINGENCY

2.4.2 Initial Configuration Screening to Meet Center-of-Mass Requirements

The aerodynamic configuration selected must, in addition to meeting the external dimensional constraints of the launch vehicle, 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, Figure 2.4-1A, 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, Figure 2.4-1A. 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.

Figure 2.4-1A CENTER OF MASS MANAGEMENT



Two single stage AOTV configurations were evaluated for a GEO delivery mission from a 65K pound orbiter.

Configuration A of Figure 2.4-1B is 45 feet long and configuration B is 60 feet long. Other dimensions are given in the notes below the table. Three arrangements are shown of configuration A and two of B.

Evaluation parameters selected are:

XCM/Lv - Center-of-mass (CM) location is critical for the re-entry mission phase to obtain the desired angle-of-attack. CM forward is better.

Available Payload Bay Length - Although required payload dimensions have not been specified for this study, it is assumed that a larger payload bay is desireable.

Growth to Return Payload - The critical item for this parameter is the effect on the center-of-mass (CM) location for the AOTV during re-entry, i.e., the AOTV configuration, A-1, returning without payload would have satisfactory CM, but with a returning payload in the aft location the CM would be too far aft for re-entry flight. The A-2 shape with the payload forward is conducive to correct CM location.

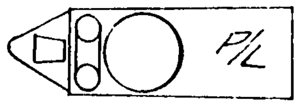
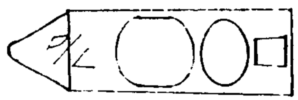
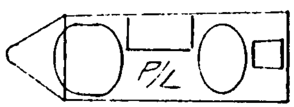
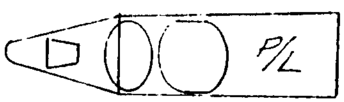
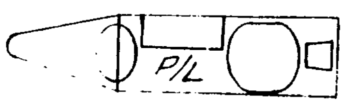
Package in Orbiter for CM Requirement - For this analysis the allowable center-of-mass range for the orbiter with 65K pounds in the bay is compared to the CM of the AOTV. Configuration A-2 is the only one that can be placed in the orbiter bay nose forward. The others require a nose aft position in the orbiter or rearrange the fuel and oxidizer tanks as mentioned.

Nose Clamshell Door - The complication in structure and mechanism of a clamshell door nose on the AOTV to provide for engine firing or payload removal results in a mass penalty.

Side P/L Door - The side payload door structure and mechanism would be an improvement over the nose clamshell, but is still less desirable than the base end insertion and removal.

Manned Capsule can be Added - Here it is assumed that the manned capsule would be added to the payload bay. In the A-1 and B-1 configuration cases the CM location would be too far aft with a returning manned capsule in the payload location.

Figure 24-1B EVALUATION OF INTERNAL TANKED AOTV CONFIGURATIONS
- DELIVERY MISSION -
65K STS

CONFIGURATION	XCM/Lv	P/L BAY LENGTH	GROWTH TO RETURN P/L	PKG IN ORBITER FOR CM REQMT	NOSE DOOR	CLAMSHELL REQD	SIDE P/L DOOR	MANNED CAPSULE CAN BE ADDED.
	.464	14.5'	NO	NOSE AFT	X			NO
	.538	13.0'	YES	NOSE FWD	X (FOR P/L)			YES
	.467	10.0'	YES	NOSE AFT OR SWITCH LH.8 LO_2 TANKS			X	YES
	.474	17.5'	NO	NOSE AFT OR SWITCH	X			NO
	.494	14.5'	YES	NOSE AFT OR SWITCH			X	YES

CONFIG.	L_V	L_N	R_N	N	θ_F	R_B
A	45'	7.5'	2'	2.6^0	2^0	7.5'
B	60'	20.0'	2'	1.0^0	2^0	7.5'

This analysis was made to evaluate internal packaging arrangements for a 60 foot long AOTV with an L/D = 1 for a nominal round trip payload of 14,000 pounds to GEO, Figure 2.4-2.

A propellant weight of 71,000 pounds was used. Tank placement and configuration were varied to conserve space in each design. A $\sqrt{2}$ ellipse was used for tank ends, except for the conical/sphere tanks in configurations 1 and 3. Fuel and oxidizer tank locations were varied to obtained a minimum XCM/Lv.

Configurations 1-3 used a rearward firing engine. Configurations 4 and 5 place a forward firing engine in the nose. The latter arrangement is more space efficient but requires an articulated nose that is open for firing and closed for the remainder of the mission.

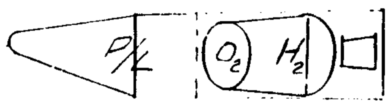
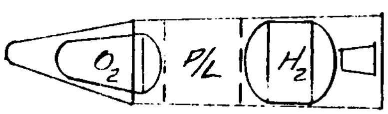
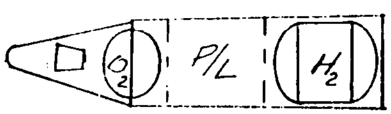
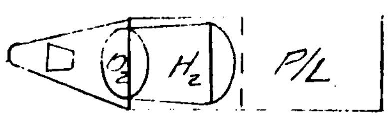
For a preliminary design, XCM/Lv limit of 0.6 was assumed. Configurations 1 and 5 just met this limit of XCM/Lv = 0.6; with number 5's returning payload mass in the center of its bay. Configuration 2 is the clear winner for this requirement with a XCM/Lv = .456. This is done by placing the return payload in the nose. During re-entry flight the propellant mass has been reduced to that required to circularize and rendezvous with the orbiter and the reserves and residuals.

Another mass property criterion for the AOTV is based on the capability of the orbiter to handle the center-of-mass of the 100K pound AOTV in its bay. It was assumed that the allowed CM span of 379 to 538 inches aft in the bay for a 65K pound payload would apply to the 100K pound capacity orbiter.

For example, Configuration 1 has a center-of-mass at its 100K pound weight at 513 inches from the nose. This dimension is within the bay allowable span of 379 and 538. Hence the AOTV can be placed nose forward in the orbiter. Configurations 3-5 have CM's that require nose-aft placement in the orbiter bay.

Payload bay length available for the configurations is listed. The payload bay length of 8 feet for configuration 1 is the shortest. Shape 2 has the longest length of 28 feet but the diameter is reduced by its nose location. Configuration 5 with a payload bay length of 25 feet has the largest volume.

Figure 2.4-2. EVALUATION OF INTERNAL TANKED AOTV CONFIGURATIONS
- MANNED RETURN -
100K STS

	AM CONFIG.	ORBITER CONFIG.			
CONFIGURATION	$X_{CM}{}^{*}/L_v$	$X_{CM}{}^{*}$	$L_v - X_{CM}$	P/L BAY L	NOTES
	0.590	513	NA	8.0'	NOSE FWD IN ORBITER
	0.456	392	NA	28.0'	P/L BAY IS TAPERED. P/L BAY DOOR ARE CLAMSHELL SHAPE.
	0.520	333	387	13.0'	NOSE AFT IN ORBITER.
	0.540	326	394	15.5'	NOSE AFT IN ORBITER. REQUIRES OPENING NOSE.
	0.600	319	401	25.0'	NOSE AFT IN ORBITER. REQUIRES OPENING NOSE BUT SHORTENS PROP LINES.

P/L = 10K L/D = 1.5 L_v = 60' W_{PROP} = 71K $\theta_F = 1^{\circ}$ R_N = 2'

* P/L IN BAY CENTER
379 TO 538 ALLOWED

Satellite designs are driven by significant cost drivers and market opportunities, Figure 2.4-3. One of these is the cost of transport from Earth to LEO. Shuttle launch charges drive satellite designers toward short, 15′ diameter satellites. European marketing opportunities limit some satellites to 12′ diameters (for Ariane compatibility). Both 12′ and 125′ satellite diameters exceed the internal payload bay capacity of a very long (60′) biconic AOTV with a 2^{o} half cone angle on the aft frustum. Consequently, for nearly all biconic vehicles, some provision will have to be made to attach payloads to the exterior of the AOTV, since most payloads will not fit inside. Obviously, half cone angles $<2^{o}$ should be evaluated.

Figure 2.4-3 SOME AOTV REQUIREMENTS – PAYLOADS

FULL SIZE PAYLOADS

- SATELLITE DESIGNERS MINIMIZE LAUNCH COSTS
  - FULL 15 FT DIAMETER MINIMIZES LAUNCH LENGTH
- SOME MISSIONS REQUIRE LONG (> 30 FT) SATELLITES
- INTERNAL PAYLOAD CAPABILITY OF 60 Ft LONG AOTV
  - MAX DIA 11 TO 13 FT
  - MAX LENGTH 12 TO 21 FT

CONCLUSION

AOTV MUST HAVE CAPABILITY OF SUPPORTING PAYLOADS OUTSIDE OF AEROSHELL

Figure 2-4.4 SATELLITE RETRIEVAL

- NO SCIENTIFIC OR COMMERCIAL REQUIREMENT TO RETURN A SATELLITE TO LEO FROM HEO
- MILITARY REQUIREMENTS UNKNOWN

CONCLUSIONS

1 PAYLOAD SPACE WITHIN AEROSHELL MUST ONLY ACCOMMODATE

A) SAT SERVICING EQUIPMENT WHICH IS RETURNED TO LEO

B) MANNED CAPSULE FOR SAT SERVICING

2 PAYLOAD SPACE LARGER THAN NEEDED BY IA & 1B) IS EXPENSIVE (SHORT VEHICLES ARE CHEAPER)

The issue has been addressed of whether a need exists for a payload bay within an AOTV, Figure 2.4-4. Since external payload provisions must be provided, payloads on the way out (LEO to GEO) do not require an internal payload bay. An internal payload bay is required of any payload which must survive the severe heating developed during atmospheric entry near the end of an AOTV mission.

The cost of returning a payload from a High Energy Orbiter (like GEO) is quite substantial. Principly, this cost is the transportation cost of delivering sufficient propellant (to LEO and then to GEO) to visit and de-orbit the retrieved satellite. If the AOTV mission on a particular flight is only satellite retrieval, the total cost of retrieval approximates the cost of 1 STS launch (~$84M) for long, large diameter AOTVs which are capable of retrieval. This $84M far exceeds the cost of most commercial satellites to date. A commercial need for a refurbished satellite would be better met by building a new satellite than by expending $84M for retrieval. This cost burden is also considered excessive for any scientified satellites which may be in a HEO.

Much lower retrieval costs are possible if the AOTV performs a satellite delivery mission in the early part of a retrieval flight. The propellant necessary for retrieval (e.g., 2400 lb to return a 6000 lb satellite) occupies space (and weight) that another paying customer (satellite) could use. Since a long, large diameter AOTV that is capable of retrieval leaves only a small amount of payload bay length for payloads, the 2400 lb of propellant occupies a place of one out of 2 or 3 payloads. Consequently, the mission cost of retrieval approximates the los income from one of two, or one of three, payloads which would have shared the total launch cost. This implies a minimum cost of retrieval of $84M ÷ 3 = $28M, when the retrieval mission is combined with a satellite delivery mission. This cost is also judged excessive for a commercial venture.

On those rare occasions when satellite retrieval is required, the mission can be flown much less expensively by using a short AOTV (without an internal payload bay) in an all propulsive mode. The additional propellant capacity needed for this mission can be stored in external tanks which can be re-used on other missions.

The only payload that must be retrieved, and survive the atmospheric maneuvering phase of a mission, is a crew capsule. However, the requirements of manned missions may drive a biconic AOTV so uniquely that a manned mission capable vehicle may be unsuited for the competitive mission of delivery of commercial satellites. Consequently, our AOTV designs for the satellite delivery missions will not contain internal payload bays.

Biconic AOTVs fly at fixed angles of attack in pitch. To maximize vehicle maneuverability, it is desirable to fly at the angle of attack which maximizes the ratio of Lift/Drag (L/D). To obtain this, the vehicle center of mass (CM) must coincide with the vehicle center of pressure (Cp).

Some amount of control of the Cp location is possible by using flaps to extend the vehicle on the windward side. However, CM variations can be very substantial. If a single vehicle is to perform all AOTV missions (a "universal AOTV"), the entry weight could vary from 10,000 lb to 24,000 lb. To enable such a substantial weight change without altering the AOTV's aerodynamic characteristics, it is recommended that the payload CM be located at the AOTV CM, which is at the same location (along the vehicle length) as the AOTV's Cp, Figure 2.4-5.

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Figure 2.4-5 CHARACTERISTICS OF A UNIVERSAL AOTV

- UNIVERSAL = ONE VEHICLE FOR ALL TASKS
- HYPERSONIC FLIGHT REQUIREMENTS
  - FLY AT FIXED ANGLE OF ATTACK (IN PITCH) FOR MAX L/D
  - CENTER OF PRESSURE (CP) AND VEHICLE MASS CENTER (CM) MUST COINCIDE
- CP FIXED BY EXTERNAL SHAPE
  - BODY FLAPS PROVIDE "FINE TUNING" OF CP LOCATION
  - "HISTORICAL" CP IS AT 55% OF VEHICLE LENGTH (MEASURED FROM NOSE)
- LARGE WEIGHT VARIATION FOR AOTV MISSIONS
  - EMPTY VEHICLE WEIGHT AT RE-ENTRY ~ 10,000 LB
  - UNMANNED SERVICING MISSION WEIGHT AT RE-ENTRY ~ 12,000 LB
  - MANNED SERVICING MISSION WEIGHT AT RE-ENTRY ~ 24,000 LB (SOW PAYLOAD WEIGHT)

CONCLUSION:

RETURN PAYLOAD CG SHOULD BE AT VEHICLE CM WHICH IS AT C_p

Some characteristics of a possible universal biconic AOTV are outlined in Figures 2.4-6, 7 and 8. The payload bays on both the single engine and two engine versions are adequate to contain a Bare Bones crew capsule. External payload attachment is possible on the aft end "docking ring". Internal payload capability is also available. The engines fire in the forward direction, after a deployable nose segment has been rotated out of the way.

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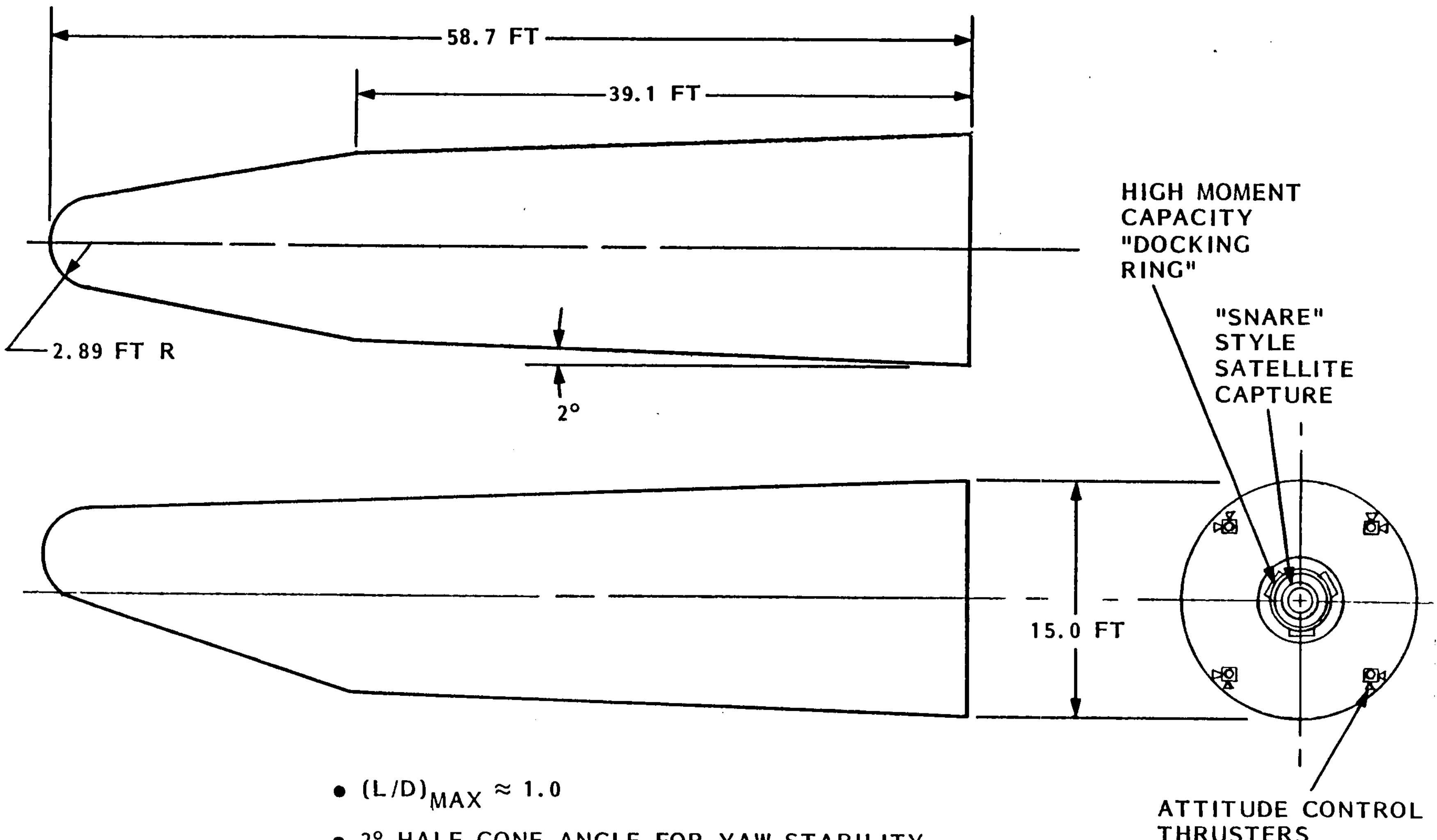
Figure 2.4-6 MODIFIED AMOOS 5B CONFIGURATION



- $(L/D)_{MAX} \approx 1.0$
- 2° HALF CONE ANGLE FOR YAW STABILITY

Figure 2.4-7 AVAILABLE PAYLOAD BAY: SINGLE ENGINE

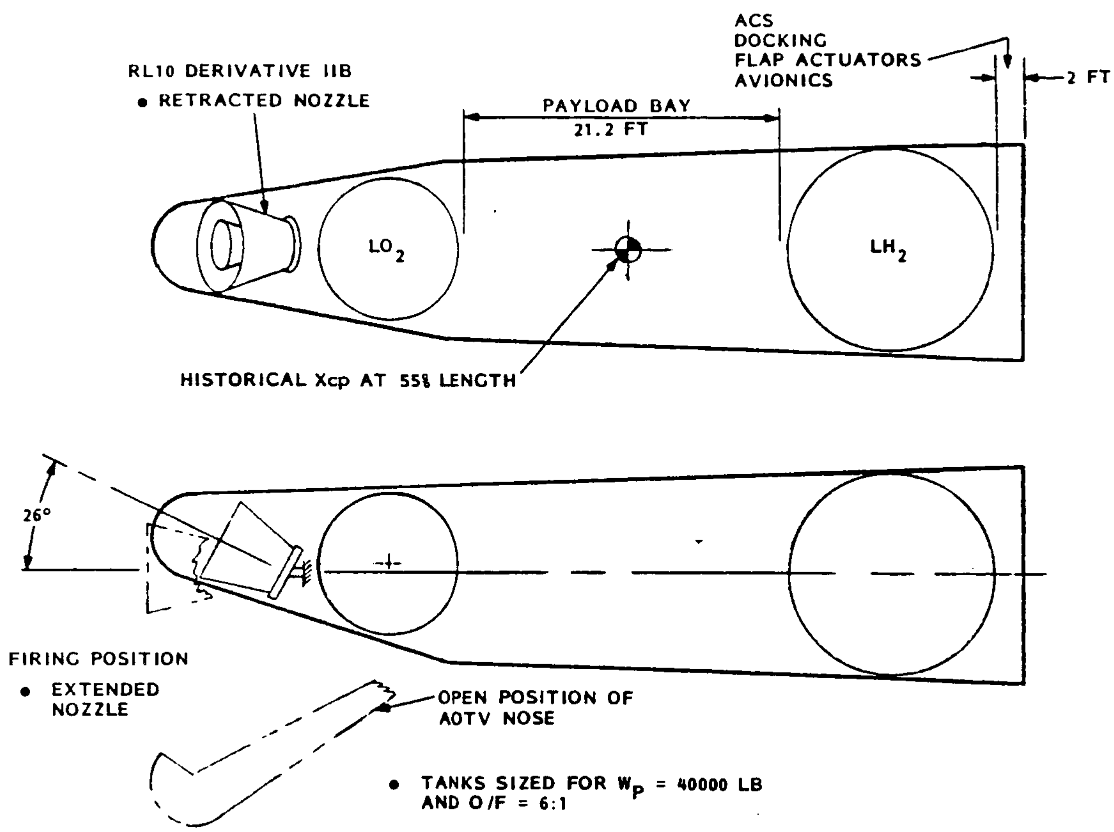
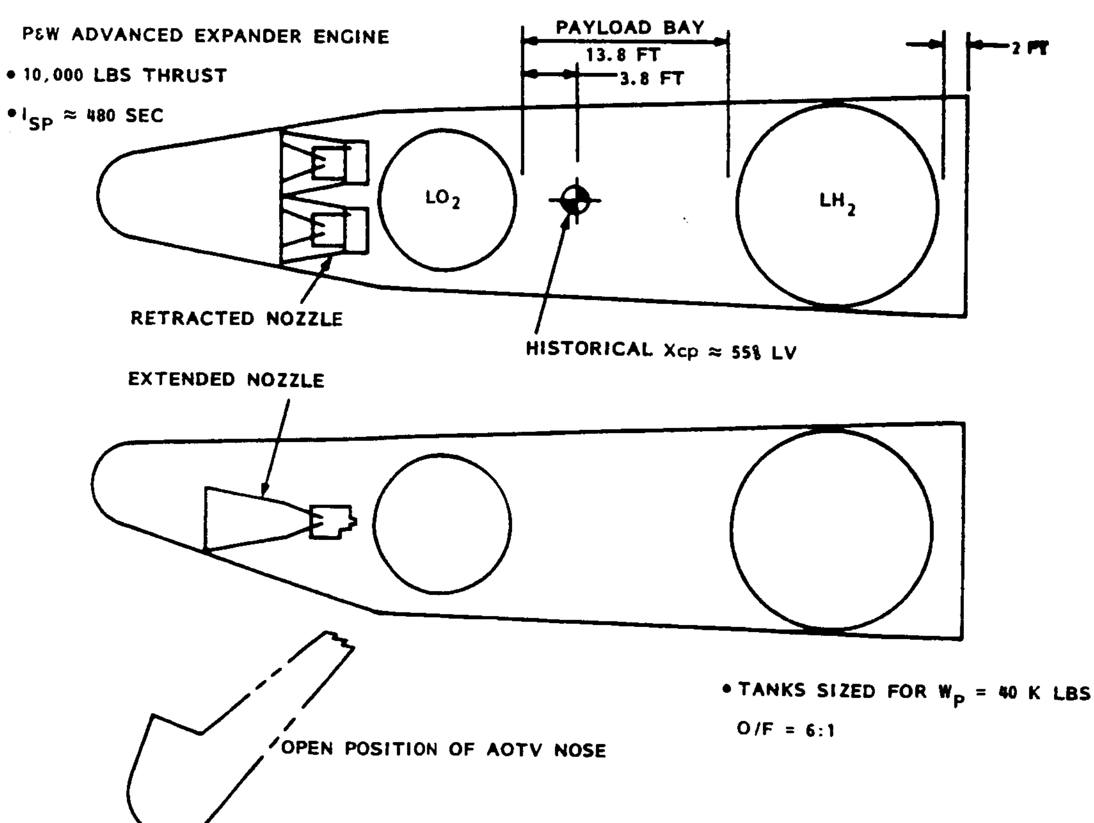


Figure 2.4-8 AVAILABLE PAYLOAD BAY: TWO ENGINES



Some of the advantages of short, ground based AOTVs are outlined in Figure 2.4-9. For example, a 3 foot length reduction will lower Shuttle launch charges by $300M in five years.

Figure 2.4-9 **BI-CONIC AOTV: SHORT IS BETTER**

- SHORT VEHICLES ARE LIGHTER BECAUSE BI-CONIC AOTVs ARE ENCLOSED BY AEROSHELL (TPS + STRUCTURE)
  - FOR 14-1/2′ ϕ, $\approx$ 80 LB/FT
  - FOR 7-1/2′ ϕ, $\approx$ 40 LB/FT
- BETTER PACKAGING WITHIN ORBITER PAYLOAD BAY
  - MORE ROOM FOR $LH_2 \rightarrow$ LOWER O/F RATIO $\rightarrow$ HIGHER Isp
  - MORE FLEXIBILITY IN MEETING LAUNCH CG REQUIREMENTS
- REDUCED LAUNCH COSTS
  - $\approx$ \$2 M/FT PER FLIGHT IN 1990s
  - FOR 3 FT OFF, 10 FLTS/YEAR, 5 YR – \$300 M.

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Even with space basing of OTVs, short AOTVs still offer economic advantages when they are used in a ground based mode, Figure 2.4-10.

A single space base at 28° inclination implies that high inclination mission will be ground based. Also, prior to total storage of a space based AOTV at the Space Station, a number of ground launched AOTV flights will occur while the technology of operating in a space based mode is being developed and verified. As many as 50 ground based flights may be required while the Space Station learns how to perform payload manifesting and transfer, propellant manifesting, storage and transfer, and AOTV inspection.

Figure 2.4-10 FURTHER ARGUMENTS FOR A SHORT AOTV

- SHORT AOTVs REDUCE LAUNCH COSTS FOR MANY STS FLIGHTS
- THERE MAY NOT BE A SPACE BASE (= SPACE STATION)
- A SPACE BASE MAY EXIST ONLY AT ONE ORBIT ($i = 28°$)
  - MANY ORBITS MAY REQUIRE ACCESS FROM A GROUND BASED LAUNCH (ORBITER TO $i = 58°$, ETC).
- FOR USE OF A SPACE BASE @ $i = 28°$ (GEO DELIVERY), MANY (≈ 50) AOTV TRANSPORT BY ORBITER WILL BE REQD FOR A VEHICLE THAT IS DESIGNED TO BE SPACE STORED.

AOTV LENGTH
WITHIN THE ORBITER
PAYLOAD BAY IS AN IMPORTANT
ECONOMIC
CONSIDERATION

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2.4.3 Unmanned/Manned Vehicles Schedule Implications

Past OTV studies have planned on a 4 to 5 year period of unmanned AOTV flights before starting operation with a manned AOTV, Figure 2.4-11. Present expectations are for a manned space station to be operational several years before the first new reuseable OTV becomes operational. With a significant manned presence in space, manned OTV missions may be desired immediately, or shortly after, a new reuseable OTV becomes operational. Consequently, the initial AOTV may be a manned (or man rated) vehicle, Figure 2.4-12.

Figure 2.4-11 **SOME AOTV REQUIREMENTS: SCHEDULE IMPLICATIONS**

PAST STUDIES

BOEING 1980
- ABOTV IOC 1986-87
- MANNED ABOTV IOC 1991-92

5 YEARS OF UNMANNED FLIGHTS

EVOLUTIONARY VEHICLE DEV'T

GENERAL DYNAMICS 1980
- OTV IOC 1988
- MANNED OTV IOC 1992
- AOTV IOC? (1992)

4 YEARS OF UNMANNED FLIGHTS

Figure 2.4-12 PRESENT EXPECTATIONS:

- SPACE STATION IOC 1990 - 91
  - PEOPLE CONTINUOUSLY IN SPACE DOING WORK
- UNMANNED AOTV IOC 1992 - 93
- MANNED AOTV IOC 1993 - 94?

→ TIME TO PROVE SAFETY OF AOTV

  - MISSIONS ARE UNCERTAIN, TIMING IS UNCERTAIN
    - SERVICE COMMERCIAL AND/OR GOVERNMENT SATELLITES
    - SERVICE NATIONAL ASSETS (SBR, LARGE PLATFORMS, ETC.)
    - CONSTRUCT/ASSEMBLE NATIONAL ASSETS

CONCLUSIONS:

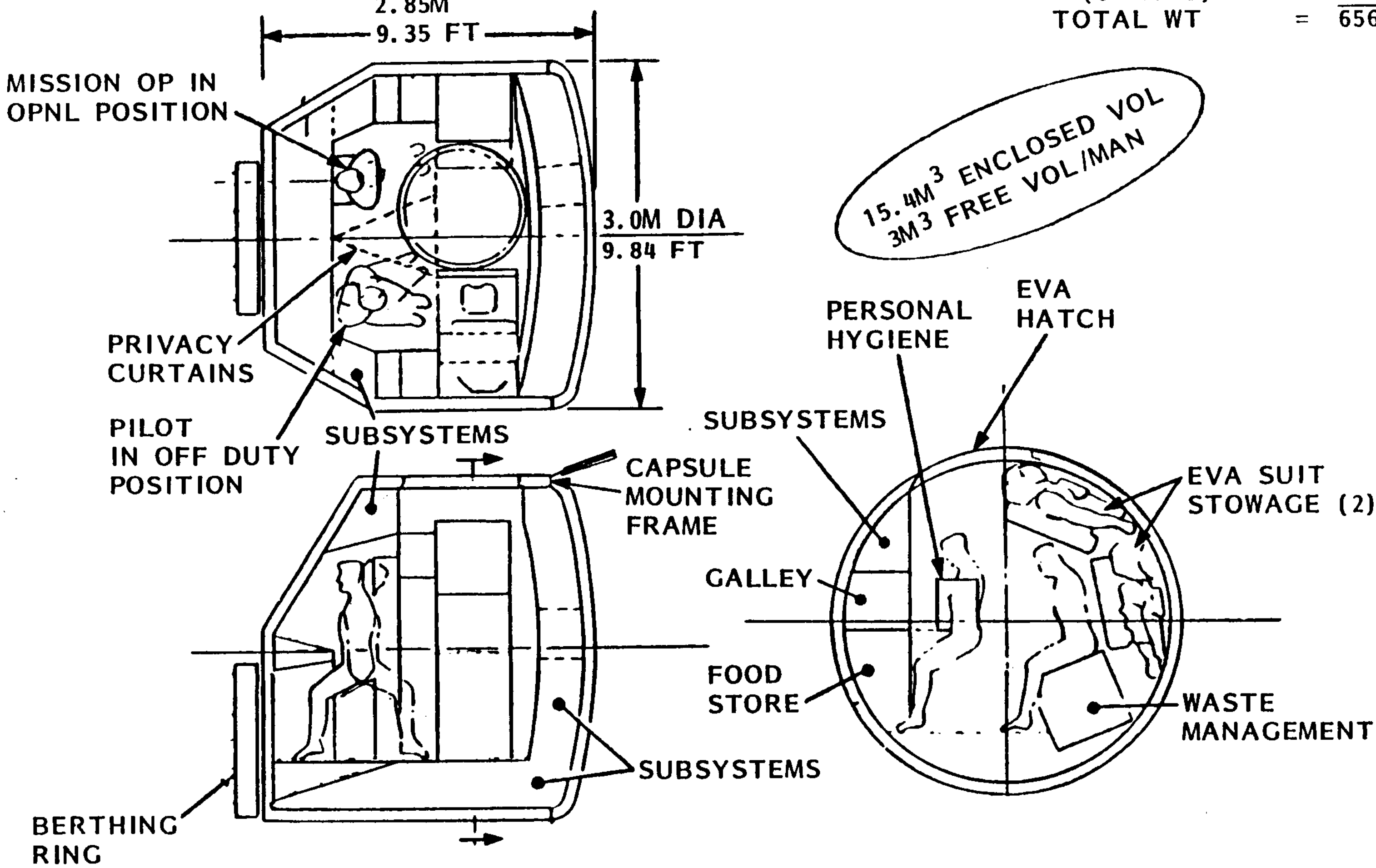
1. INITIAL AOTV MAY BE MAN RATED AND INITIALLY FLOWN UNMANNED
2. DEFINE ONE AOTV FOR ALL MISSIONS
3. EXAMINE EVOLUTIONARY EXPANSION OF AOTV CAPABILITY WITH DIFFERENT VEHICLES

2.4.4 Manned Cabin Definition

The crew capsule which was chosen as the baseline for Manned Orbit Transfer Vehicles by Grumman/Johnson Space Center during the 1978 to 1982 study called "Manned Geosynchronous Mission Requirements and Systems Analysis Study" is illustrated in Figure 2.4-13. The capsule is intended to adequately sustain a crew of 2 for missions which last as long as 30 days.

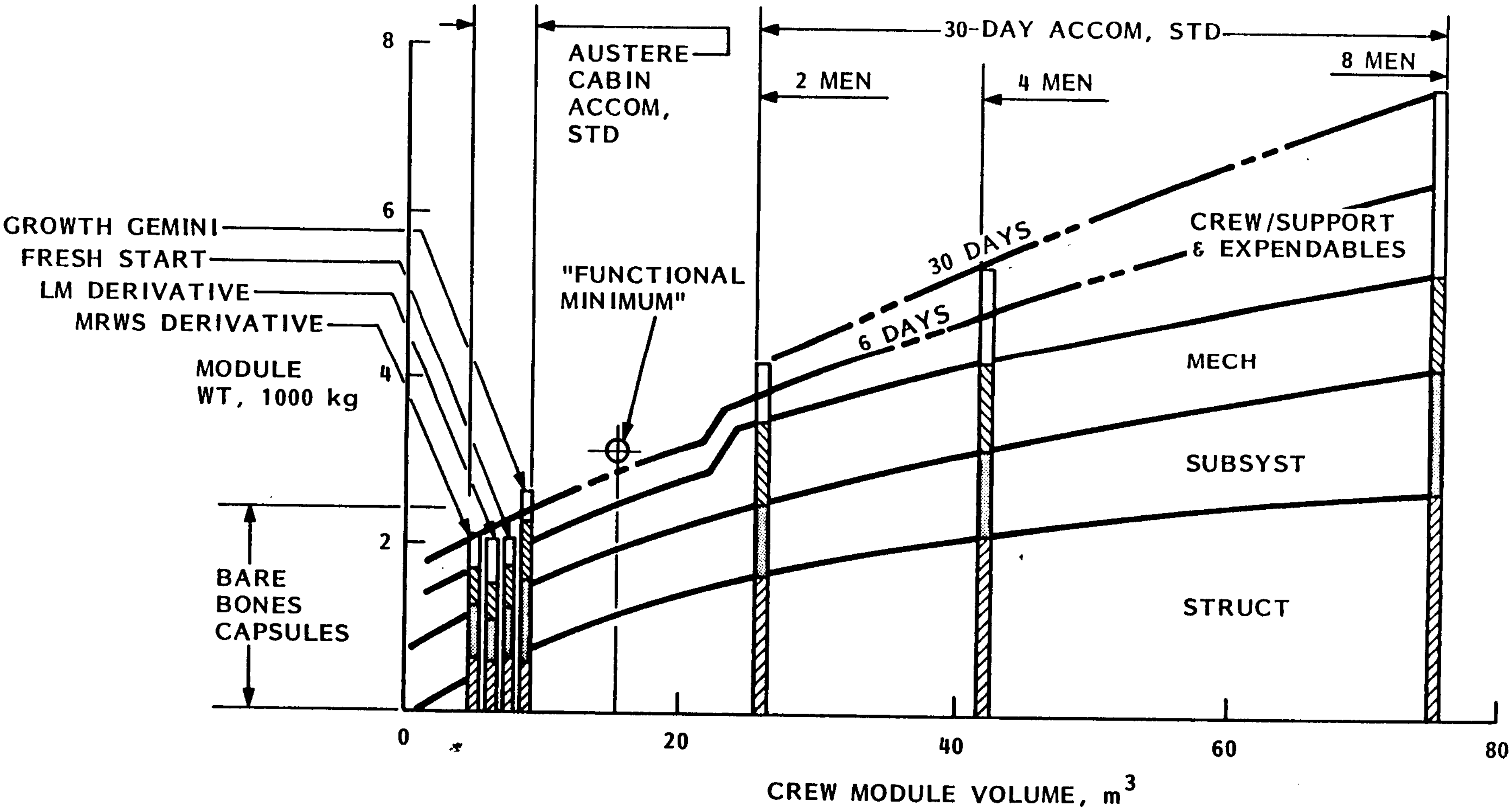
Figure 2.4-13 FINAL BASELINE MOTV CREW CAPSULE "FUNCTIONAL MINIMUM"

DRY WEIGHT	=	6118 LB
CREW (2)	=	360 LB
CONSUMABLES (6 DAYS)	=	78 LB
TOTAL WT	=	6560 LB



The weight of crew capsules (subsystems and total weight) as a function of crew module volume is illustrated in Figure 2.4-14. Four austere cabins are shown which weigh a little more than 200 kg (4400 lb) each. The four "Bare Bones" cabins have virtually the same crew module volume, but are shown spread out along the abscissa for clarity. These four "Bare Bones" cabins were designed during the MOTV study in 1978.

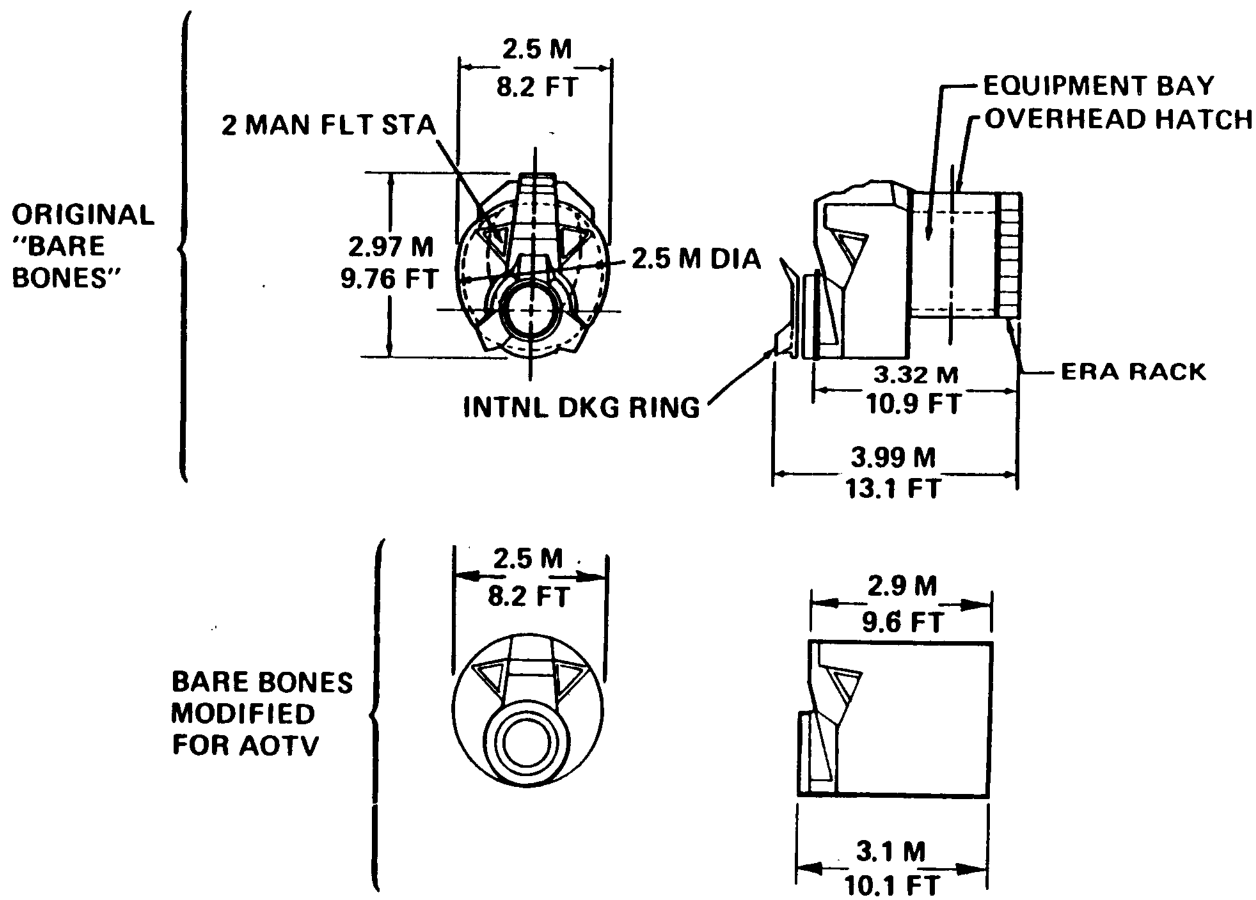
Figure 2.4-14 PRELIMINARY CREW CAPSULE VOLUME/WEIGHT SUMMARY*



*REFERENCE: MANNED GEOSYNCHRONOUS MISSION REQUIREMENTS AND SYSTEMS ANALYSIS STUDY, MID-TERM REVIEW-PHASE 1", FEB 15, 1978, GAC

The "Bare Bones" capsule which has been selected for AOTV use is the design which was derived from the LM Ascent Stage. This configuration is shown on the upper portion of Figure 2.4-15. The general arrangement has been modified (as shown in the lower part of the figure) for AOTV studies by incorporating the same volume into the maximum diameter of the original LM derivative. This change reduced the length of the capsule by 3', permitting easier packaging within an AOTV.

Figure 2.4-15 MODIFIED LM DERIVATIVE BARE BONES CABIN



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The weight statement shown in Figure 2.4-16 compares an actual weight, which was measured on Lunar Module 12's Ascent Stage, with the weight which is designated for a Bare Bones Crew Capsule. Weight peculiar to the Lunar Mission has not been included within Bare Bones. Some of the LM12 subsystem weight (avionics and electrical power supply) has been divided between Bare Bones and AOTV Avionics.

It is believed that the total capsule weight of 3850 lb is conservative when compared with most preliminary design weight estimates. One reason for this belief is that the weights are based upon the measured weights of an actual vehicle. Another reason is the 1960s technology level of the reference vehicle. Significant weight reductions have been made in structures, avionics, and other subsystems since the 1960s.

Figure 2.4-16

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DERIVATION OF BARE BONES CREW CAPSULE WEIGHT FROM LM ASCENT STAGE

LM 12 ASCENT STAGE		BARE BONES	CAPSULE	AOTV AVIONICS
	(LB)		(LB)	(LB)
STRUCT & MECHN	1406	LESS TANKS ETC	1100	
STAB & CONT	189	20% IN CAPSULE	38	151
NAV & GUIDANCE	191	40% IN CAPSULE	76	115
CREW PROVISIONS	175	LESS LUNAR EQUIP	146	
ECLS	297		297	
ILLUMINATION (INSTRUMENTATION)	132		132	
EPS	737	40% IN CAPSULE	295	442
PROPULSION	469		---	
REACTION CONTROL	242		---	
COMMUNICATION	114	50% IN CAPSULE	57	57
CONT. & DISPLAYS	232		232	
EXPLOSIVES	29		---	
MANUFACTURERS VARIATION	(-60)			
	4153 LB		2373 LB	765 (REF)
KSC INSTALLED CREW EQUIP	390	LESS LUNAR EQUIP	334	
FOOD	8		36	
LIQUIDS & GASSES		50% IN CAPSULE	68	
PROPELLANTS-NON TANKED	41		---	
INTERNATIONAL DOCKING RING	---		931	
CONTINGENCY	---		187	
	4728		3929	
CREW & SUITS	---		576	
	4728 LB	BARE BONES TOTAL SHOWN ON CHART	4505 LB	
	=(2145 KG)		(2043 KG)	
		BARE BONES FOR AOTV (- DOCKING RING)	~ 3500 LB	

The MOTV study (7) identified seven different types of manned missions at GEO. For each mission (several for each mission type), they identified the weight of the "on orbit mission equipment" (tools, manipulators, grapplers, etc.) which is needed to perform that mission task. For representative missions (from each mission type), these weights are shown in Figure 2.4-17. The average value of these weights is approximately 1320 lb. This weight will be used for all AOTV manned missions.

Figure 2.4-17

ON - ORBIT MISSION EQUIPMENT WEIGHT FOR MANNED MISSIONS

- REFERENCE: "MOTV CAPABILITIES HANDBOOK & USER GUIDE" GRUMMAN AEROSPACE CORPORATION, FEB 1981
  - SECTION 3, "POTENTIAL APPLICATIONS", FIG. 3-8
- SEVEN TYPES OF MISSIONS ARE EXAMINED
- FOR AOTV – ONE REPRESENTATIVE MISSION WAS SELECTED FOR EACH MISSION TYPE
  - AVERAGE WEIGHT FOR 7 MISSIONS SELECTED

INSPECTION 1	1090 LB
SERVICE 2	1320
EMERGENCY REPAIR 1	1180
RETRIEVAL 1	1290
OPER LRG SYST 1	1070
DEBRIS REMOVAL 1	1450
CONSTRUCTION 2	1830

$\Sigma = 9220$ LB

MEAN VALUE = 1317 LB

- CHOOSE FOR AOTV MISSIONS ⟶ 1320 LB

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The Statement of Work (SOW) for this study stipulates that manned missions to GEO shall consider a total payload of 14,000 lb to and from GEO. The arguments presented in Figure 2.4-18 indicates that this requirement imposes a length (and weight) lenalty for biconic AOTVs.

Both candidate crew capsules, and their associated hardware and equipment, weigh substantially less than 14,000 lb. The "Functional Minimum" is about 9400 lb ($\sum$T1) while the "Bare Bones" is about 6400 lb ($\sum$T2). The differences between these weights and 14,000 lb (ΔWT1 and ΔWT2) could be used to transport avionic components to GEO. If these avionic components were stored internally on the AOTV, they would add about 3/4' (for "Functional Minimum") or 1-1/4' (for "Bare Bones") to an AOTV's length. This corresponds to about 60 lb or 100 lb additional weight to the AOTV. These length and weight penalties will exist for all missions.

Figure 2.4-18 MANNED CAPSULE & AOTV VOLUME

MISSION WEIGHTS

- MISSION HARDWARE 1000 LB
- ON-ORBIT MISSION EQUIPMENT 1800 LB

$\left.\begin{matrix}1000 \text{ LB} \\ 1800 \text{ LB}\end{matrix}\right\} \sum_{MH\&E} = 2800 \text{ LB}$

- "FUNCTIONAL MINIMUM" CAPSULE $+ \sum_{MH\&E} = 9360$ LB $= \sum_{T1}$
- "BARE BONES" CAPSULE (- DOCKING RING = 3580 LB) = 6380 LB $= \sum_{T2}$
- SOW WEIGHT FOR MANNED FLT TO AND FROM GEO = 14000 LB

IMPLICATIONS FOR AOTV VOLUME

- ΔWEIGHT = SOW $- \sum_T$ = MODULES AND PROPELLANTS
  - ΔW_{T1} = 4640 LB
  - ΔW_{T2} = 7620 LB
- IF REQUIRED TO RETURN 14000 LB, ALL ΔW_T = MODULES
  - TYP DENSITY OF ELECTRONICS = 0.04 LB/IN.^{3}
  - PACKAGING EFFICIENCY IN AOTV ~ 75%
  - IMPLIED INCREASE IN AOTV SIZE TO RETURN ΔW_T INTO ATMOSPHERE
    - "FUNCTIONAL MINIMUM" VEHICLE ➡ 89.5 FT^3 (0.73 FT @ 12.5 FT DIA)
    - "BARE BONES" VEHICLE ➡ 147 FT^3 (1.2 FT @ 12.5 FT DIA)

If the objects which are represented by $\Delta WT1$ and $\Delta WT2$ in Figure 2.4-18 are stored external to an AOTV, they would not impose a vehicle penalty. Since there is no economic need to return to earth (from GEO) discarded or spent objects which have been replaced by the new ΔWT objects, there appears to be no need to return 14,000 lb from GEO. The only objects that must return are the crew capsule, and the tools used by the crew. Spent or discarded objects can be placed in a debris orbit above GEO or placed into a burn up orbit just before the AOTV enters the atmosphere.

Since short, low weight AOTVs offer the most attractive opportunities for low cost operation, we recommend that biconic AOTVs not be required to contain large quantities of spent or discarded objects within their aeroshells. Also, we recommend that the smaller of the two candidate crew capsules, the "Bare Bones", be chosen as the baseline crew capsule for all AOTV manned missions, Figure 2.4-19.

Figure 2.4-19 MANNED CAPSULE AND AOTV VOLUME (CONT)

ALTERNATIVE MODE OF OPERATION

- STORE NEW AND SPENT MODULES EXTERNAL TO AOTV AEROSHELL
- DISCARD SPENT MODULES (& STORAGE RACK) INTO "BURN-UP" ORBIT BEFORE AOTV ENTERS ATMOSPHERE

RECOMMENDATIONS

- SELECT "BARE BONES" CAPSULE AS BASELINE FOR ALL AOTV STUDIES
- DO NOT REQUIRE UNIVERSAL AOTV TO CONTAIN LARGE QUANTITIES OF SPENT MODULES WITHIN AEROSHELL

REDUCTION OF AOTV LENGTH & WEIGHT IMPROVES PERFORMANCE ON MOST MISSIONS (GEO DELIVERY ONLY)

- IMPROVES COMPETITIVE POSITION WITH RESPECT TO APOTV

2.4.5 Small Manned AOTV

The major performance characteristics of an AOTV which is capable of bringing people to Geosynchronous orbit, Figure 2.4-21, from the earth, on a single 65K payload shuttle are presented in Figure 2.4-20.

For this mission, 3700 pounds are available for ASE, although Figure TBD indicates that only 2900 lb are required. AOTV "H-1M" (or "H-1") delivers to GEO, and back to LEO, a crew of 2 and all the tools necessary to perform satellite servicing tasks at GEO. Additionally, it delivers 2100 lb of payloads which are left at GEO. These payloads are propellants and replacement avionics modules which refurbish the serviced satellites. None of this payload weight is returned by H-1M.

This AOTV uses advanced LOX/Hydrogen engines with a specific impulse of 479 seconds at an oxidizer/fuel ratio of 6.5:1. It performs its mission using a single perigee burn Hohmann transfer from LEO to GEO, with a thrust/weight at ignition of .2. On its return from GEO ($\Delta V3$), only 13.5^{o} of plane change is performed propulsively. An additional 15^{o} of plane change is obtained by maneuvering during atmospheric deceleration. To perform this aerodynamic plane change, a real L/D = 1.0 is required. H-1M has a Newtonian L/D - 1.35, which is estimated to degrade to 1.20 in an actual interaction between a hypersonic airstream and the vehicle. The fourth significant propulsive burn ($\Delta V4$) of the mission provides circularization and rendezvous (with orbiter) after H-1M has lofted out of the atmosphere in a low eccentricity ellipse.

H-1M is not offered as an optimized vehicle design. Rather, it is representative of a class of vehicles that can perform a number of very attractive missions at a cost of 1/3 to 1/2 of previous estimates. H-1M is the first vehicle designed in this study with an external hydrogen tank (H). The designation "M" denotes a manned vehicles.

Figure 2.4-29

SMALL MANNED AOTV "H1M" PERFORMANCE ON MANNED GEO ROUND TRIP

LEO CAPABILITY	65,000 LB	
AIRBORNE SUPPORT EQUIPMENT (ASE)	3,700 LB LIMIT	
AOTV DRY WEIGHT (INCLUDING CREW & EQUIP)	14,200 LB	
PROPELLANT (USEABLE)	43,500	= 45,000 LB
RESERVES & RESIDUALS	900	
INFLIGHT LOSSES	600	
PAYLOAD DELIVERED	2,100 LB	
PAYLOAD RETURNED	0	
AMOTV ENTRY MASS	14,560 LB	
I_{SP}	479.1 SEC	

Δ V BUDGET: ΔV_1 = 8140 FPS
ΔV_2 = 6000 FPS
ΔV_3 = 5200 FPS (Δi = 15° FROM AERO)
ΔV_4 = 300 FPS

- NEWTONIAN $(L/D)_{MAX} \approx 1.35$
- O/F = 6.5:1, EXTERNAL H_2 TANK

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H-1M configured for orbital operations is illustrated in Figure 2.4-21. One half of the aeroshell (over the aft cone frustum) has been rotated 180° to its open position. The crew capsule has been rotated 90° (about a pivot on the bottom of the mid-frustum bulkhead) to a position which permits easy crew access to a worksite. The on-orbit mission equipment shown consists of two dextrous arms and a grappler (to hold on to a worksite). The 2100 lb payload ("AOTV payload") is shown in front of the crew hatch. The payload retainer is rotated to one side to allow crew egress/ingress. A single mission throw-away hydrogen tank is shown at the forward end of H-1M. This tank is placed in a burn up orbit, and the deployable nose is closed, prior to H-1M's entry into the atmosphere. A small amount of internal hydrogen storage is provided to allow the final burn ($\Delta V4$) after leaving the atmosphere.

The vehicle is shown in a cold soak orientation - the solar vector is perpendicular to the aft bulkhead, which casts a shadow over the entire vehicle. This offers two advantages: boil off of liquid hydrogen is minimized and the temperature of the vehicle's thermal protection system (TPS) can be reduced to $-100^{\circ}F$ prior to atmospheric entry. Low aeroshell TPS temperatures at start of heating permit a 20% reduction in aeroshell TPS weight.

H-1M contains an internal, reuseable liquid oxygen tank and 4 Aerojet advanced expander cycle engines. Hydrogen is piped from the foward bulkhead to a manifold attached to the mid-frustum bulkhead. The manifold distributes flow entry to the engines. The high expansion ratio engines (ϵ = 1000:1) have fixed nozzles and are fixed (i.e., non-gimballed) to the vehicle. Thrust vector control is obtained by throttling back on various engines. The thrust eccentricity developed by a single engine failure condition (at GEO) can be compensated by shutting down the diametrically opposite engine. Slow roll rotation of H-1M about its centerline will compensate for small center of mass (CM) offsets from the plane containing the thrust vectors of the two operating engines. To safety return a crew after sustaining two engine failures, both operating engines will be used. Thus, the lane of engine thrust may be offset 4 feet from the vehicle c.g., To maintain pitch stability, high roll rates are required of H-1M.

Prior to main engine burns, the crew capsule is rotated 90° within the aeroshell so the vertical centerline (which indicates a turntable bearing which provides 1 degree of freedom in rotation of the crew capsule relative to its deployment structure) in the figure becomes parallel, and coincident with, the vehicle centerline. The aeroshell is then closed. When high roll rates are used to stabilize H-1M, the crew is isolated from the centrifugal effects by the turntable bearing between the capsule and its deployment structure.

Isolating the crew from spin induced forces, and draining propellants from a spinning vehicle, ay prove unattractive after some analyses. Consequently, H-1M can be built with an additional pair of engines on the horizontal (in the figure) centerline. These would preserve the option of shutting down diametrically opposed engines after sustaining two engines failures. The resulting cg offsets will then be small (and comparable to a single engine out with a four engine configuration).

C-2

SMALL MANNED AOTV "H-1M"

Figure 2.4-21

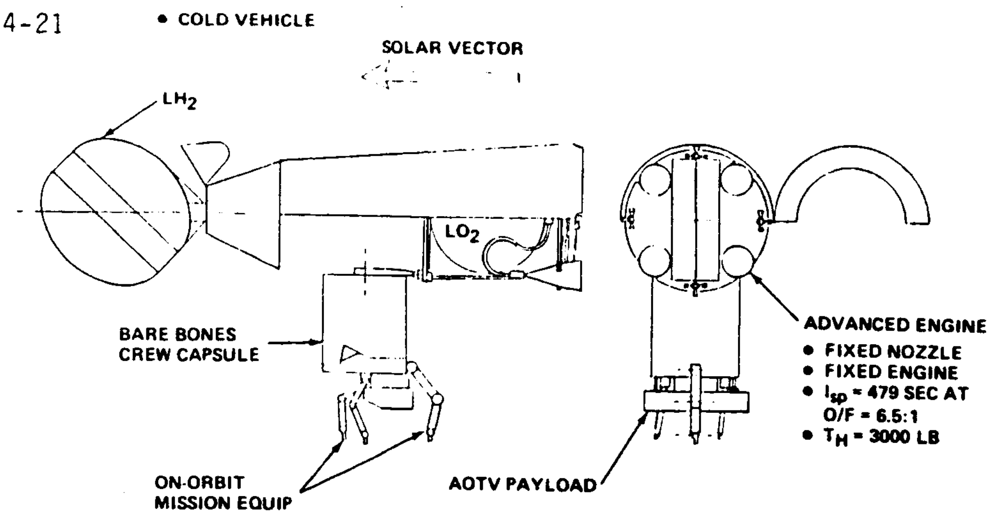


Figure 2.4-22

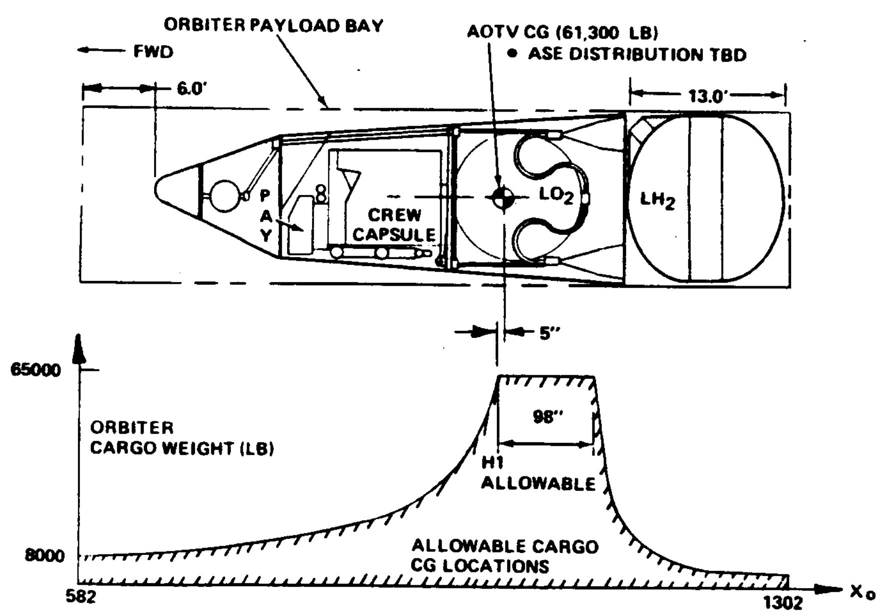
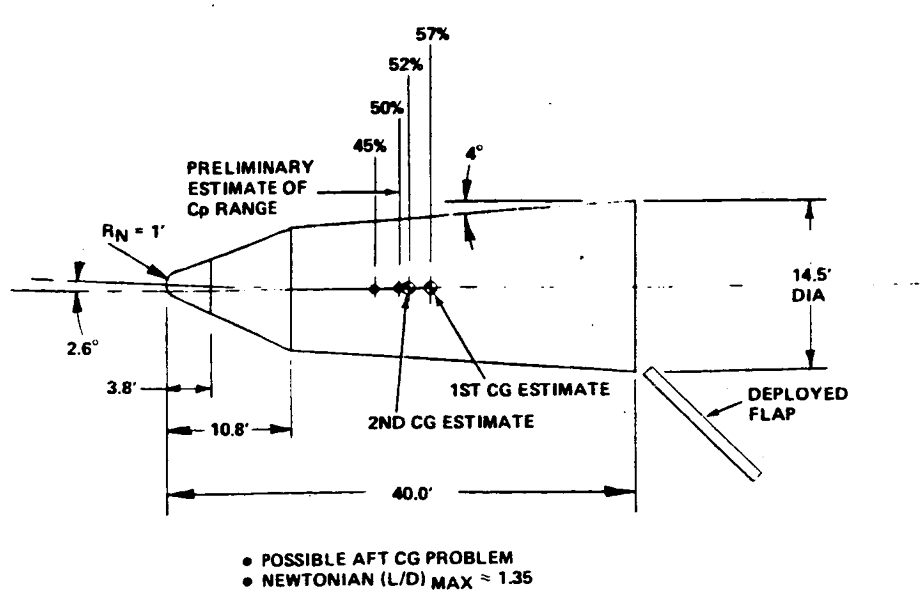


Figure 2.4-23



A weight breakdown is provided for H-1M in Figure 2.4-24.

Within the crew capsule subsystem, Bare Bones and on orbit mission equipment are defined in Figures TBD and TBD. "Deployment" provides structure and mechanism to permit the crew capsule to rotate away from the vehicle body. "Antirotation" provides a degree of freedom of the capsule relative to the vehicle, and allows the crew to be isolated from high roll rates during major engine burns with two engines malfunctioning. The 200 lb allocated for this category approximates the weight of two additional engines (222 lbs). Consequently, the weight and performance figures for H-1M are valid for either technique of obtaining the desired "fail safe, fal safe" redundancy in primary propulsion.

The weights for "Structure", "TPS Tank Insulation", "Propulsion Plumbing", "ACS", "EPS", and "Avionics" were derived from surveys of past OTV studies.

Engine weights were obtained from Aerojet Technical Services Corporation (Figure TBD). TPS Shell Insulation weights were derived from General Electric data on re-entry vehicle TPS. All vehicle weights contain 10% for contingencies.

Figure 2.4-24 **SMALL MANNED AOTV**

CONFIGURATION "H1M" MASS PROPERTIES

SUBSYSTEM	MASS	(LB)
CREW CAPSULE		5300
BARE BONES CAPSULE (2 MEN)	3580	
ON-ORBIT MISSION EQUIP	1320	
DEPLOYMENT	200	
ANTIROTATION	200	
STRUCTURE		3310
SHELL	1000	
SUPPORT	800	
FUEL TANK (EXTERNAL)	380	
OXIDIZER TANK	220	
FLAPS	580	
BERTHING/DOCKING	330	
THERMAL PROTECTION SYSTEM		1140
TANK INSULATION	240	
SHELL INSULATION	900	
PROPULSION		1114
ENGINE (4 3000# "SMALL MAT'L ADV")	444	
GIMBALS & ACTUATORS	0	
PLUMBING	670	
ACS		450
EPS		900
AVIONICS		700
		12196
	10% CONTINGENCY	1292
	TOTAL DRY WT =	14200 LB

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Two advanced technologies permit the H-1M vehicle to perform its mission on a single 65K STS flight: aeroassist (aerobraking and maneuvering) and small, high performance engines, Figure 2.4-25.

Figure 2.4-25 H-1M ENABLING TECHNOLOGIES

- MANNED GEO MISSION -

- H-1M MISSION ENABLED BY TWO TECHNOLOGIES: AEROASSIST & SMALL, HIGH PERFORMANCE ENGINES
  - AERODYNAMIC DECLERATION IS REQUIRED
  - AERO MANEUVERING PERMITS USEFUL PAYLOAD DELIVERY
  - HIGH I_{SP} IS REQUIRED

    $\approx$ 105 LB OF PAYLOAD FOR 1 SEC OF I_{SP}
  - SHORT, SMALL ENGINES PERMIT MEETING ORBITER PAYLOAD CG REQUIREMENTS
  - MULTIPLE LOW THRUST ENGINES IMPROVE MAN'S SURVIVABILITY AND PROBABILITY OF MISSION SUCCESS

The results of an analysis to determine the importance of aeromaneuvering and engine performance to the H-1M mission are illustrated in Figure 2.4-26. The left side of the figure outlines the baseline H-1M deliverables. Cross hatched entries stay with H-1M from LEO to GEO to LEO. The weights for these items include a 10% contingency factor. The upper entry on the left bar chart, 2100 lb represents the amount of propellants, tanks, avionics, etc., that H-1M brings to GEO and leaves there. These equipoments are to refurbish GEO satellites.

The bar chart on the right, and the data above it, defines a reduced capability H-1M mission without the 2100 lb of delivered payload. On a mission of this type, a crew could perform inspection and repair, but not the important task of replenishment (unless the supplies were delivered by another vehicle). The analysis indicated that loss of the capability to deliver 2100 lb would result from either.

- Elimination of aeromaneuvering by performing all plane changing with propulsion at GEO (although aerobraking is still required to LEO)

or

- Reduction of engine I_{sp} to 459 seconds.

It is emphasized that degrading both Isp and aeromaneuvering, to the levels outlined above, will preclude on H-1M mission at the previously reported vehicle weights. Consequently, a need is noted for engine specific impulse around 480 seconds and an $L/D \approx 1.0$ for the most attractive ground based manned missions to GEO on a single STS launch.

Figure 2.4-26 SOME H-1M ENHANCING TECHNOLOGY

H-1M GEO PAYLOAD: 7930 LB

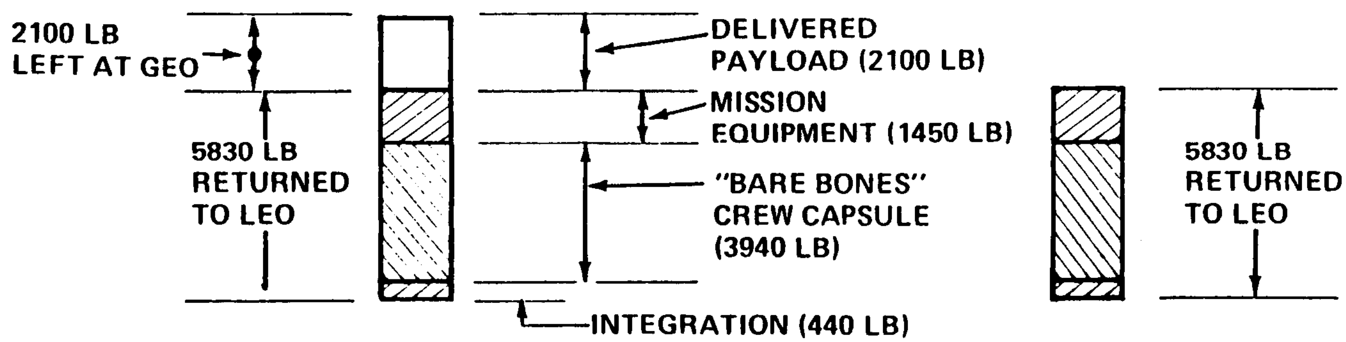
Δi_{AERO} = 14.6°
I_{SP} = 479 SEC

DEGRADED H-1M
GEO PAYLOAD: 5840 LB

Δi_{AERO} = 0 (AEROBRAKING)
& I_{SP} = 479 SEC

OR

Δi_{AERO} = 14.6° & I_{SP} = 459 SEC



1 TON OF DELIVERED PAYLOAD REQUIRES
- $L/D_{MAX} \geq 1.04$ &
- I_{SP} = 479 SEC

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2.4.6 Perigee Kick AOTV

An argument that asserts that the single most important characteristic of an upper stage is the cost of using it, that these costs are reduced by using a small AOTV for satellite delivery, that AOTV size and propellant needs are substantially reduced by staging, and, that the COMSAT community believes that its own satellites should be their own apogee stages, is outlined in Figure 2.4-27.

Expanding on the last point, a GEO satellite which performs station keeping contains all the subsystems which are required for performing apogee propulsion. An example of this capability was the self rescuing of the TDRSS after an IUS failure. The only satellite penalty associated with having an satellite perform its own apogee burn is the need for enlarged tankage on the COMSAT. This penalty is small when compared with the benefits of two stage delivery from LEO to GEO.

Figure 2.4-27

SOME USER LAUNCH COST CONSIDERATIONS

- THE MAJOR LAUNCH COST IS RENTING SHUTTLE PAYLOAD BAY
  - PAYLOAD + STAGE LENGTH > 45′
  - PAYLOAD + STAGE WEIGHT > 50,000 LB

  $80M + IN 1990
- SUBSTANTIAL EFFICIENCY IS AVAILABLE FROM ROCKET STAGING
  - VEHICLE SIZE & PROPELLANT NEEDS GREATLY REDUCED
- COST OF PROPELLANT & THROWAWAY STRUCTURE
  - AN ORDER OF MAGNITUDE LESS THAN STS LAUNCH
- COMMERCIAL USER WANTS GEO DELIVERY AT MIN COST
  - REDUCE UPPER STAGE LENGTH TO MINIMUM
  - REDUCE UPPER STAGE WEIGHT TO MINIMUM
  - STAGE AT GEO ALTITUDE & $I \approx 26°$
  - PAYLOAD CONTAINS ITS OWN APOGEE KICK STAGE
    - SMALL COST TO SATELLITES WHICH ARE AUTONOMOUS VEHICLES

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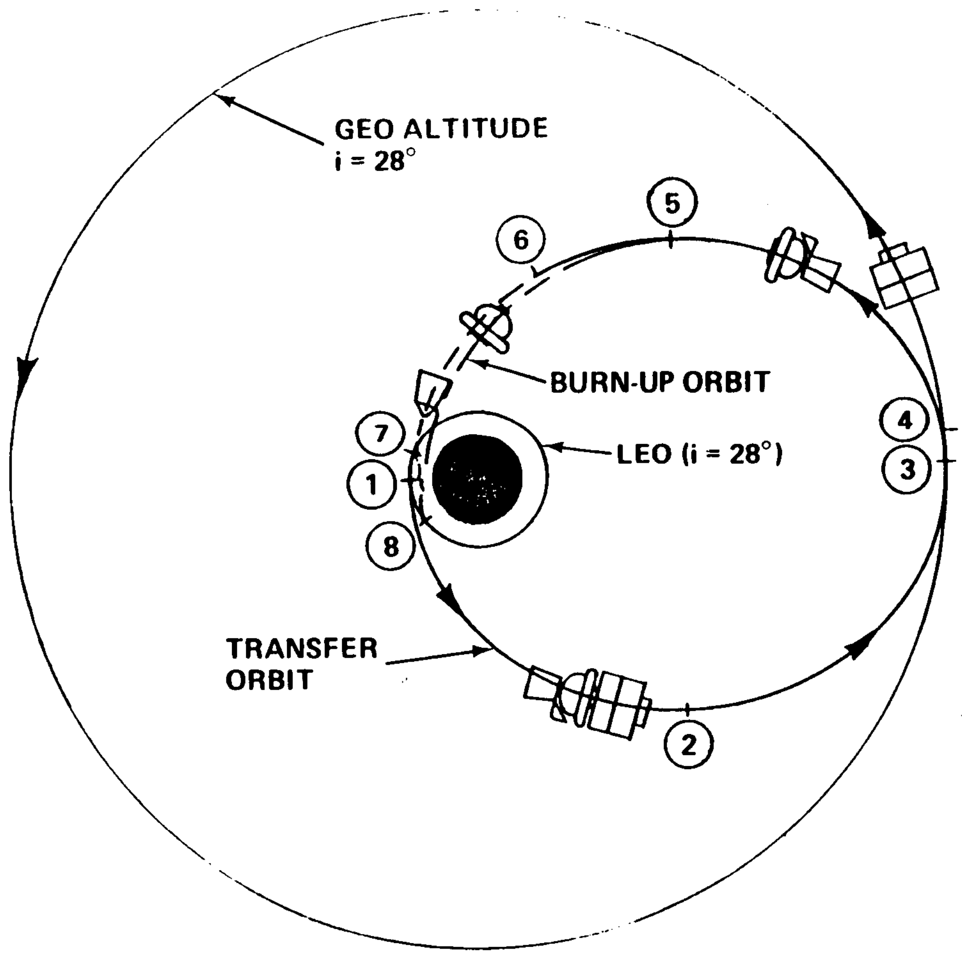
A scenario for two stage delivery using an ADOV (Aeroassisted delivery only vehicle) is outlined in Figure 2.4-28. A single perigee burn Hohmann transfer takes place with an ADOV, a set of drop tanks, and a payload (or payloads) structurally joined together. Before apogee, the payload separates. At apogee, the payloads delivers its own "apogee kick" and inserts itself into Geostationary orbit. The ADOV separates from its drop tanks. Small impulses change the drop tank trajectory to one which will cause the tanks to burn up when they hit the atmosphere. The ADOV trajectory has also been changed to provide the proper flight path entry angle for atmopsheric capture.

Figure 2.4-28 **OPERATING SCENARIO FOR ADOV WITH STAGING**



EVENTS

1. GEO TRANSFER BURN
   $\Delta V_1 = 8084$ FT/SEC
2. MIDCOURSE CORRECTION
   $\Delta V_2 = 50$ FT/SEC
3. STAGE PAYLOAD
4. PAYLOAD PERFORMS GEO CIRCULARIZING & PLANE CHANGE BURN
   $\Delta V_4 = 5850$ FT/SEC
5. STAGE DROP TANK. DROP TANK BURN
   $\Delta V_5 \approx 50$ FT/SEC
6. AOTV MIDCOURSE CORRECTION
   $\Delta V_6 = 50$ FT/SEC
7. AOTV ENTERS ATMOSPHERE
8. AOTV LEO CIRCULARIZING & PHASING BURN
   $\Delta V_8 \approx 500$ FT/SEC

The performance characteristics of a small (10'), light (4000 lb) AOTV, OH-1, which is designed for two stage satellite delivery: perigee kick from teh cryo AOTV and apogee kick from the satellite, is summarized in FIgure 2.4-29. A system utilizing a single set of drop tanks can deliver 11,000 lb to GEO Transfer Orbit while only utilizing 1/2 of the orbiter Cargo Bay. Thus, substantial cost sharing is available with other Cargo Bay users.

OH-1 is the first vehicle designed in this study with external oxygen (O) and hydrogen (H) tanks.

Figure 2.4-29

MIN LENGTH AOTV "OH-1": PERFORMANCE ON DELIVERY ONLY TO GEO ALT (h = 19300 N Mi) & i = 28°

LEO CAPABILITY	65,000 LB	I_{sp} = 480 SEC
ASE	2,900 LB	ΔV_1 = 8084 FT/SEC
PROPELLANT:	11,750 LB	ΔV_2 = 500 FT/SEC
DROP TANK: USEABLE	(11,200)	
DROP TANK: RESERVES & RESIDUAL	(230)	$(T/W)_1$ = .218
DROP TANK: INFLIGHT LOSSES	(170)	
AOTV INTERNAL	(150)	$(T/W)_2$ = 1.45
AOTV DRY WEIGHT	4,000 LB	
DROP TANK "½ STAGE"	750 LB	LAUNCH WT = 30,400 LB IN ORBITER
PAYLOAD DELIVERED	11,000 LB	LAUNCH LENGTH = 16 FT OR 22 FT IN ORBITER
PAYLOAD RETURNED	0	
AOTV ATMOS ENTRY MASS	4,150 LB	

- ½ OF ORBITER PAYLOAD BAY REMAINS EMPTY
  - SUBSTANTIAL LAUNCH COST SHARING WITH OTHER CARGO

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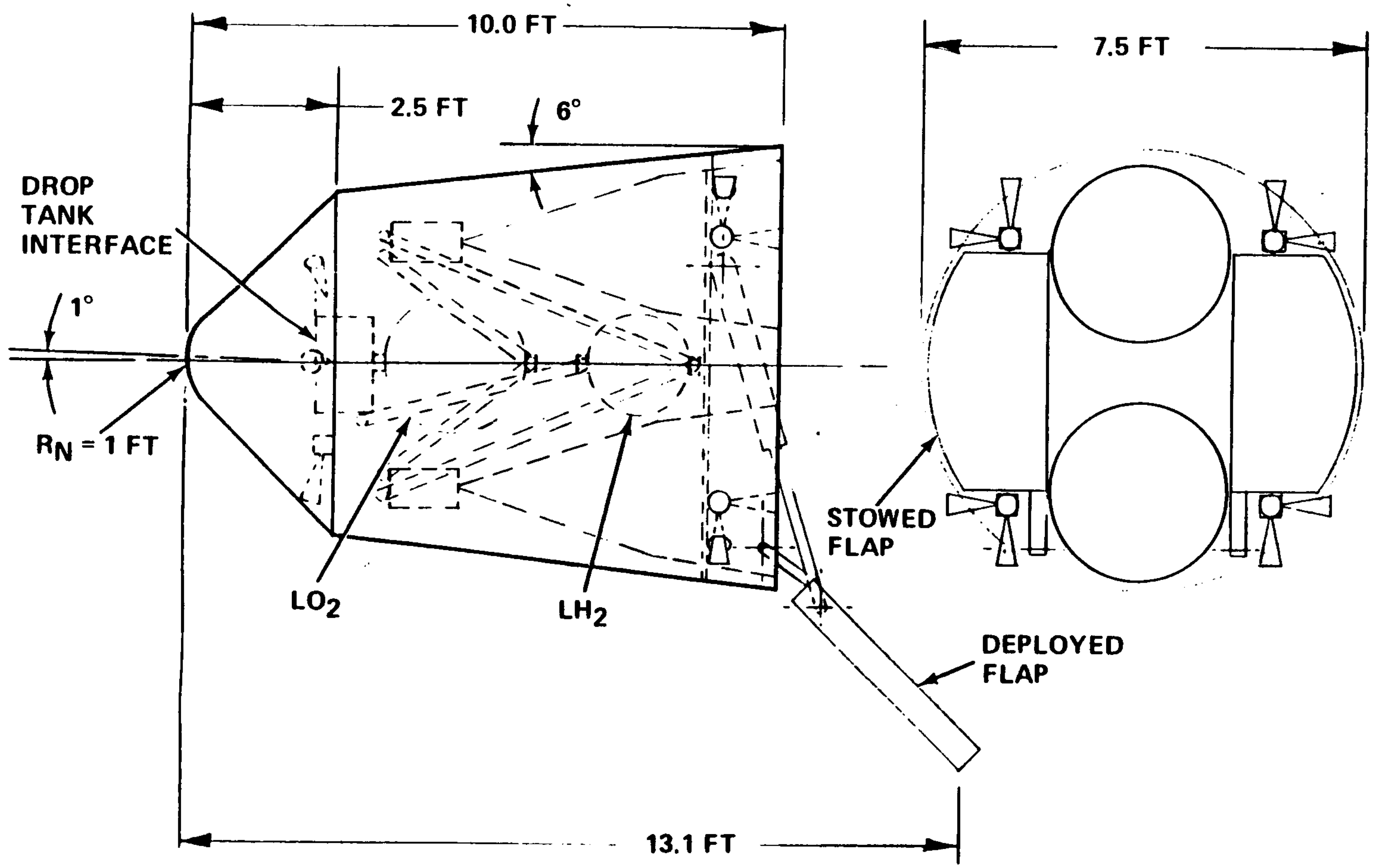
OH-1 length (10′) is determined by engine length and nose size, Figure 2.4-30. The engines choosen for OH-1 are Aerojet advanced expander cycle (LOX/LH burning) with an area ratio of 1000:1. The two engine vehicle is not intended to operate with one engine. Engine throttling is used to align the thrust vector with launch CM displacements in the plane of the two engines. CM offsets perpendicular to this plane will produce a yawing moment during engine firing. An opposing yawing moment is maintained by firing the appropriate OH-1 ACS thrusters. Propellant for this length ACS burn is stored on the drop tanks, and its weight is chargeable to payload weight.

Figure 2.4-30 **MIN LENGTH AOTV "OH-1": ATMOSPHERIC ENTRY**

- HYPERSONIC $(L/D)_{MAX}$ = TBD. PROBABLY, $(L/D)_{MAX} < .75$
- 2 ADVANCED EXPANDER ENGINES (FIXED, WITH FIXED NOZZLES)



- LARGE QUANTITIES OF ACS PROPELLANT ARE STORED ON THROWAWAY TANK. WEIGHT IS CHARGED TO AOTV PAYLOAD.

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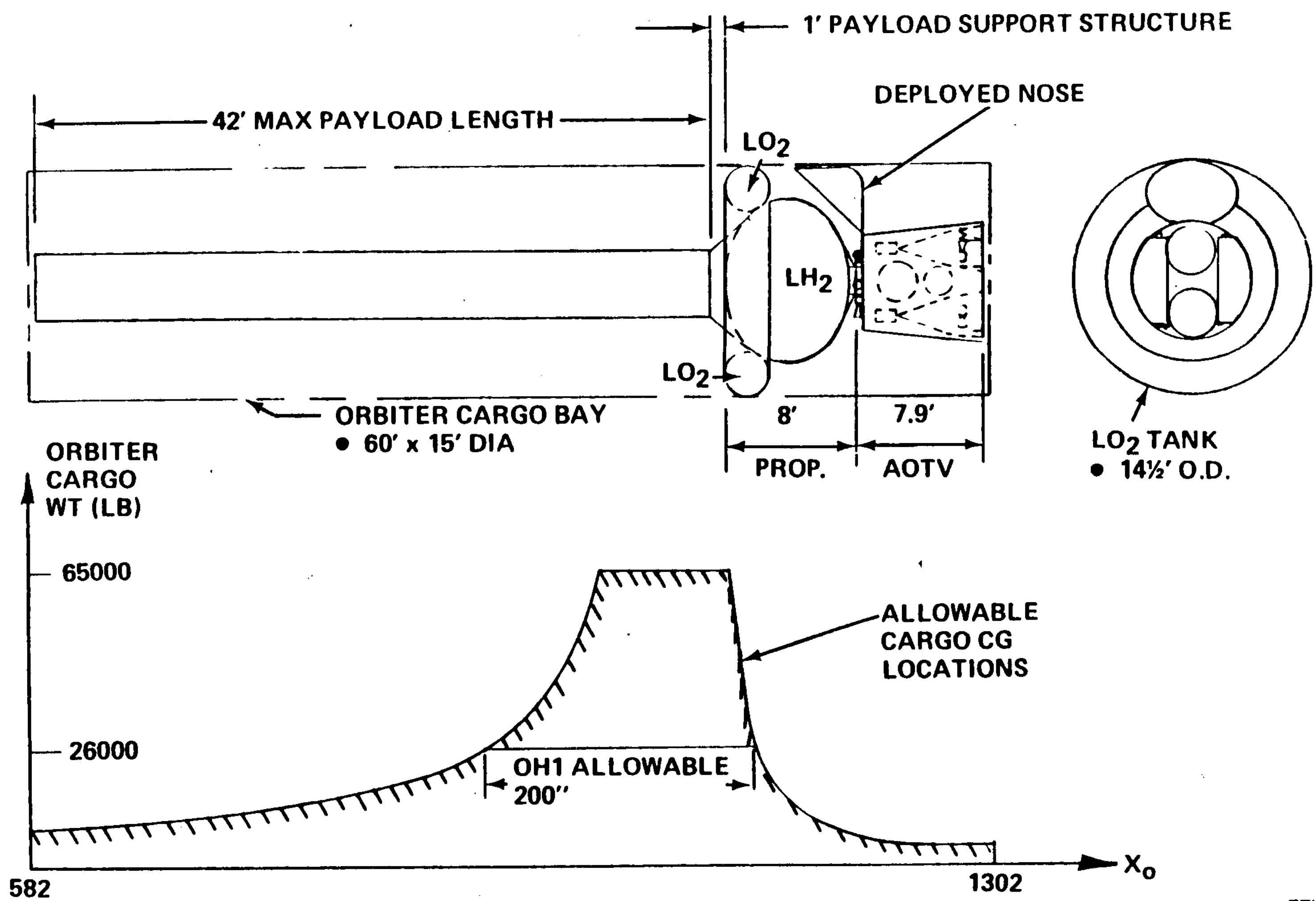
A major advantage of a short upper stage is displayed in Figure 2.4-31: the capability to deliver long payloads using a single STS flight. With its nose open to access the fuel transfer bulkhead, OH-1 is 8 feet long. An 8 foot drop tank plus a one foot payload support structure completes the 17 foot long upper stage. Allowing 1/2′ clearance at each end of the orbiter, 42′ long payloads are deliverable with this system.

The drop tanks are composed of an ellipsoidal hydrogen tank and a toroidal oxygen tank. Since, for a perigee kick mission which depletes all external tank propellants in a single burn, the toroidal tank is either full or empty, it probably does not require the extensive internal baffling that is typica of most toroidal tank designs.

The plot of center-of-mass vs. cargo weight allowables in the lower sketch shows that the 26,750 lb AOTV center-of-mass can have a 200 inch range in the orbiter.

Figure 2.4-31 **MIN LENGTH AOTV "OH1": LAUNCH CONFIGURATION TOROIDAL LOX TANK**



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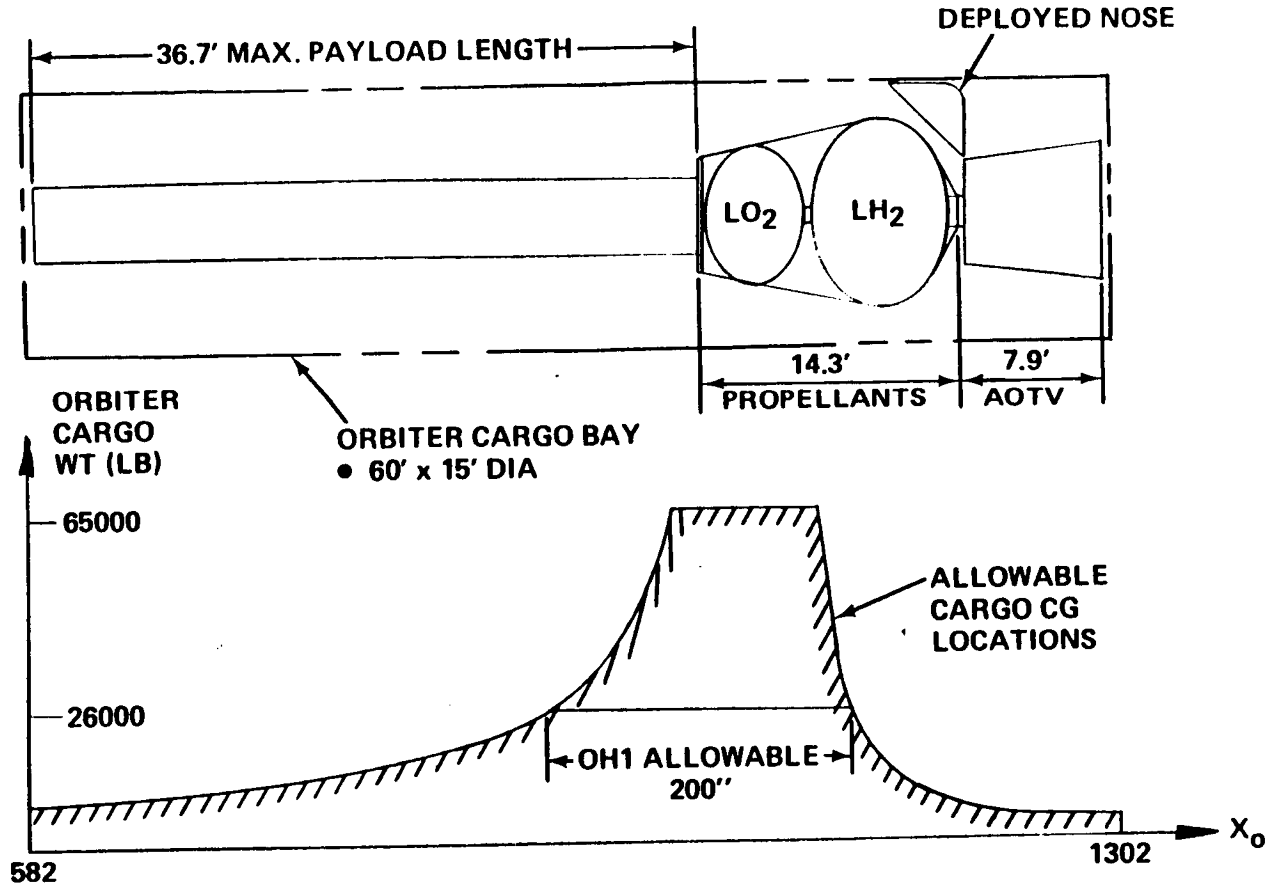
Ellipsoidal drop tanks offer the opportunity for multi-burn missions without the weight penalty of conventional toroidal tanks. However, they occupy a greater length of cargo bay and decrease the allowable payload length, Figure 2.4-32.

Figure 2.4-32 **MIN LENGTH AOTV "OH1": LAUNCH CONFIGURATION ELLIPSOIDAL LOX TANK**



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Subsystem weights for OH-1 were estimated by the technique outlined in Section 2.4-1, the same technique that was applied to H-1M and are summarized in Figure 2.4-33. Again, a 10% contingency factor is included in the vehicle total of 4000 lb.

Figure 2.4-33 MINIMUM LENGTH AOTV AEROASSISTED DELIVERY ONLY VEHICLE (ADOV) CONFIGURATION "OH-1" MASS PROPERTIES

SUBSYSTEM		MASS (LB)
STRUCTURE		1270
SHELL	140	
SUPPORT	480	
FUEL TANK	30	
OXIDIZER TANK	20	
FLAPS	350	
DOCKING/LATCHING	250	
TPS		135
TANK INSULATION	10	
SHELL INSULATION	125	
PROPULSION		732
ENGINE (2 3000 LB "SMALL MAT'L ADV")	222	
GIMBALS & ACTUATORS	0	
PLUMBING	510	
ACS		380
EPS		440
AVIONICS		610
DRY WEIGHT =		3567
10% CONTINGENCY =		357
TOTAL DRY WEIGHT =		3920 ≈ 4000 LB

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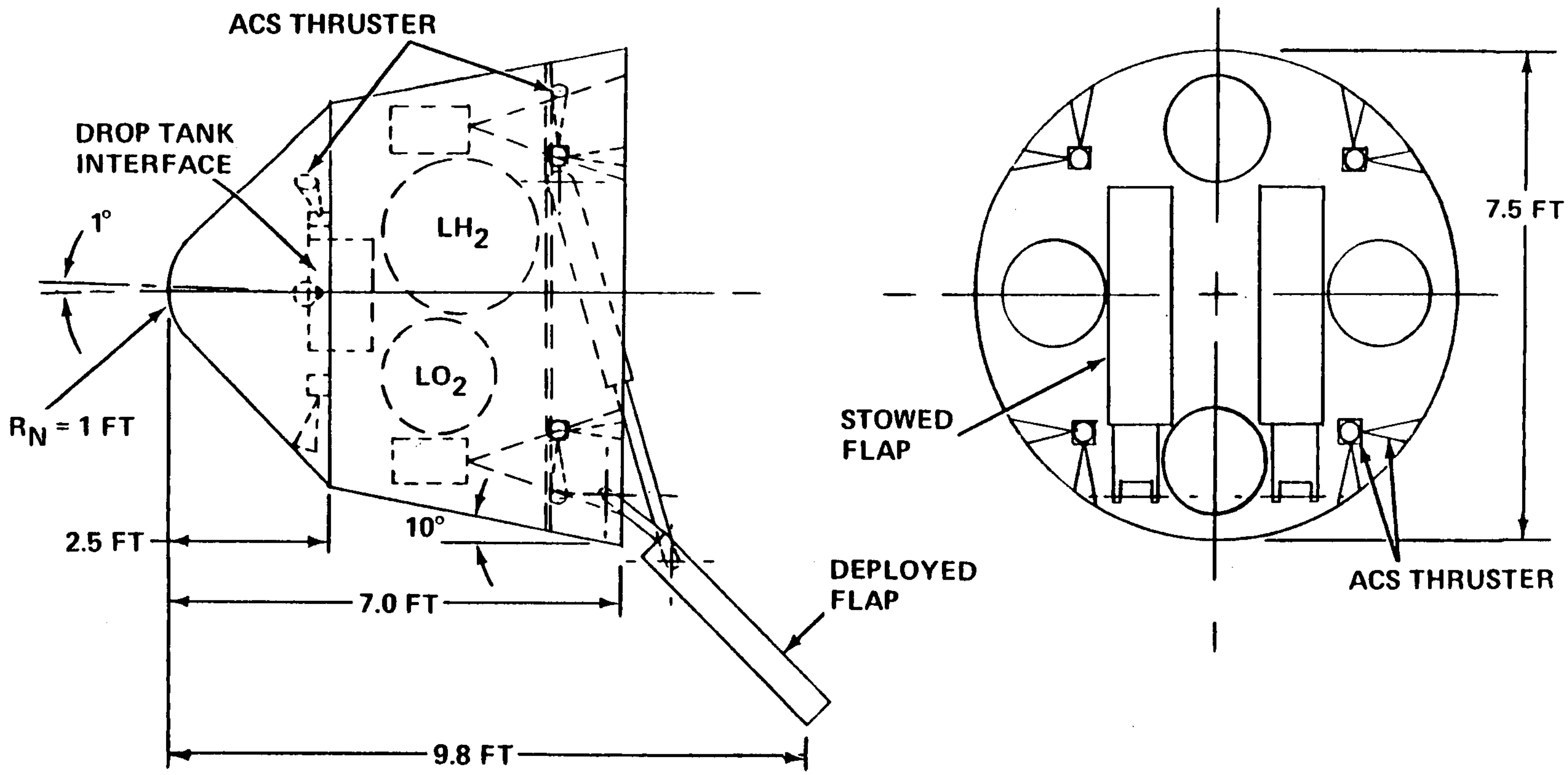
Another small perigee kick stage (for payload delivery missions) is defined in Figure 2.4-34. OH-3 utilizes 4 of the shortest high performance engines in the current stable of advanced LOX/Hydrogen engine designs: the Aerojet core engine, with a fully regeneratively cooled nozzle.

Alignment of the resultant thrust vector (from the 3 engines) and vehicle CG (with and without payloads) is obtained by throttling back on individual engines until pitching or yawing moments are eliminated. If significant attitude control system propellant is carried on drop tanks (or payloads), a mission can be successfully completed even if an engine fails. For this case, the diametrically opposed engine is shut down, the the ACS thrusters create a moment to counteract the moment created by the small offset between stage CG and the plkane of the two thrusting engines.

Figure 2.4-34 **MINIMUM LENGTH AOTV "OH-3": ATMOSPHERIC ENTRY**

- HYPERSONIC (L/D) MAX = 0.65 , W/C_DA = 100 LB/FT^2
- FOUR ADVANCED EXPANDER ENGINES (FIXED NOZZLE, FIXED ENGINE)
  - ϵ = 400:1, I_{SP} = 475 SEC AT O/F = 6:1, THRUST = 3000 LB
  - LARGE CG OFFSET TOLERANCE
  - POSSIBLE ENGINE OUT OPERATION



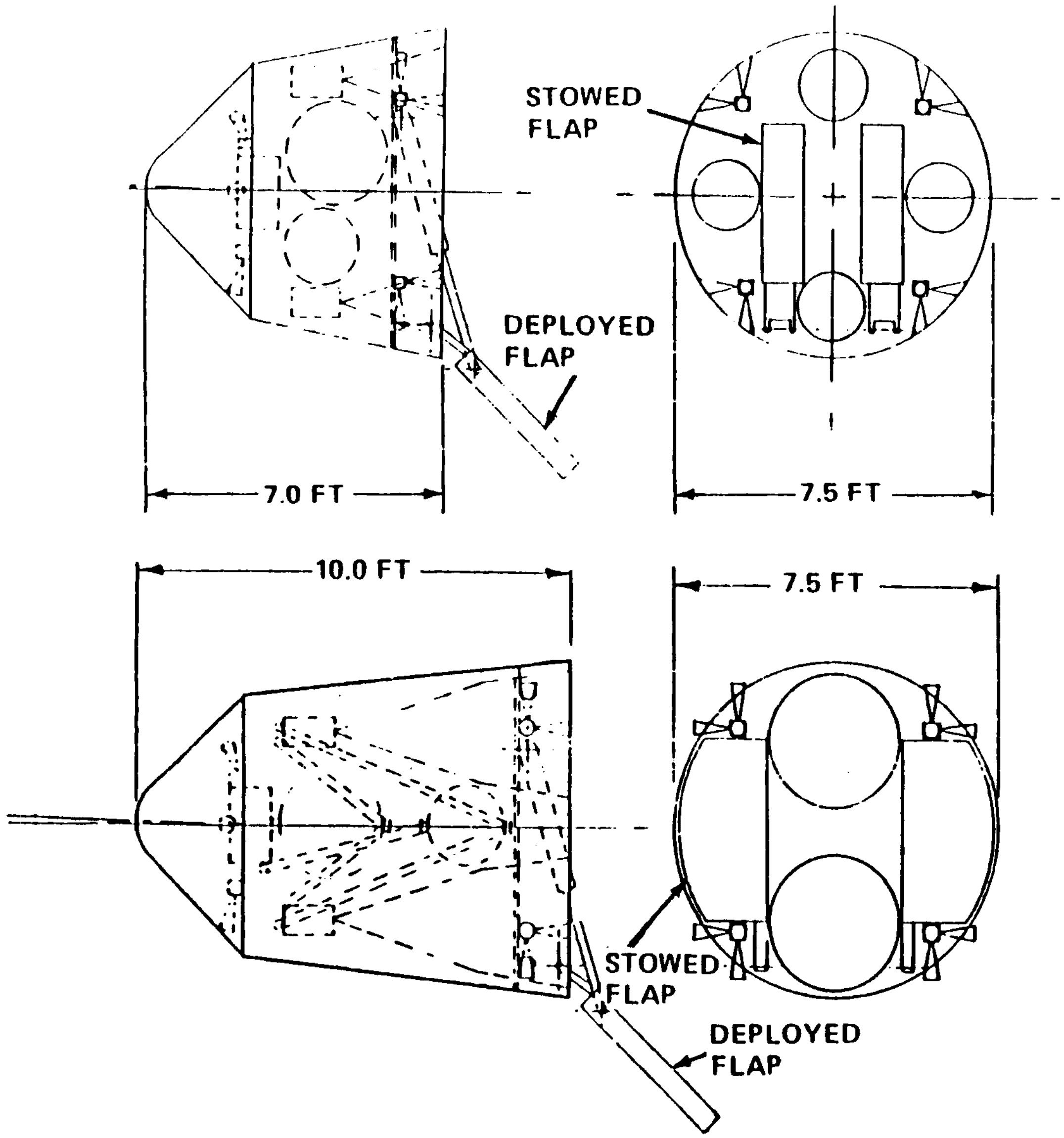
GRUMMAN

V83-1652-026

A comparison of the performance of OH-3 and OH-1 in delivering the payload of Figure 2.4-38 is outlined in Figure 2.4-35. Both AOTV systems deliver the same weight to a GEO transfer orbit at 28° inclination: 11,400 lb. The payload gain expected of the high expansion ratio, high Isp engines of OH-1 are offset by gravity loses produced during a single perigee kick burn with a stage thrust-to-weight ratio at ignition of 0.21.

Both OH-1 and OH-3 have the same dry weight since the extra weight added to OH-3 from two additional engines and extra plumbing was lffset by reduced structure and TPS on the 3' shorter vehicle.

Figure 2.4-35 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

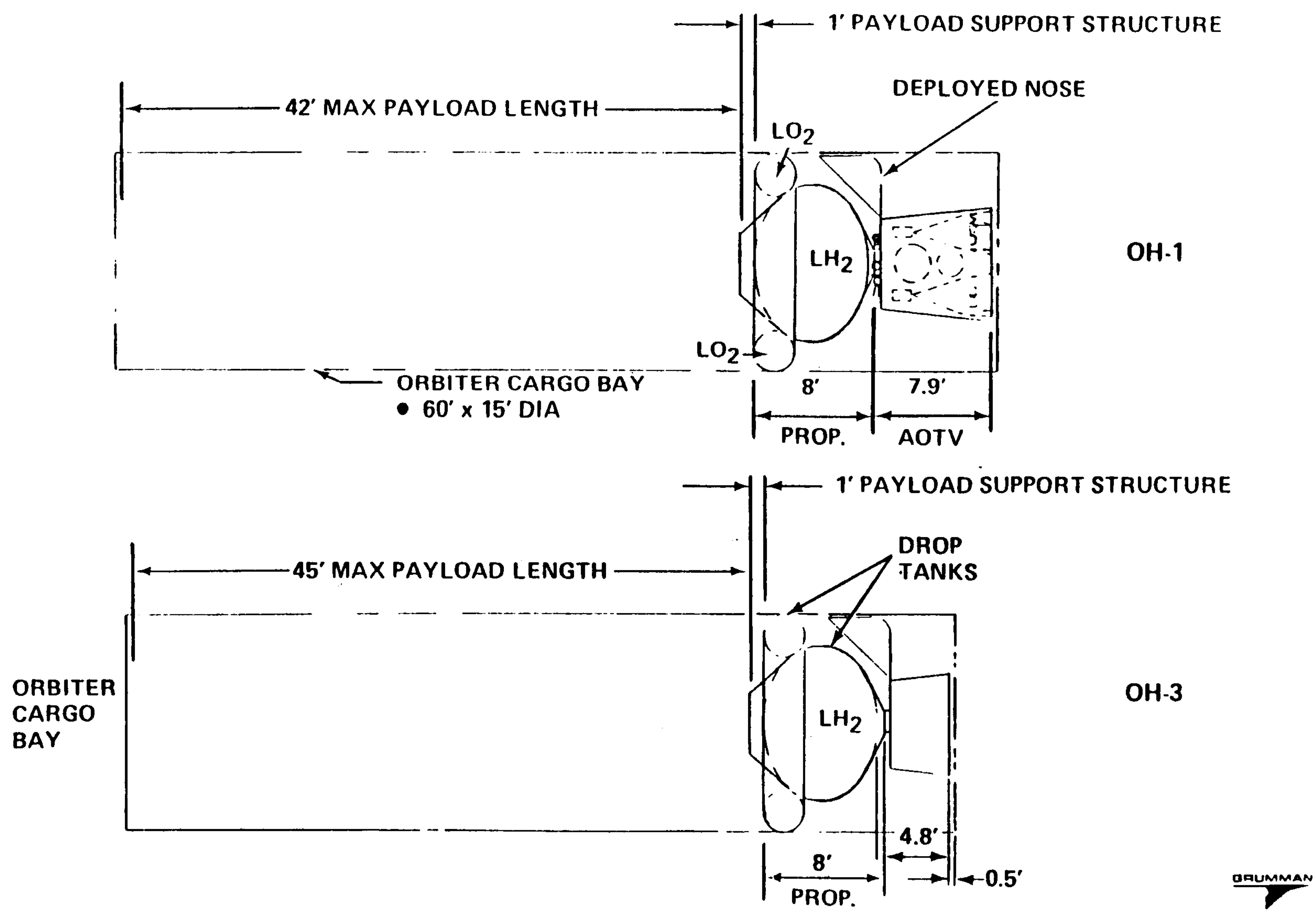
- TWO ADVANCED EXPANDER ENGINES
  ϵ = 1000: 1
  I_{SP} = 480 SEC
- TOTAL THRUST = 6000 LB
- 4000 LB DRY WEIGHT
- PAYLOAD DELIVERY TO GEO TRANSFER ORBIT WITH 11,930 LB OF PROPELLANT: W_{PAY} = 11,400 LB

GRUMMAN

V83-1652-038W

OH-1 and OH-3 with the same pair of drop tanks, waiting for installation of a payload are illustrated in Figure 2.4-36. OH-3 can accept a 45′ long payload, while OH-1 can accept a 42′ long payload. Since STS launched payloads (satellites and upper stages) are frequently charged by the length of payload bay they occupy ($\sim$\$2M/foot), the shorter OH-3 configuration offers significant advantages.

Figure 2.4-36 **COMPARISON OF MAXIMUM PAYLOAD LENGTHS: OH-1 & OH-3**



V83-1652-023W

A comparison of the masses of OH-1 and OH-3 indicates that they weigh the same: 4000 lb, Figure 2.4-37. Except for the engine system, all subsystem weights assume current state of the art technology.

Figure 2.4-37 COMPARISON OF MASS PROPERTIES: OH-1 & OH-3

SUBSYSTEM	MASS (LB) OH-1		OH-3
• STRUCTURE			
– SHELL	140		90
– SUPPORT	480		360
– FUEL TANK	30		30
– OXIDIZER TANK	20		20
– FLAPS	350		350
– DOCKING/LATCHING	250		250
• TPS			
– TANK INSULATION	10		10
– SHELL INSULATION	125		80
• PROPULSION			
– ENGINE (3000 LB "SMALL MATL ADV")	222		404
– GIMBALS & ACTUATORS	0		0
– PLUMBING	510		550
• ACS	380		380
• EPS	440		440
• AVIONICS	610		610
DRY WEIGHT =	3567		3574
10% CONTINGENCY =	357		356
TOTAL DRY WEIGHT =	3920 ≈	4000 LB ≈	3930 LB

GRUMMAN

V83-1652-008W

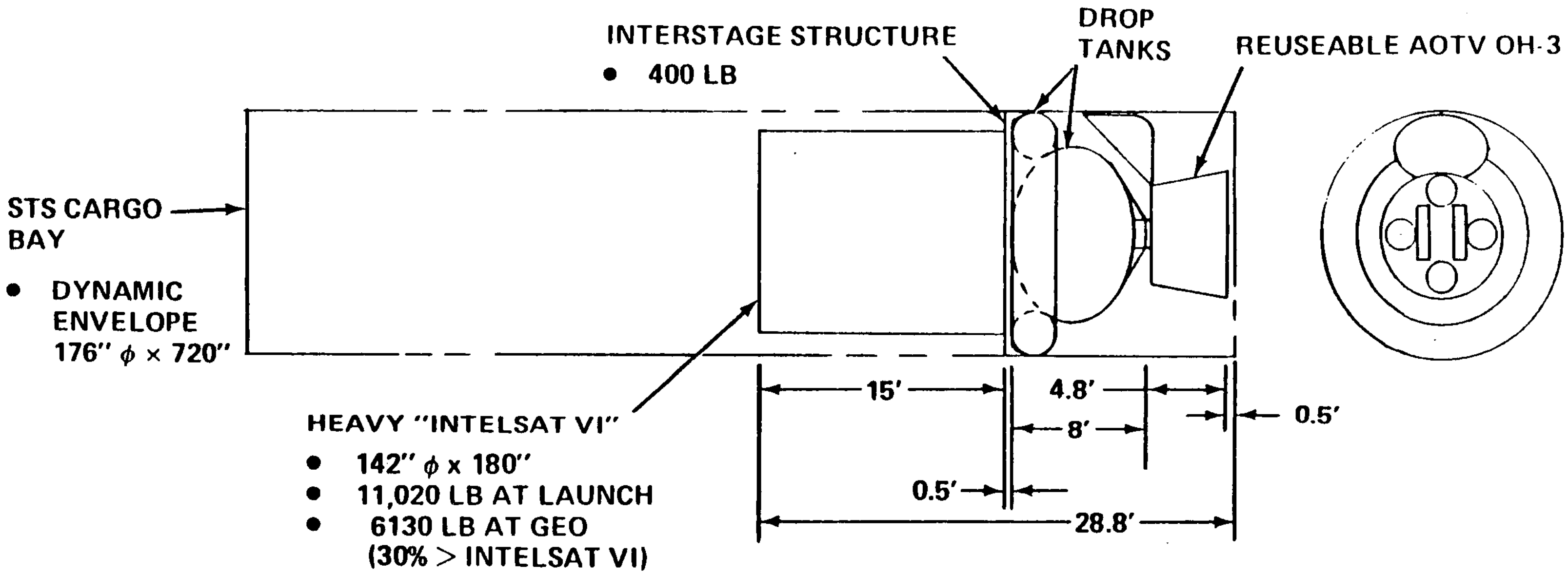
All of the perigee kick AOTV's can be configured with different size drop tanks. Figure 2.4-38 illustrates OH-3 with a single set of drop tanks. These tanks have been designed to produce a minimum length upper stage. The ellipsoidal hydrogen tank fits beneath the deployed nose of OH-3 and nests within the toroidal oxygen tank.

The "Intelsat VI" sized satellite is shown attached to the drop tanks via a 400 lb interstage structure. This interstage structure connects the upper stage to the Orbiter. The weight of the satellite (11,020 lb) is determined by the propellant capacity of the drop tanks and Oh-3 internal tanks (a total of 11,930 lb). The density of this satellite at launch (.0039 lb/in^3) is within 10% of the optimum density for Shuttle payloads. The resulting satellite weight contains all the propellant it needs to perform the circularization and plane changing burn at apogee for GEO insertion. Since the satellite contains all subsystems necessary for this burn (with its station keeping propulsion), the only residual weight penalty associated with this additional apogee burn is the larger size tank that must be added to hold the additional propellants. AT GEO (after the apogee burn), the "Heavy Intelsat VI" satellite weighs 6130 lb, which is 30% heavier than Intelsat VI (a low density satellite). Subtracting the weight of extra tankage (added for apogee burn) produces the weight the satellite would have if it were delivered to GEO by a single stage vehicle. This weight, "Useful Payload at GEO" (see Figure TBD), is 6030 lb for the satellite on the facing page.

The weight statement on the left of the figure indicates a total weight of 30,800 lb within the Orbiter Cargo Bay, less than 1/2 of the 65,000 lb cargo capacity. The length of upper stage plus payload is 29', also less than half the orbiter's capacity. Consequently, a short, light system like this shares STS launch costs with other payloads.

Figure 2.4-38

ADOV OH-3 & HEAVY "INTELSAT VI"



OH-3 PERFORMANCE TO GEO TRANSFER ORBIT (i = 28°):

INTERSTAGE STRUCT	400 LB
PROPELLANT	11,930 LB
DROP TANK	750 LB
AOTV OH-3	4000 LB
PAYLOAD	11,020 LB
Σ_1 = GLOW =	28,100 LB
AIRBORNE SUPPORT EQUIPMENT	2700 LB
Σ_2 = STS CARGO WT =	30,800 LB

I_{SP} = 475 SEC; T = 12,000 LB

$(T/W)_1 = 0.43 \rightarrow$ GRAVITY LOSS = 25 FT/SEC

ΔV_1 = 8004 FT/SEC

$(T/W)_2 = \begin{Bmatrix} 3.0 \text{ FOR 4 ENG} \\ 1.5 \text{ FOR 2 ENG} \end{Bmatrix} \rightarrow$ GRAVITY LOSS = 0

ΔV_2 = 500 FT/SEC

GRUMMAN

V83-1652-025W

A technique for supporting multiple payloads (in tandem) on one Orbit Transfer Vehicle is outlined in Figure 2.4-39. The device, a Multiple Payload Support Structure (MPSS), is compatible with a 176" diameter constraint in the Orbiter Cargo Bay.

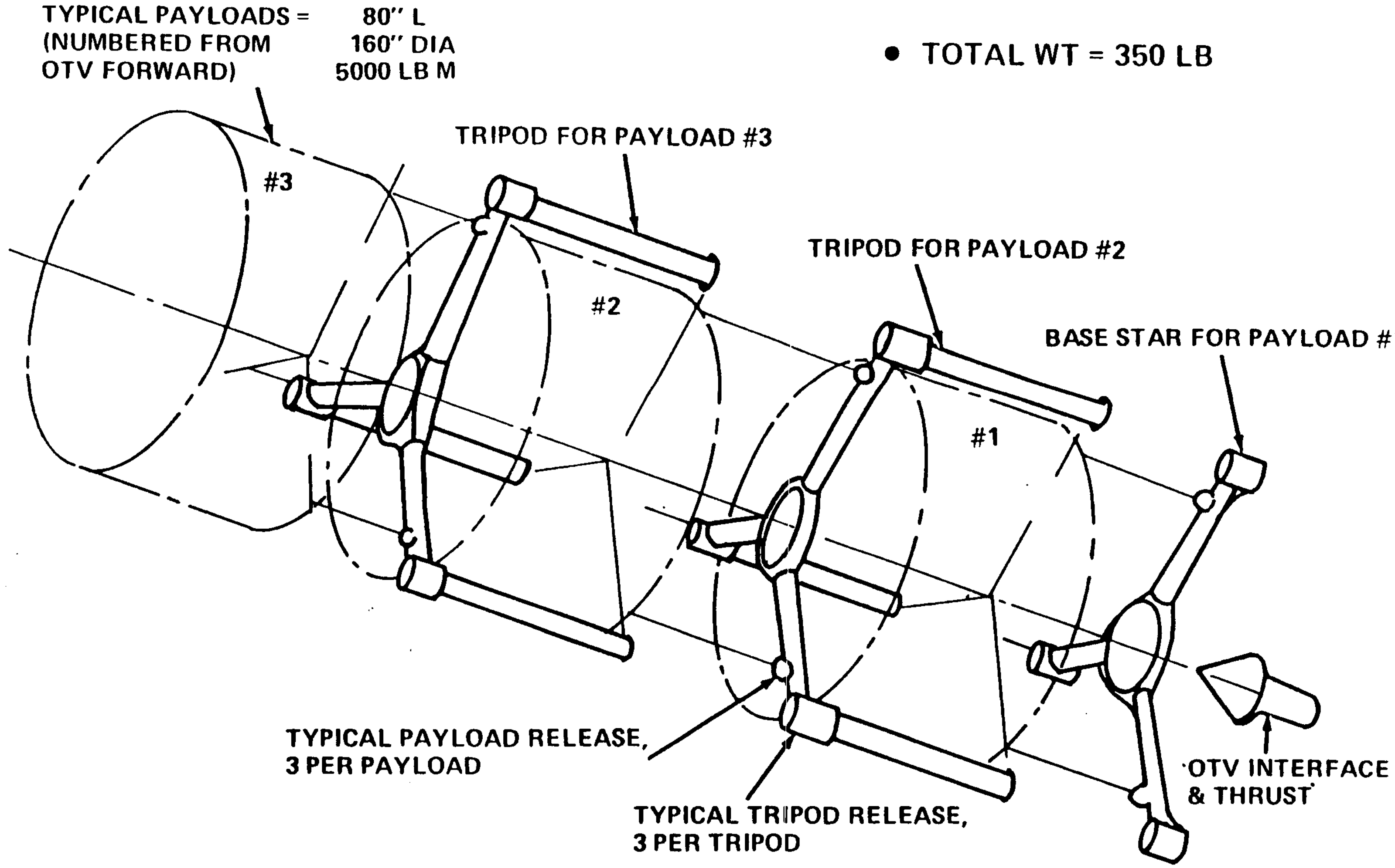
Payloads are supported at 3 points in a diametrical plane. A system of latches and springs deploys payloads sequentially: first Payload #3, then the tripod which supported it, then Payload #2 then the tripod which supported Payload #2, then Payload #1.

This preliminary design of MPSS was developed to form a ROM estimate of the weight and cost of such a system. The 350 lb estimate for the system shown on the facing page is varied with total payload weight.

Figure 2.4-39

MULTIPLE PAYLOAD SUPPORT STRUCTURE (MPSS)

• TOTAL WT = 350 LB



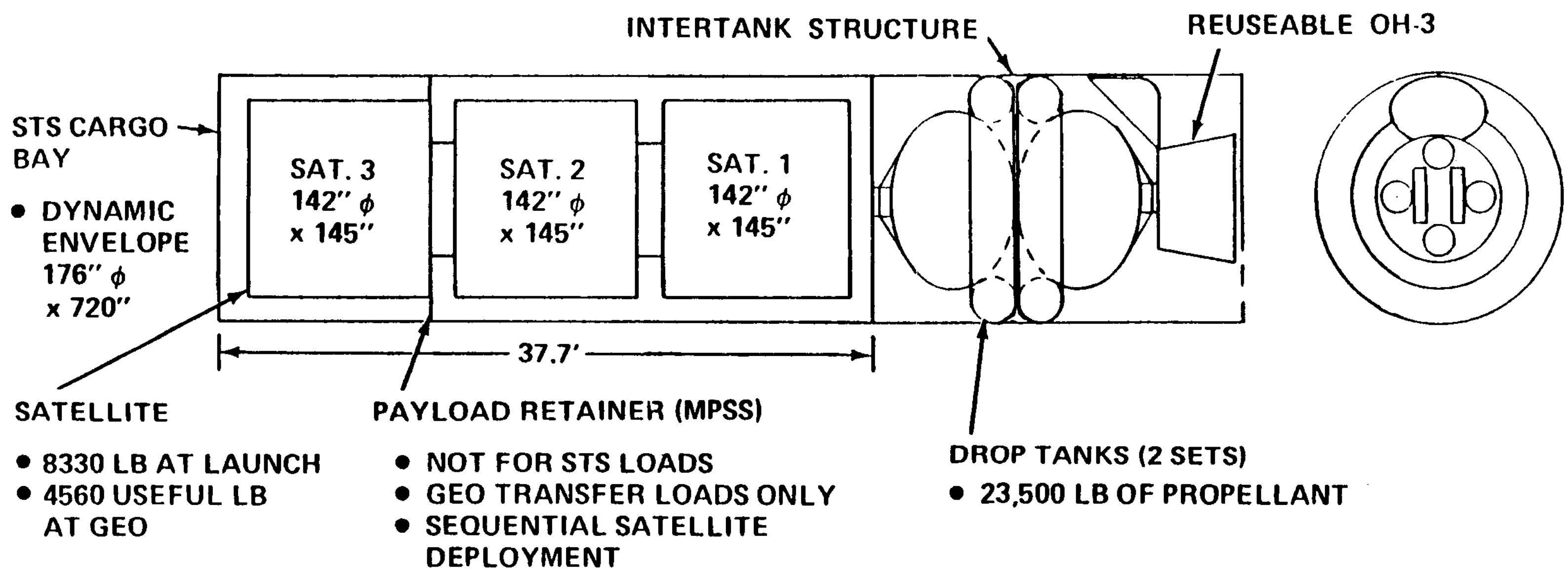
GRUMMAN

V83-1652-046W

A method of using OH-3 which attempts to utilize the full capacity of a 65,000 lb shuttle launch system is outlined in Figure 2.4-40. Two identical sets of "minimum length" drop tanks contain 23,500 lb of LOX/Hydrogen. This permits delivery, to the apogee of a GEO Tranfer Orbit, of 25,000 lb of satellites as well as the tanks, MPSS (payload retainer), intertank structure and AOTV OH-3. After self insertion at GEO, the 3 satellites represents 13,680 lb of useful weights at GEO.

The total weight of all cargos, in this example, is 58,000 lb. This system underutilized the STS launch capacity by 7,000 lb. It has been estimated that a system which made use of the additional 7,000 lb which are available (by providing different drop tanks with more propellant) will permit placement of 16,000 lb of useful weight at GEO (an increase of 2,300 lb over the example in Figure 2.4-40).

Figure 2.4-40 ADOV OH-3 & MULTIPLE PAYLOADS



OH-3 PERFORMANCE TO GEO TRANSFER ORBIT (i = 28°):

AOTV OH-3	4000 LB
INTERTANK STRUCTURE	500 LB
PROPELLANT	23,500 LB
DROP TANKS (2 SETS)	1500 LB
PAYLOAD RETAINER	400 LB
PAYLOADS (3)	24,990 LB
Σ_1 = GLOW =	54,890 LB
ASE	3300 LB
Σ_2 = STS CARGO WT =	58,190 LB

$I_{SP} = 475$; T = 12,000 LB

$(T/W)_1 = 0.22$; GRAVITY LOSS = 97 FT/SEC

$\Delta V_1 = 8076$ FT/SEC

$(T/W)_2 = \left\{ \begin{array}{l} 3.0 \text{ FOR 4 ENG} \\ 1.5 \text{ FOR 2 ENG} \end{array} \right\} \Longrightarrow$ GRAVITY LOSS = 0

$\Delta V_2 = 500$ FT/SEC

GRUMMAN

V83-1652-024

2.5 Payload Delivery Capability

For purposes of this study, the cargo bay dimensions and the lift capability to low earth orbit of the various launch vehicles was specified by NASA, Table 2.3-1. The trend of launch vehicle lift capability with orbit inclination was taken from (24 Figure 3-5) and is presented in Figure 2.5-1.

NASA-S-77-3159

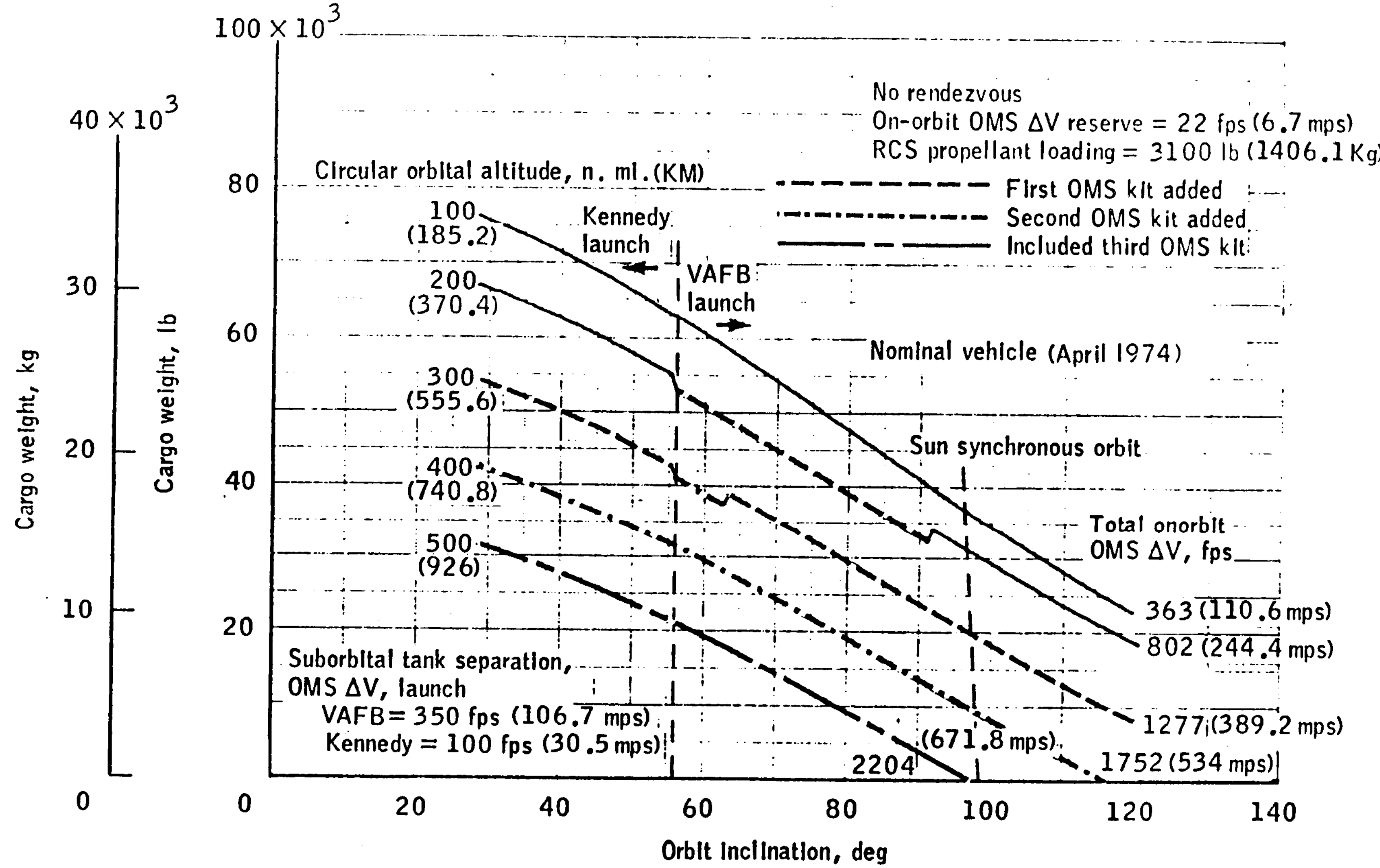


Figure 2.5-1 Cargo weight versus inclination for various circular orbital altitude - delivery only.

During the initial activity, AOTV payload delivery capability was computed employing the ΔV budgets established in Section 2.1 and the AOTV dry weights generated employing the initial mass scaling techniques. These results are summarized in Table 2.5-1 for an I_{sp} = 460 sec and state-of-the-art technology. Note that for an $L/D \geq 1.04$, 9300 lbs can be delivered to GEO and 7900 lbs can be delivered to GEO with 2000 lbs returned using a 65K lbs STS lift capability. Employing a 100K lbs lift capability of an advanced STS, it was estimated that 10,300 lbs could be round-tripped to GEO for a 12K lb AOTV. An advanced STS or SDV lift capability of 115K lbs would be required to round trip 14K lbs to GEO. The dry weight of the GEO servicing vehicle is greater than the GEO delivery vehicle in this example due to the larger mass of the return vehicle, thus requiring a larger body flap, a larger attitude control subsystem, and a larger main propulsion subsystem.

Table 2.5-1 INITIAL BASELINE AMOTV PAYLOAD DELIVERY CAPABILITY

I_{SP} = 460 SEC

	GEO DELIVERY	GEO SERVICING		GEO MANNED ROUND TRIP	
LEO CAPABILITY	65000	65000	64000	100000	115040
ASE	2000	2000	2000	3108	3591
AMOTV DRY WEIGHT	8488	8685	8488	12042	12042
PROPELLANT	43950	45107	44359	72483	83128
RESERVES & RESIDUALS	894	923	894	1456	1668
INFLIGHT LOSSES	373	388	373	611	611
PAYLOAD DELIVERED	9300	7897	7886	10300	14000
AMOTV ENTRY MASS	9512	11571	9542	24129	28181
PAYLOAD RETURNED	0	2000	2000	10300	14000

During the initial activity, 6 hour polar and Molniya payload delivery capability of a mid L/D AOTV was compared to an aerobraked and an all propulsive OTV, Figures 2.5-2 and 3. State of the art technology with Isp = 460 sec was assumed. For this comparison the dry weights of all the vehicles were assumed identical (an unrealistic assumption, but necessary at this time to illustrate order-of-magnitude trends). The clear advantages of the aeroassisted vehicles over the all propulsive are illustrated in Figure 2.5-2 for the six hour polar delivery mission. Results of the Molniya evaluations are summarized in Figure 2.5-3. In order for the in atmosphere maneuver to produce an orbital plane change, that portion of the trajectory must be flown at a nodal crossing. To accomplish this, a large ΔV must be applied to rotate the apses of the Molniya orbit. As can be noted, the payload capability with 35° apsis rotation and L/D is less than that of an aerobrake (L/D = 0) with small or no apsis rotation required or a mid L/D AOTV with small apsis rotation flying only in the braking mode. Thus it is recommended that for return from a highly elliptical orbit, such as the Molniya, the mid L/D AOTV be employed only in the braking mode, in order to maximize payload delivered.

Figure 2.5-3 MOLYNIA DELIVERY CAPABILITY FOR BASELINE VEHICLE

65K STS FOR $i = 28.5^{\circ}$

ARG OF PERIGEE = 220°

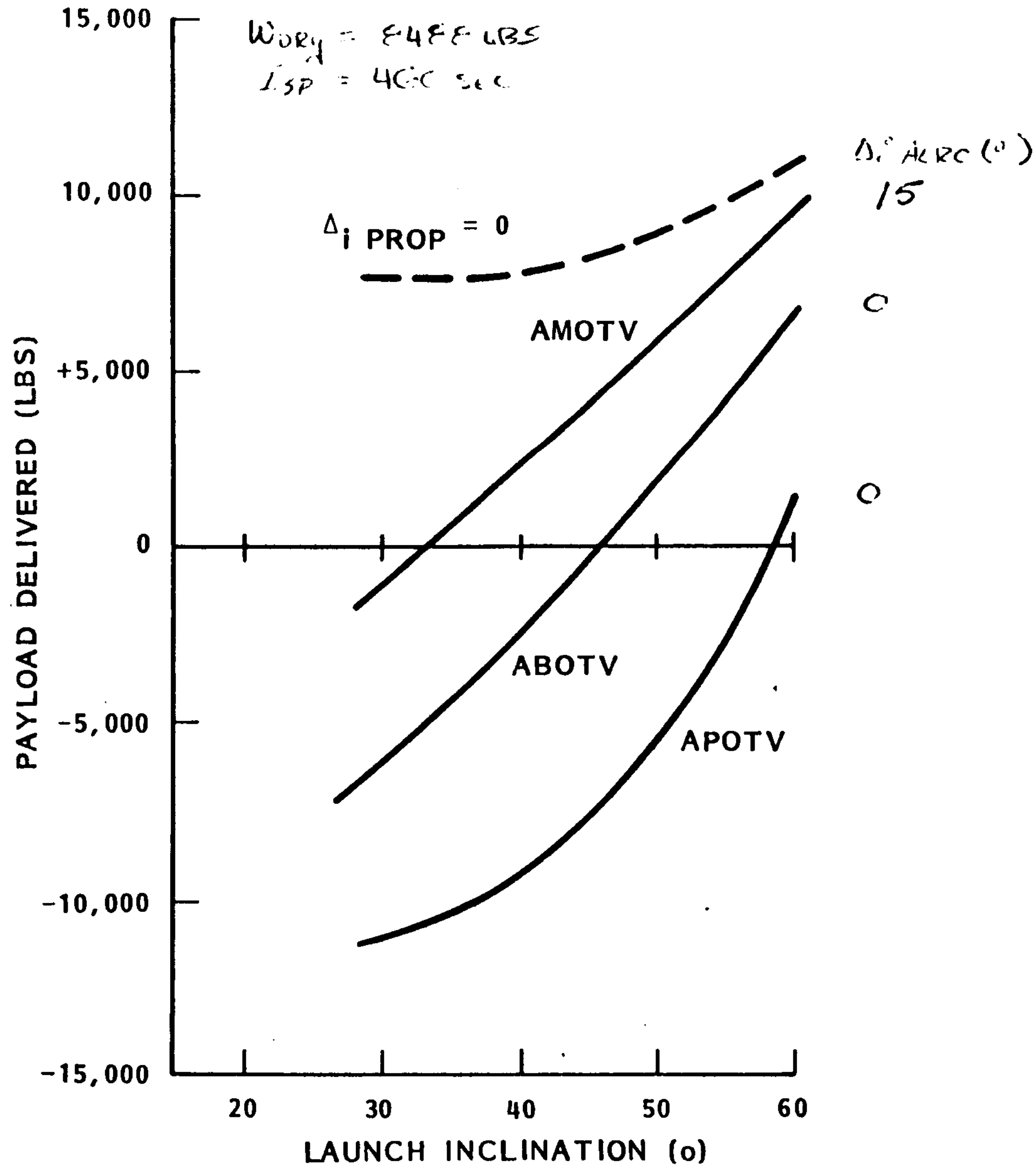
LAUNCH INCLINATION	APSIS ROTATION	L/D = 0	L/D = 1.04	L/D $\longrightarrow \infty$
		$i_{AERO}=0^{\circ}$	$i_{AERO}=15^{\circ}$	$i_{AERO}=35^{\circ}$
28.5	5°	11,970	14,844	17,279
28.5	35°	8,324	9,500	10,200

CONCLUSION:

RETURN APSIS ROTATION ΔV HAS LARGE EFFECT ON MOLYNIA DELIVERY CAPABILITY

RECOMMEND: LIFT VECTOR USED FOR FLIGHT CONTROL ONLY

Figure 2.5-2 SIX HOUR POLAR DELIVERY
65K STS FOR i = 28.5° & 40°
56K STS FOR i = 59.9°



Roundtrip payload capabilities to GEO were evaluated during the second quarter activity for an advanced STS lift capability of 100K lbs, Table 2.5-2. Initially a 60 ft, L/D = 1.04, single stage AOTV was evaluated. Note that for the estimated AOTV dry weight of 11,300 lbs, only 10K lbs could be round tripped using current propulsion technology (I_{sp} = 443). When an advanced technology propulsion subsystem was assumed (I_{sp} = 480), the round trip payload increased to 12,750 lbs for the 60 ft vehicle. Crew compartment investigations indicated that with usage of the "bare bones capsule" (Figure 2.4-15) the AOTV need not be 60 ft in length, but may be considerably shorter and therefore considerably lighter than 11.3K lbs. However, if the AOTV dry weight is reduced to 9300K lbs (by shortening or introduction of some advanced technology), and an advanced engine with I_{sp} = 477 sec employed, 14K lbs could be round tripped to GEO.

GE AOTV

Table 2.5-2 Manned Payload – LEO to GEO & Return

$i=28.5^{\circ}$

STS Lift Capability(K LBS)	100	100	100	100	1116
ASE (LBS)	4600	4600	4600	4600	4600
L/D	1.5	1.5	1.5	1.5	1.5
Vehicle Length (FT)	60	60	60	50	60
Vehicle Dry Weight (LBS)	11300	11300	11300	9300	11300
Specific Impulse (SEC)	443	460	480	477 (MR=7)	460
Payload Weight (LBS)	10050	11300	12750	14000	14000

One Improved STS

After initial configuration screening, three were selected for further evaluation that spanned the mid L/D range, Figure 2.3-10. Mass estimates were made for these configurations employing the original scaling relationships discussed in Section 2.3.1. With a 10% contingency added, the estimated weights were 8103, 8535, and 9377 lbs for the L/D = .75, 1.04 and 1.5 vehicles respectively. Employing the aerodynamic plane change capability presented in Section 2.1.3, the payload delivery capability to GEO is illustrated in Figure 2.5-4. Note the decreasing payload capability with increasing L/D for these three vehicles. This results from the combination of two main factors. The increased AOTV L/D was obtained in this case, by lengthening the vehicle and the structure and TPS mass estimates, which at this point were based on vehicle length and not surface area.

A single stage 38 ft GEO delivery vehicle with propellant tanks sized for a mixture ratio of 7 and a single engine has been described. Except for the advanced engine (I_{sp} = 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. The improved payload delivery of these lighter vehicles is illustrated in Figure 2.5-4, and compared to previous AMOOS results, Reference 9.

The largest effect of L/D on payload delivery capability is for the polar orbits. The impact of AOTV dry weight on polar delivery capability was evaluated and is summarized in Figure 2.5-5. As can be noted, more L/D increases payload delivery capabiity only if it can be added at an acceptable weight increase.

Figure 2.5-4

SINGLE STAGE GROUND BASED VEHICLES

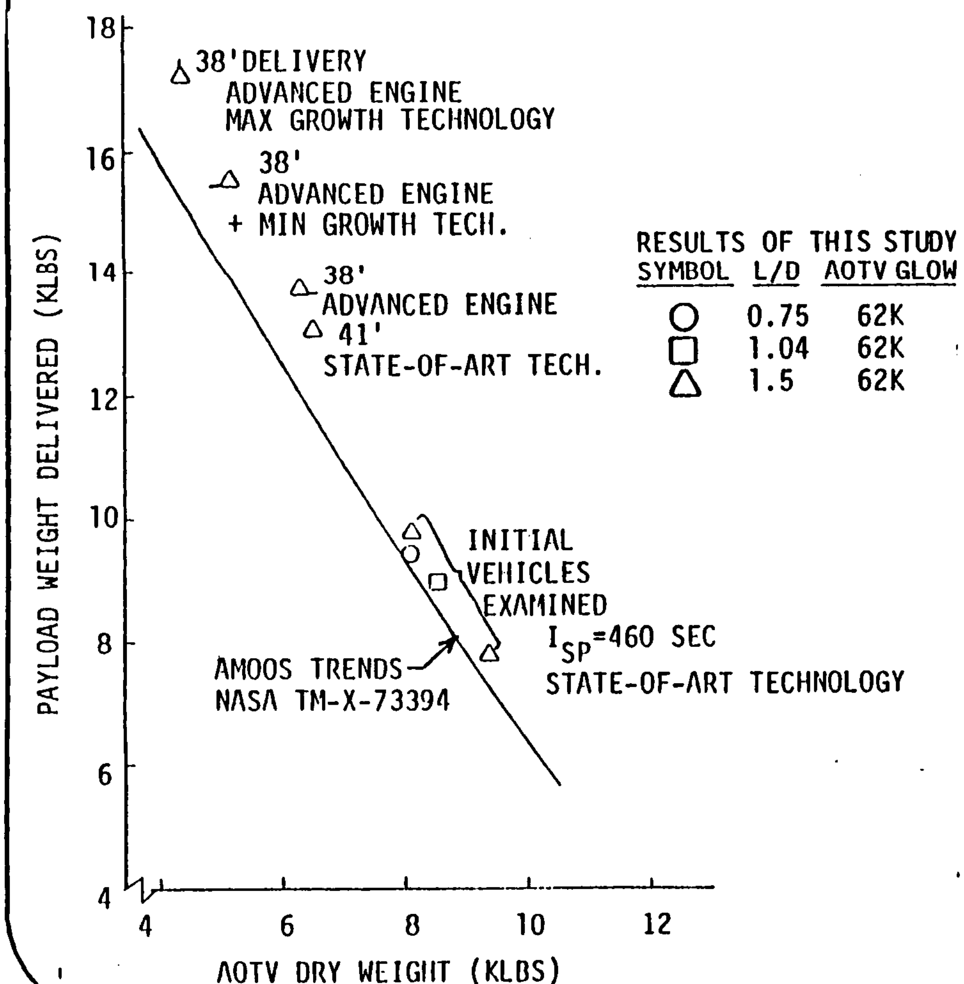
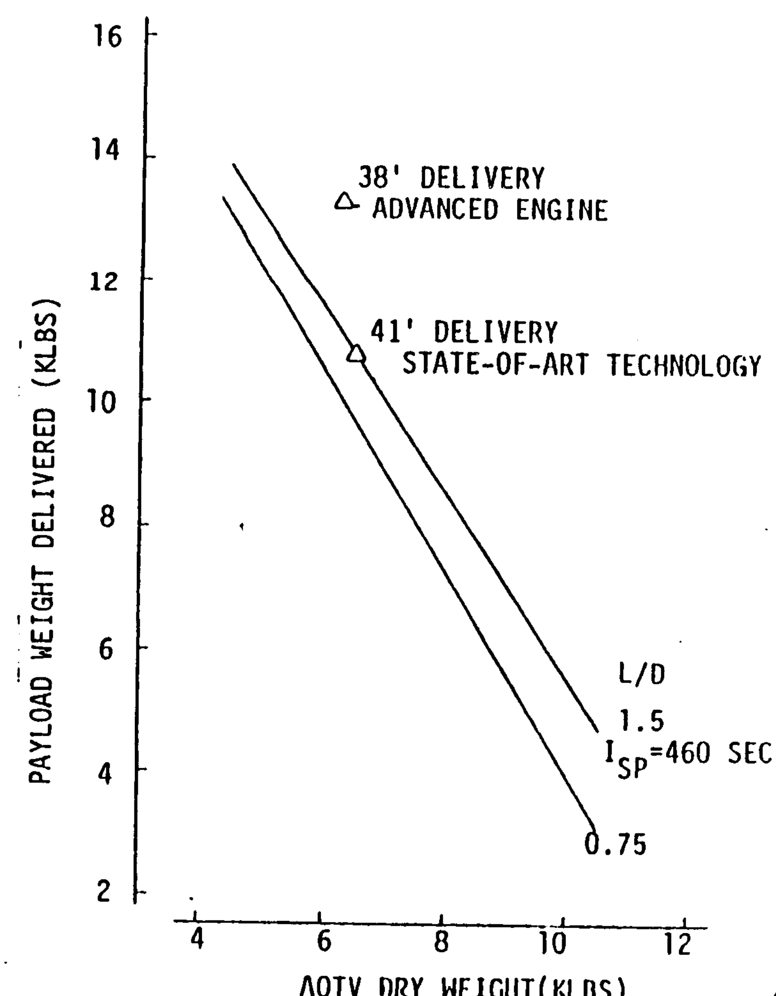


Figure 2.5-5

6-HR POLAR DELIVERY CAPABILITY
SINGLE STAGE GROUND BASED VEHICLES
$i=59^0$



Configuration variations of the 38 ft internal tanked GEO delivery vehicle were explored for a Six Hour Polar Mission to determine effect on payload weight/length, Figure 2.5-67. Here, it is noted that incorporating an aft conical frustum angle of 1° results in increased payload length. Lesser 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. Longer allowable 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 area 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. Another first order conclusion is that even though the L/D = 1.5 biconic is heavier than the L/D = .75, the payload delivered weight capability is 10% greater. This trend is opposite to the trend observed historically, and should be evaluated with more detailed computations in the areas of AOTV structure and external TPS mass trends.

DELIVERED PAYLOAD WEIGHT & LENGTH FOR INTERNAL TANKED AOTV – SIX HOUR POLAR

Figure 2.5-6

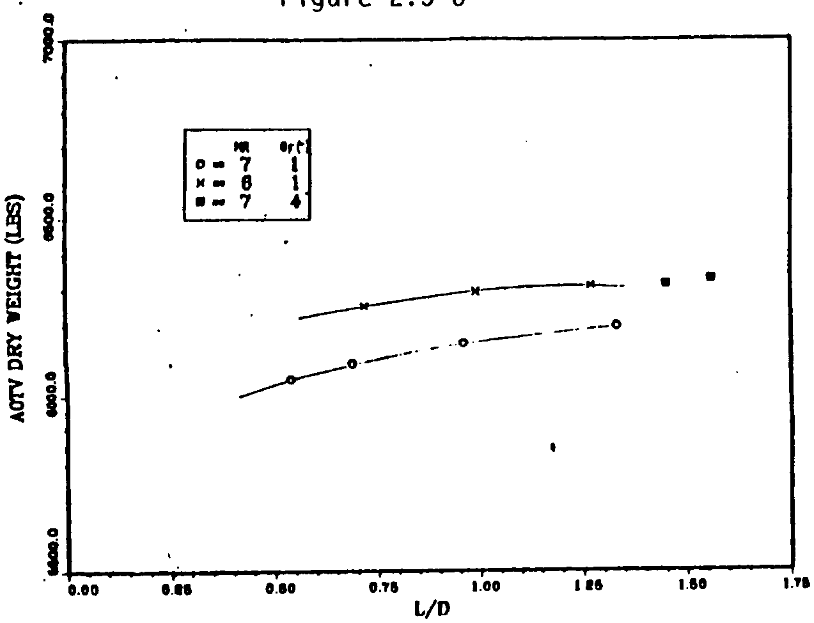
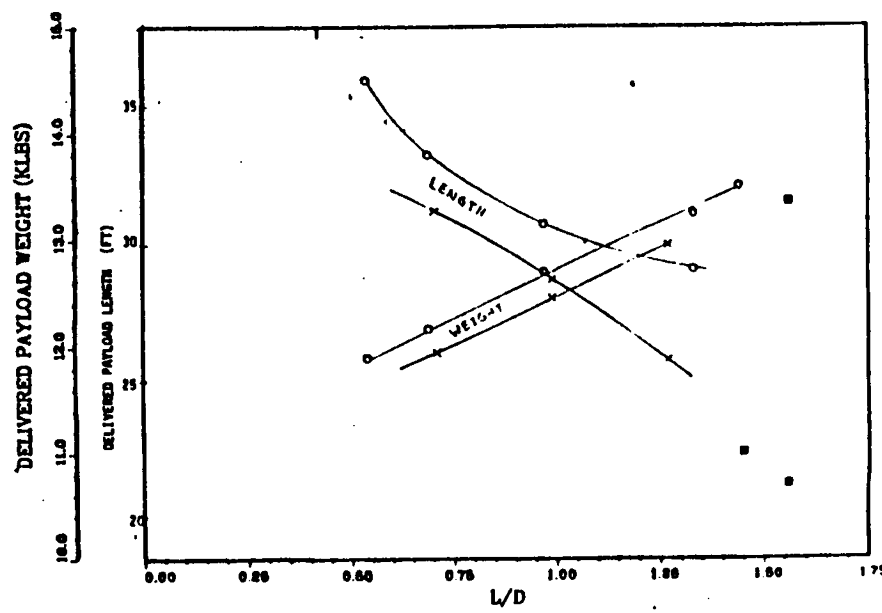


Figure 2.5-7



Comparisons have been made between the payload capability results generated in this study and reference results from previous studies. The trends of increasing payload capability with decreasing AOTV dry weight are presented in Figures 2.5-4 and 2.5-8. The trends of increasing payload capability with increasing shuttle lift capability are presented in Figure 2.5-9.

Figure 2.5-8

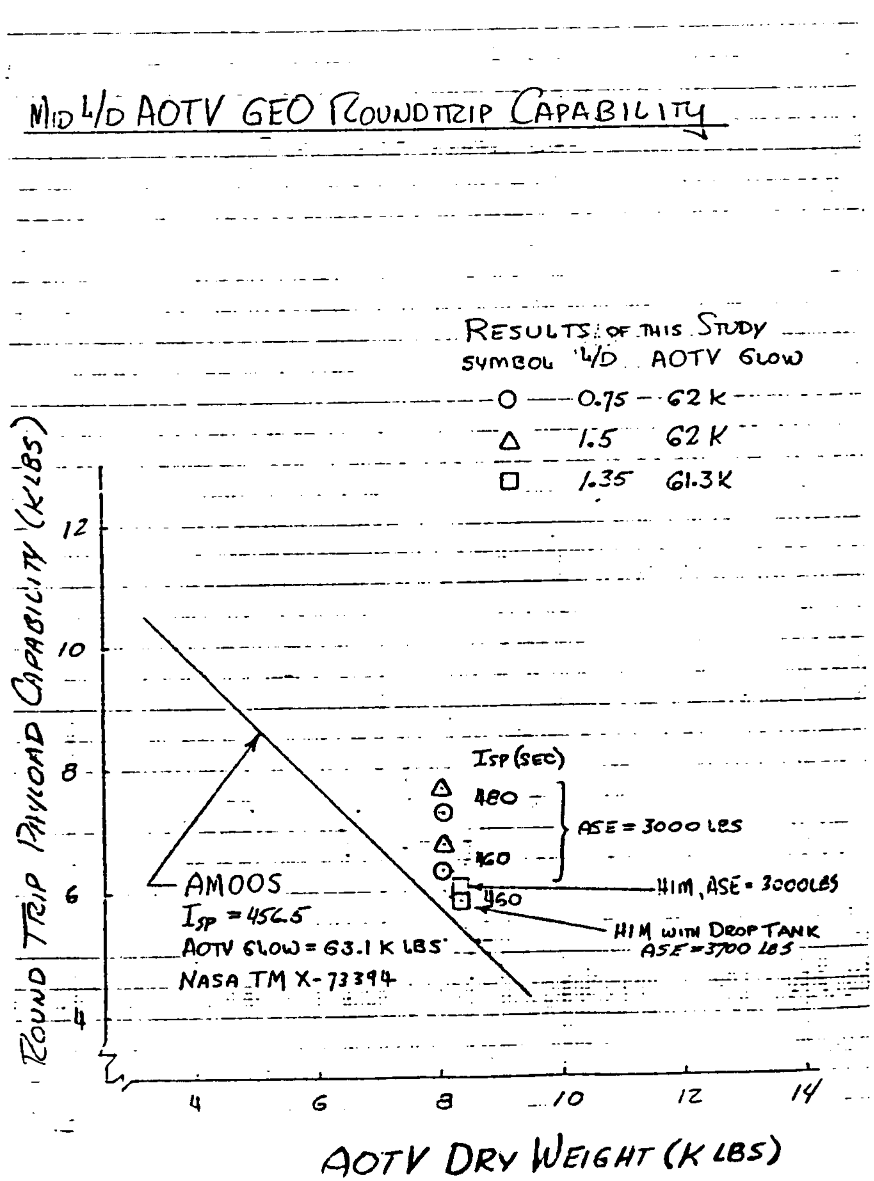
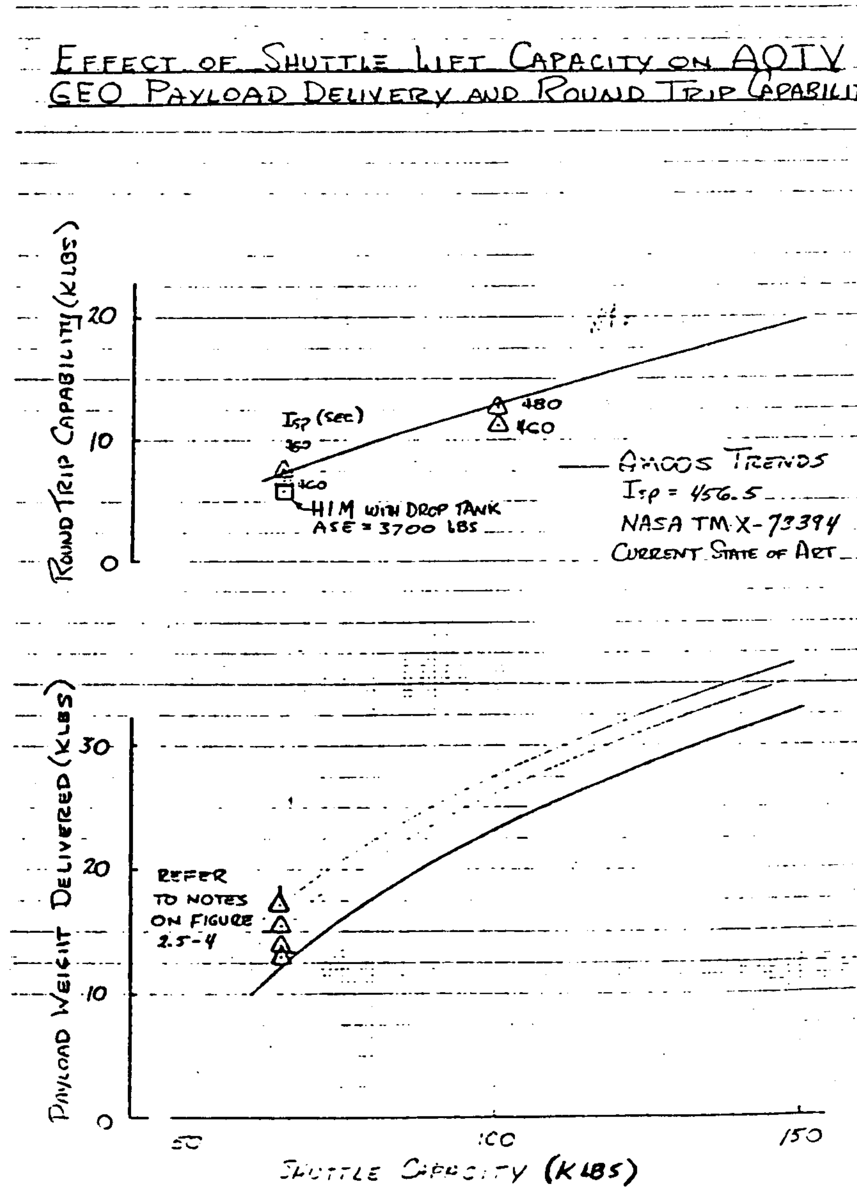


Figure 2.5-9



Flight performance and payload delivery sensitivities across the mid L/D range for a single stage AOTV are summarized in Table 2.5-3. The incremental increase in payload delivery capability, given a reduction in vehicle dry weight, or an increase in engine specific impulse or an incremental 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, ΔW P/L/ΔL/D, between the GEO and 6 hour polar delivery missions.

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

Table 2.5-3

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
		6 HR POLAR	58	50
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

3.0 SYSTEM/SUBSYSTEM TRADES

3.1 Man-Rating

Man rating is a system of obtaining assurance of mission success and crew safety through the use of alternate methods or paths for performing required technical activities (redundancy). In general, a minimum of one failure, and usually at least two failures, are tolerated without endangering the crew. Historically, the level of redundancy needed for man rating has been selected by the management for each particular program, Figure 3.1-1.

For this AOTV study, we have selected a "fail-safe, fail-safe" criteria. This implies that a manned AOTV will allow safe return of a crew after experiencing any (except as outline below) two independent, non-explosive failures. This requirement is modified in the traditional way, namely that structure, pressure vessels (lines and tanks) and Thermal Protection System (TPS) passive components are exempt from the redundancy requirements, but they will be designed to the higher margin of safety (1.5) that is associated with man rated components.

In practice, implementation of a "fail-safe, fail-safe" system usually results in a "fail-operational, fail-safe" system.

Figure 3.1-1 WHAT IS MAN-RATING?

- REDUNDANCY PLUS HIGH RELIABILITY
- SYSTEMS WHICH TOLERATE TWO FAILURES
- REFERENCE: JSCM 8080 "MANNED SPACECRAFT CRITERIA & STANDARDS", STANDARD NUMBER 12A (REV 5/19/75)
  - EXEMPTS STRUCTURE, TPS AND PRESSURE VESSELS
  - FOR CRITICAL FLIGHT VEHICLE SUBSYSTEMS, MINIMUM REDUNDANCY REQUIREMENT IS "FAIL SAFE"
    - FAILURE OF ANY SINGLE PIECE OF SUBSYSTEM HARDWARE WITHOUT LOSS OF LIFE OR VEHICLE
- SHUTTLE DESIGN PHILOSOPHY (FOR SOME COMPONENTS)
  - FAIL OPS/FAIL OPS/FAIL SAFE
- SHUTTLE PAYLOAD REQUIREMENTS
  - FAIL SAFE/FAIL SAFE

The addition of redundant components in man rated vehicles increases the total number of parts in the vehicle. This increases the probability of failure of some part within the man rated vehicle (relative to a vehicle designed with a non-redundant philosophy). Thus, degraded system reliability results (more time spent in repair). However, the redundant comonents significantly improve the probability of mission success (improved mission reliability). The smaller number of aborted or lost missions is a major cost advantage of man rated systems.

The satellite user community has some economic incentives favoring high mission reliability which are difficult to quantify or cover with insurance. One of these is related to the very significant investment which exists in ground stations which support communication satellites. When a satellite launch fails, a replacement satellite cannot utilize the ground stations for many months or years. Consequently, investment in the ground station is (particularly) wasted during the time it takes to get a replacement satellite in orbit.

Another intangable loss incurred by the satellite user community when a launch failure occurs is the loss of new business opportunities. Existing COMSAT capacity of a particular owner will be taxed or exceeded when a new COMSAT launch fails. Sales growth of the parent company will be limited until additional capacity is placed on orbit.

Considerations like these, Figure 3.1-2, indicate a strong benefit will accrue to satellite owners and users if high mission-reliability upper stages are used to deploy satellites. Implementation of most man rating criteria will produce enhanced mission reliability at a small penalty to operational costs. A reuseable AOTV which is designed to a fail-safe, fail-safe criteria should be welcomed by the COMSAT community.

Figure 3.1-2 ADVANTAGES OF MAN-RATED VEHICLES

- MAN-RATED VEHICLES HAVE A HIGHER EXPECTATION OF MISSION SUCCESS (DESPITE RELIABILITY NUMBERS)
- SUCCESSFUL SATELLITE DEPLOYMENT VERY IMPORTANT TO COMMUNICATIONS INDUSTRY
  - TYPICAL GROUND SEGMENT COSTS GREATLY EXCEED SPACE SEGMENT (10 X OR 20 X FOR A SINGLE LAUNCH)
  - UNDER UTILIZATION OF GROUND ASSETS IS BAD BUSINESS
    - COST OF $ TIED UP ON GROUND
    - LOST BUSINESS OPPORTUNITIES
  - LAUNCH COSTS (FOR MAN-RATING) ARE A SMALL PART OF TOTAL PROGRAM $

<u>CONCLUSION</u>:

HIGHER CONFIDENCE OF MISSION SUCCESS FROM MAN-RATED VEHICLES SHOULD BE WELCOMED BY USERS DESPITE HIGHER LAUNCH COSTS

At conferences and meetings which occurred during the Phase I AOTV studies, significant differences of opinion arose among members of the engineering community on the subject of man rating an OTV and the minimum number of engines required for a man rated OTV. Figure 3.1-3 summarizes the position of this study group on some of these issues.

A major issue is returning a crew after they have arrived at GEO. Analogies with single engine aircraft (or two engine aircraft) ignore the fact that additional systems aid human survival in an aircraft. Wings allow commercial aircraft to glide to airports or ditch in water after all engines have failed. Military aircraft have parachutes to allow crew escape after all engines fail.

Dedicated rescue vehicles may make sense when manned operations in space are extensive enough to require many manned vehicles in space at the same time. However, throughout this century, that level of activity is not anticipated. For ground based manned missions, it seems absurd to expect an entire shuttle to be in a standby mode for possible launch on a rescue mission with a back-up manned GEO AOTV. Do you cancel the primary mission if technical problems surface with the back up Shuttle? A similar problem occurs with space based rescue while AOTV resources are small in number.

Some of the problems associated with immediate crew rescue are manageable if the crew can take shelter in temporary housing at GEO, a "line shack". A single "line shack" could easily cost $100M to build and another $100M to transport to GEO. It may take the crew of a damaged AOTV 5 or 10 days to move from their location at GEO to the "line shack". If a solar flare occurred during this period, the crew will die. Consequently, additional "line shacks" (at $200M per unit) may be required. These costs are excessive when compared with the small costs (associated with the small weight differences) obtained from multiple engine AOTVs.

The man rating criteria selected by this AOTV study team is fail safe-fail safe for the crew. In most instances, application of this rule results in a fail operational-fail operational system. Certain passive subsystems (which can be adequately tested, with comfortable safety margins) are exempt from this man rating criteria. These traditional exemptions are listed at the bottom of the figure.

Figure 3.1-3

SUMMARY OF POSITION ON MAN RATED AOTV

- CREW AT GEO MUST BE RETURNED TO LEO
- SINGLE ENGINE AT GEO IS UNACCEPTABLE
  - NO WINGS & NO PARACHUTES AT GEO
- CREW PROTECTION OUTSIDE AOTV PROGRAM IS NOT RELIABLE
  - DEDICATED RESCUE VEHICLE NOT CREDIBLE ALL THE TIME
  - TEMPORARY GEO HOUSING ("LINE SHACK") TOO EXPENSIVE
- AOTV TECHNOLOGY STUDY MAN RATING CRITERIA:
  - "AOTV WILL SURVIVE ANY TWO INDEPENDENT NONEXPLOSIVE FAILURES & RETURN CREW TO ORBITER"
    - PRESSURE VESSELS, PRIMARY STRUCTURE, & TPS ARE EXEMPT

FAIL SAFE/FAIL SAFE

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Most past OTV studies did not select a totally new cryogenic engine for their vehicle, Figure 3.1-4. Rather, they selected a modification to the RL-10 which would raise the engine's specific impulse from 447 seconds to 460 seconds.

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Figure 3.1-4 PROPULSION IMPLICATIONS OF MAN-RATING: HISTORY

PAST STUDIES

- MOST OTV STUDIES SELECTED RLIO DERIVATIVE II ENGINE FOR CRYOGENIC UPPER STAGES
  - BOEING, GENERAL DYNAMICS, GRUMMAN, MC DAC & MSFC FROM 1973-1980
- PAST GRUMMAN RATIONALE
  - PERFORMANCE ADVANTAGES OF ADVANCED ENGINES DID NOT JUSTIFY LARGE DEVELOPMENT EXPENSE
  - RL 10 DERIVATIVE II WAS CHEAP COMPROMISE

The acceptance of a minimum man rating criteria like "fail safe" precludes the use of a single engine for manned vehicles. Some members of the engineering community do not want to apply man rating criteria to the main propulsion system. The logic outlined in Figure 3.1-5 is addressed to them, and attempts to use statistical arguments to show that a single engine is not economically viable for manned vehicles.

During the 1979 MOTV study at Grumman, the mortalilty rates of a number of "high risk" occupations (e.g., police, test pilots, etc.) were compared. A decision was made to equate the lifetime career risk of an OTV astronaut (10 flights in MOTV) with the lifetime (>30 years of flying) career risk of the pilot of a commercial airline (>10,000 flights) in order to come up with a quantitative assessment. This yielded a probability of a fatal accident (i.e., a system degradation that would not permit the crew to return to LEO) of 1/50 per flight (or 1 catastrophe in 1000 OTV missions).

During the 1980 OTV study at Boeing, the 1/50 probability of failure (an OTV mission reliability of .99867) was allocated among the various subsystems of an OTV. This allocation produced a reliability requirement, for a single engine OTV, of .99965 for the engine. A single RL-10 derivative IIB engine has a reliability of .99166 for an 8 firing mission. This reliability is inadequate for the manned mission and ".... that dual engine installations are required to meet the reliability goal ... in an abort from GEO operations mode."

During the 1980 OTV Engine Study by Pratt & Whitney, it was determined (in Task 12) that an extordinarily large number of engine firings are required to demonstrate the single engine reliability (.99996) that is necessary to obtain the OTV engine reliability (.99959) that Pratt & Whitney believes is implied by the Grumman MOTV crew member risk of 1 in 50 per flight. As of April 1980, the operational RL10A-3-3 had accumulated over 1400 firings with no failures and had only demonstrated a reliability of .9984 at 90% confidence. To demonstrate a reliability of .99965 (the Boeing reliability value, which is much lower than the .99996 that Pratt & Whitney thinks should be demonstrated) at 90% confidence, about 3200 engine firings are required. This implies about 6 years of testing. Pratt & Whitney stated that "Development resources probably will not permit demonstration of reliability to the degree which is an acceptable risk for a manned OTV".

A summary of the above arguments follows. Even if high levels of risk are assigned to AOTV crew members, the required level of engine reliabilty, for a single engine AOTV, will not be demonstrated during an engine development program. Since proven levels of engine reliability will produce unacceptable risks to a crew on single engine AOTVs, multiple engines (or multiple methods of returning a crew to safety) are required for manned AOTVs.

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Table 3.1-5 PROPULSION IMPLICATIONS OF MAN-RATING: HISTORY (CONT)

STATISTICAL APPROACH

- GAC, NOV 1979, "MANNED ORBITAL TRANSFER VEHICLE", VOL 3
  - RISK TO MOTV CREWMEMBER FOR 10 FLIGHTS SET EQUAL TO CAREER RISK OF COMMERCIAL AIRLINE PILOT
  - MOTV CREW RISK 1/50 PER FLIGHT $\rightarrow$ 1 CATASTROPHE IN 1000 OTV MISSIONS
- BOEING, 1980, "ORBIT TRANSFER VEHICLE CONCEPT DEFINITION STUDY", VOL 3
  - SINGLE ENGINE DOES NOT MEET RELIABILITY REQUIREMENTS OF 1/50 RISK
- P&W, 1980 "ORBIT TRANSFER VEHICLE ENGINE STUDY, TASKS 8-12 REPORT"
  - FUNDING LIMITS WILL PREVENT DEMONSTRATION OF REQUIRED RELIABILITY (0.99959)

CONCLUSIONS:

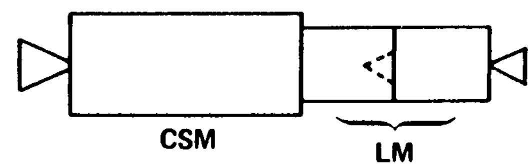
- MULTIPLE ENGINES REQUIRED FOR MAN-RATED AOTV
- RL 10 DERIVATIVE II TOO LARGE FOR MULTI-ENGINE USE ON EARLY AOTV
- PRELIMINARY ENGINE TYPE SELECTION
  - ADVANCED EXPANDER CYCLE, ISP ~ 480 SEC

Some of the Apollo Program history is pertinent to current AOTV man rating discussion, and is outlined in Figure 3.1-6.

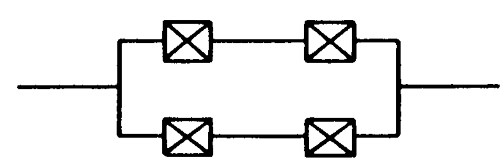
Although the Apollo vehicle only had a single engine designated for each mission phase, other engines were available as back ups for most of the mission duration. In fact, the LM descent engine provided the lunar propulsion that saved the lives of the Apollo 13 astronauts after an explosion in the command Service Module.

During some phases of the mission, only safe operation of a single engine could assure crew survival. To ensure this the Apollo program engines were extremely simple, particularly when compared with the high turbine speed cryo engines that are currently being considered for AOTV. The Apollo engines had no moving parts. They consisted of a combustion chamber and a nozzle. When the hypergollic propellants are admitted into the combustion chamber, they self ignite. The components in the Apollo propulsion system which had moving parts (the valves which controlled the flow of fluids to the engines) were made highly redundant ("quad redundancy") with 4 valves where normally a single valve would suffice.

Figure 3.1-6 **MAN RATING & REDUNDANCY: APOLLO PROGRAM MAIN PROPULSION**



- SINGLE ENGINE FOR MAJOR MISSION FUNCTIONS
  - COMMAND/SERVICE MODULE: CIS LUNAR PROPULSION
  - LUNAR MODULE DESCENT STAGE: DESCENT TO LUNAR SURFACE
  - LUNAR MODULE ASCENT STAGE: ASCENT FROM LUNAR SURFACE
- BACKUP ENGINE AVAILABLE FOR SAFE ABORT FOR MOST PHASES OF MISSION
  - LM DESCENT ENGINE USED ON APOLLO 13 TO RETURN ENTIRE VEHICLE TO LEO
  - LM ASCENT AVAILABLE DURING DESCENT TO LUNAR SURFACE
  - ONLY ASCENT FROM MOON SURFACE HAD NO BACKUP MODE
- EXTREMELY SIMPLE ENGINES
  - HYPERGOLIC BIPROPELLANTS
  - NO MOVING PARTS OR IGNITORS
- PRESSURE FED PROPELLANTS & "QUAD REDUNDANT" VALVES
  - PRESSURE REGULATORS
  - CHECK VALVES
  - FEED VALVES



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3.2 Propulsion Subsystem Trades

LOX-H2 fueled engines of the RL10-3-3A type are considered the baseline current state-of-the-art engine for this study. As part of the Advanced OTV Propulsion System Program currently underway, improvements in specific impulse for LOX-H2 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 3.2-1. Note that the payoff for increased specific impulse is about 60-65 pounds of payload for each second of specific impulse improvement. For the case of a GEO servicing mission where some payload and/or a manned capsule will be returned, the improvement in payload delivery capability is illustrated in Figure 3.2-2.

It is also expected that reduction in engine size/thrust level for operation as multi low thrust engines in a man rated system, and operation at higher mixture ratio to reduce the H2 tank are possible.

Increased Specific Impulse Provides Major AOTV Performance Payoffs for Both GEO & Polar Missions $\ell_{VEHICLE} = 45'$

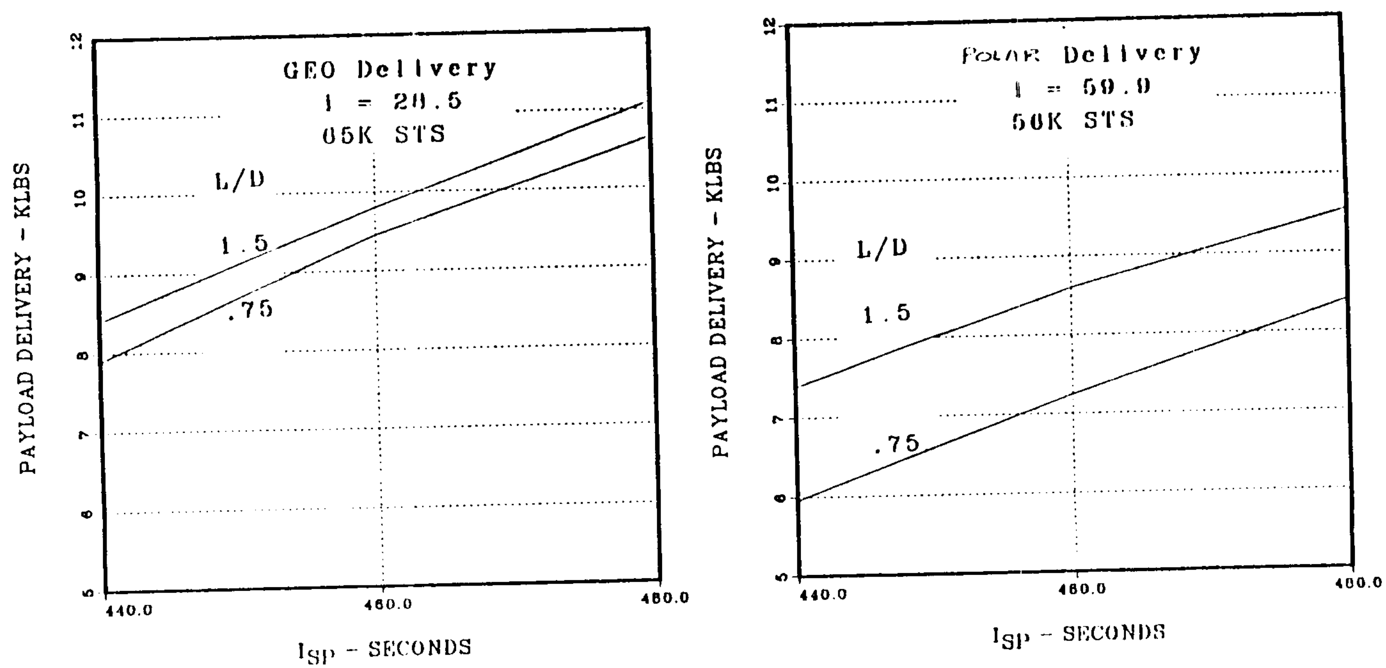
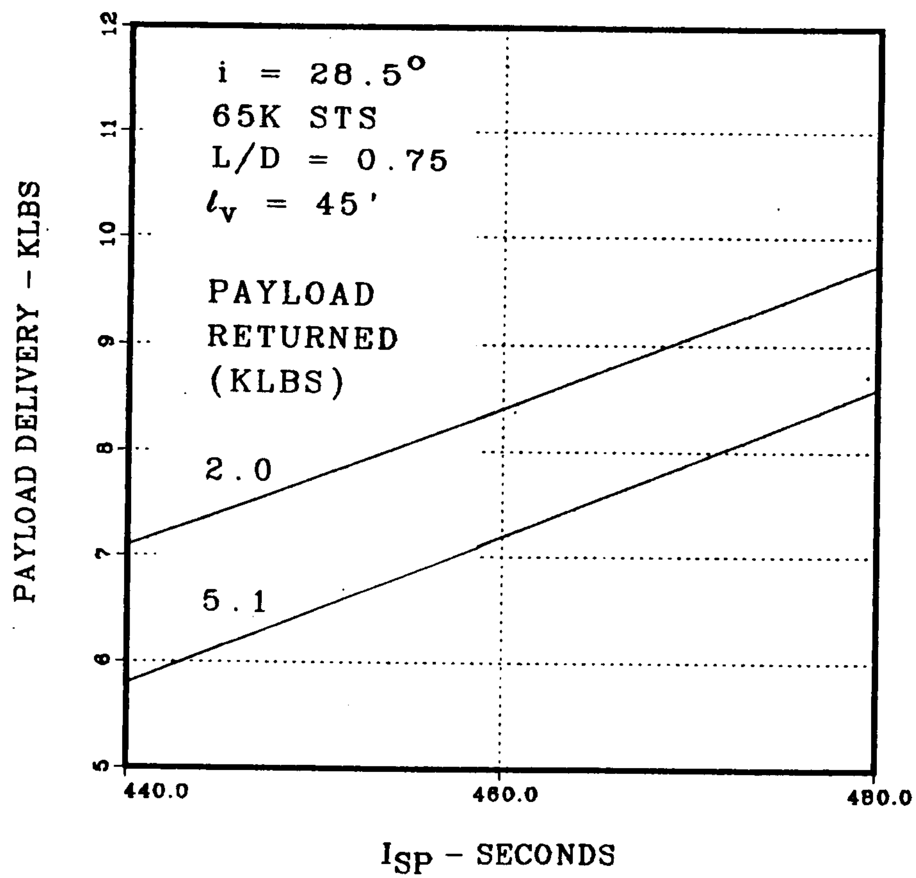


Figure 3.2-1

Increased Specific Impulse Provides Major AOTV Performance Payoffs for GEO Servicing Mission

Figure 3.2-2



The improved shuttle AOTV was sized to accommodate 71,500 lbs of propellant. Several combinations of vehicle performance were evaluated, Figure 3.2-3. It was concluded that using a 100K STS, a 50 ft long AOTV, with weight contingency eliminated (10%) and an advanced cryofueled engine (I_{sp} = 477, MR = 7), 14,000 lbs could be round tripped to GEO with a payload length of 14 ft.

AOTV

Figure 3.2-3 Manned Payload – LEO to GEO & Return

i=28.5°

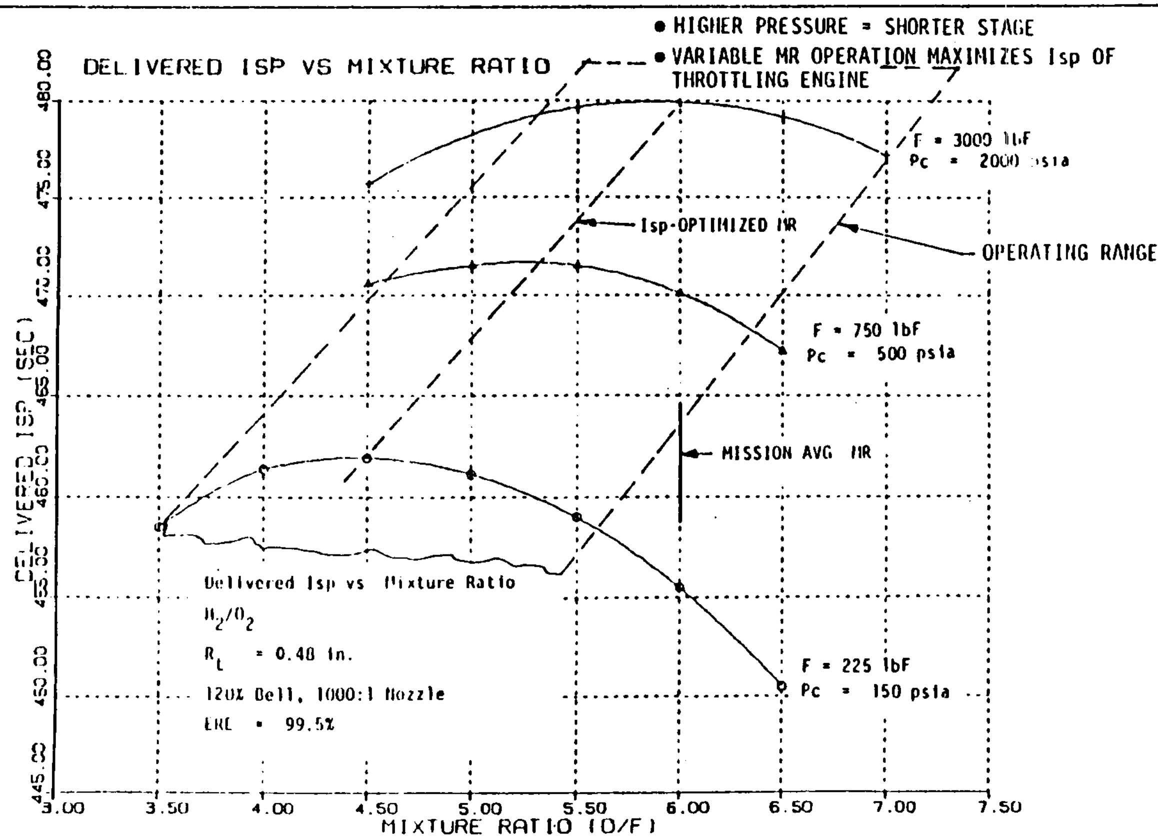
STS Lift Capability(K LBS)	100	100	100	100	$\boxed{111.6}$
ASE (LBS)	4600	4600	4600	4600	4600
$^{L}/_{D}$	1.5	1.5	1.5	1.5	1.5
Vehicle Length (FT)	60	60	60	50	60
Vehicle Dry Weight (LBS)	11300	11300	11300	9300	11300
Specific Impulse (SEC)	$\boxed{443}$	$\boxed{460}$	$\boxed{480}$	477 (MR=7)	460
Payload Weight (LBS)	10050	11300	12750	14000	14000

One Improved STS

The advantage of variable mixture ratio (MR) operation to maximize the specific impulse of a throttable engine was identified (10), Figure 3.2-4. 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, Figure 3.2-5.

Orbit Transfer Rocket Engine Technology Program

Predicted Throttling Performance



Aerojet Liquid Rocket Company
Feb. 1983

Figure 3.2-4

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Effect of LOX/H_2 Mixture Ratio on H_2 Tank Length GEO Delivery

$F = 3000$ LBS
$P_C = 2000$ PSIA

Mixture Ratio	I_{SP} (SEC)	65K STS $W_{p/L}$ (LBS)	65K STS $\Delta l_{H_2 TANK}$ (FT)	100K STS $W_{p/L}$ (LBS)	100K STS $\Delta l_{H_2 TANK}$ (FT)
6.0	480.0	0	0.0	0	0.0
6.5	479.0	−59	-.58	−72	-.96
7.0	477.3	−160	-1.08	−191	-1.79

Figure 3.2-5

During the second quarter activity, an evaluation was conducted to determine the effect of propellant mixture ratio on the propellant volume and the total OTV tank and subsystem volumes. The following parameters were used:

- A GDC detailed design (4) was used to determine the packaging efficiency factors of 1.77 x propellant tank volume.
- The propellant tank volume factor of (propellant volume/0.9) allows space for insulation, baffles and ullage.
- Liquid hydrogen volume is based on 4.4 p.c.f.
- Liquid oxygen volume is based on 70.9 p.c.f.
- 45K pounds of propellant

A volume reduction of 3944-3605 = 339 ft^3, or 9% can be had by changing the mixture ratio from 6.0 to 7.0, Figure 3.2-6.

The volume required analysis of the preceeding chart was extended by adding the calculations required to find the external shell surface area. The shell surface area is directly relatable to the mass of the shell structure and thermal protection.

Increasing the mixture ratio decreases the liquid hydrogen volume required resulting in a smaller vehicle, Figure 3.2-7, with less surface areas, Figure 3.2-8.

Increasing the aft frustum angle causes a loss in packaging volume efficiency and thus greater external surface area, Figure 3.2-8.

The lengths shown are those required to package propellant and mission subsystems. Payload volume is not included.

The following assumptions were used:

- An AOTV vehicle base diameter of 15 feet
- A nose configuration
- Volume factors from Figure 3.2-6
- 45K pounds of propellant

AOTV

Minimum AOTV Volume Required–Geo Delivery

* Representative detailed OTV Design Selected–GDC

* Packaging Parameters Derived:

  OTV Volume(Tanks+Subsystems)= 1.77 x Propellant Tank Volume

  $$\text{Propellant Tank Volume} = \frac{\text{Total Propellant Volume}}{0.9}$$

* Some Typical Results– For 45 K LBS Propellant

Mixture Ratio	6	6.5	7.0
Volume of H_2(FT^3)	1461	1364	1278
Volume of O_2(FT^3)	544	550	555
Total Propellant Volume(FT^3)	2005	1914	1833
Total OTV Volume(FT^3)	3944	3765	3605

Figure 3.2-6

AOTV

Effect of Mixture Ratio and Aft Frustum Angle on Minimum AOTV Length Required—GEO Delivery

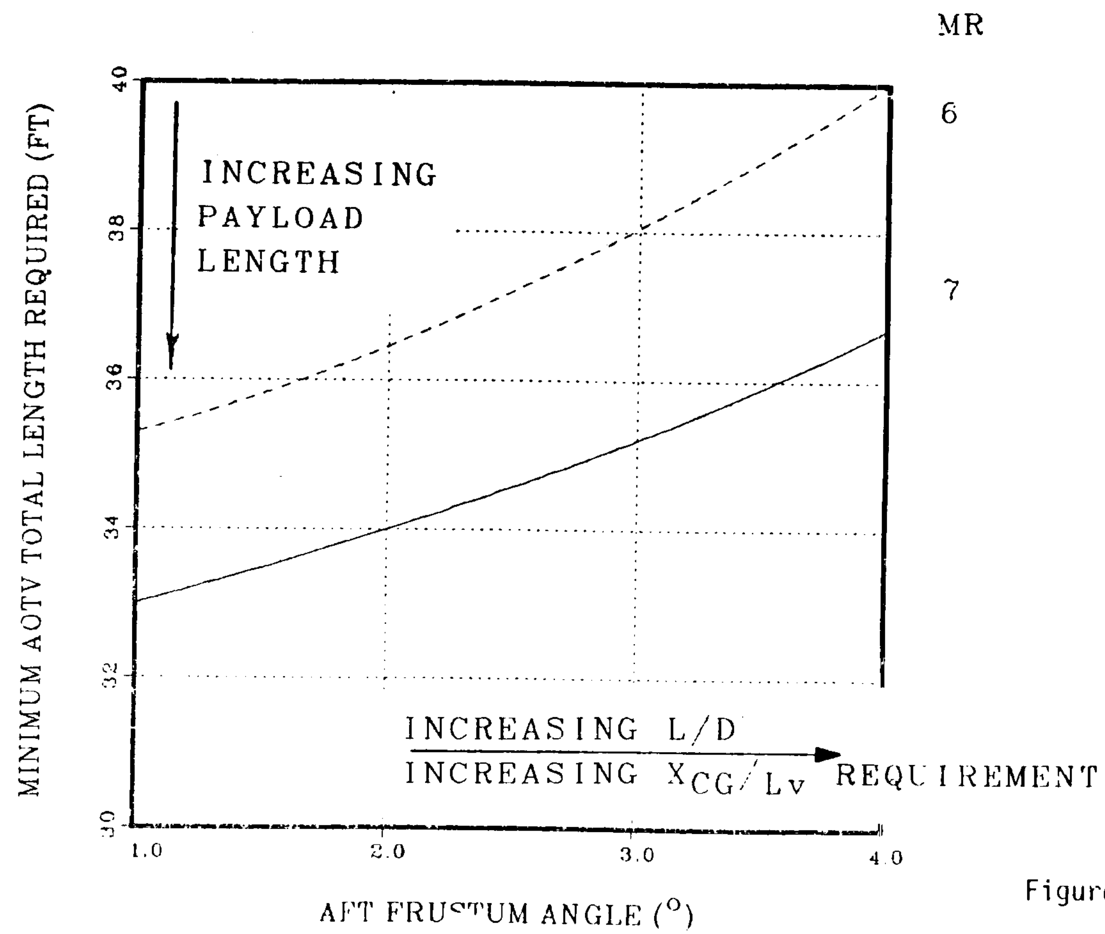

Figure 3.2-7

Effect of Mixture Ratio and Aft Frustum Angle on AOTV Total Surface Area

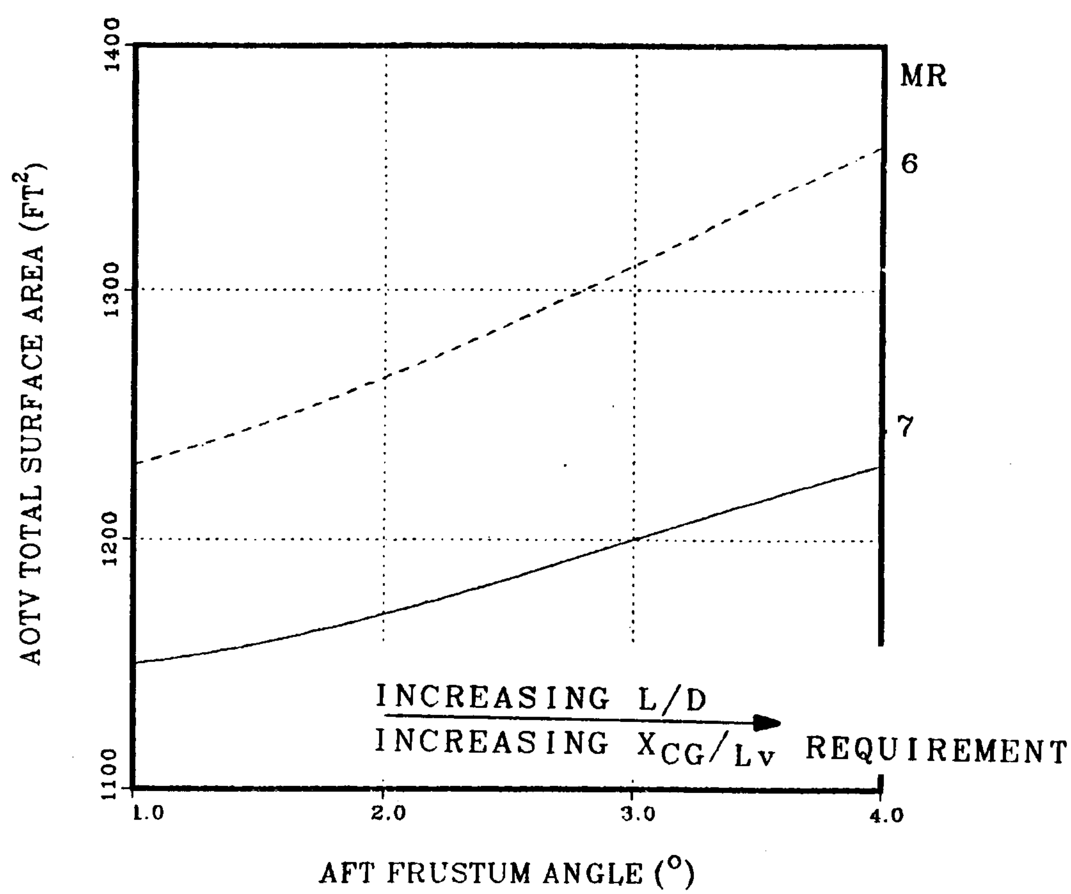

Figure 3.2-8

A rough order of magnitude estimate comparing the propulsion system weight penalty associated with two engines versus a single engine is shown on Figure 3.2-9. A system using two 10,000 lb thrust engines will weigh about 500 lb more than a system using a single 15,000 lb thrust engine. The additional 500 lb buys a "fail operational, fail operational, fail safe" propulsion system. The single 15,000 lb thrust system contains many single point failures which could prevent safe return of the crew.

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Figure 3.2-9 PROPULSION IMPLICATIONS OF MAN-RATING

FAIL OPS/FAIL OPS/FAIL SAFE	ROM WT OVER 1 RLIO DIIB ENGINE
• 2 MAIN ENGINES (E.G., 10K THRUST) + REDUNDANT ACS UTILIZING MAIN PROPELLANT (LO_2 & LH_2)	+ 420 LB
- FOR 2ND ENGINE FAILURE AFTER CIRCULARIZING AT APOGEE, LOSS OF 25 SEC OF ISP FROM USING ACS THRUSTERS CONSUMES 350 LB OF ADDITIONAL PROPELLANT FOR LEO RETURN	+ 20 LB (OVER RESERVES)
• DUAL FAILURE TOLERANT PROPELLANT SUPPLY	
- 25 ADDITIONAL SOLENOID VALVES (MAINLY FOR He LINES)	+ 40 LB
	+ 480 LB

A concept of producing a substantial amount of primary propulsion redundancy at a small weight penalty to the propulsion system is outlined in Figure 3.2-10. A rough order of magnitude comparison of propulsion system weight was made for six small, low thrust, fixed engines versus a single RL10 Derivative IIB engine system. The ROM 170 lb weight penalty for the 6 engine approach was refined in a more detailed analysis (Figure 3.2-11 through Figure 3.2-20) which disclosed a 135 lb penalty to the propulsion system, but a 260 lb weight advantage to the AOTV which uses 6 fixed engines.

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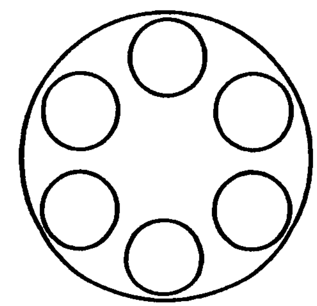
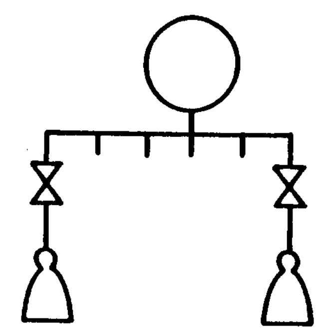
Figure 3.2-10 PROPULSION IMPLICATIONS OF MAN-RATING

FAIL OPS/FAIL OPS/FAIL OPS/FAIL OPS/FAIL SAFE	ROM WT OVER 1 RLIO DIIB ENGINE
• AEROJET CONCEPT: 6 SMALL, LOW THRUST (3000 LB) ENGINES	+ 200 LB
- NON-GIMBALLED	
- FIXED NOZZLES	
- CONTINUOUSLY? THROTTLEABLE	
- EXPANDER CYCLE (EACH PROPELLANT DRIVES ITS OWN PUMP)	
- EXPECTED Isp ~ 480 SECONDS	
• THRUST VECTOR ALIGNMENT WITH VEHICLE CG	- 20 LB
- ALL ENGINES ON: THROTTLE VARIOUS ENGINES (TVE)	
- 1 ENGINE OUT: SHUT OFF OPPOSITE ENGINE & TVE	
- SEVERAL ENGINES OUT: SPIN AOTV ABOUT ROLL AXIS	
• REQUIRED SPIN RATE IS TBD	
• SMALL, SIMPLE VALVE ARRANGEMENT ON EACH ENGINE LINE, LESS COMPLEX THAN MULTIPLE REDUNDANT LINES AND VALVES FOR SINGLE ENGINE SYSTEM	- 10 LB
- NORMALLY CLOSED VALVE	= 170 LB
- 2 FAILURES REQUIRED FOR OPEN FAILURE	





In the engine trade study prsented on the following pages, the effect of varying engine location and number of engines is explored. By considering the effect of the engine tradeoff on the vehicle, not just the propulsion system, we have arrived at a surprising conclusion and a weight optimized solution for a man rated OTV.

The trade study is presented in two parts. Figures 3.2-11 through 3.2-20 present the study of the propulsion system. Figures 3.2-21 and 3.2-22 describe the effects of engine location and quantity of engines on the AOTV.

Figure 3.2-11 shows in graphical form a weight breakdown of a propulsion system for an AOTV. The figures shown are for a ground based cryogenic, hydrogen/oxygen, 40K-propellant weight, internal elliptically tanked OTV with a oxidizer to fuel ratio of 6/1. The trends exhibited by the configuration presented should be representative of most similar OTV configurations. Figure 3.2-12 is a tabular summary of the data presented in Figure 3.2-11.

The weight gain associated with increasing the number of engines with respect to the fill, drain and vent system and the pressurization system is due to the increase in valving required as the number of engines grows. The fluctuation in the values for the propellant feed system weight and the related dip in the propellant feed weight curve in Figure 3.2-11 are caused by a simplification in the fail safe/fail safe propellant line manifolds. This effect is discussed in more detail on page TBD.

The boxed weight figures at the bottom of Figure 3.2-12 are the calculated weight of the propulsion system for the analyzed OTV. As shown in the table the weight figure for the six engine configuration does not include a figure for the weight of the gimbals and actuators. For a six engine fail safe/fail safe system, thrust alignment with stage c.g. can be controlled by throttling selected engines. This can be done for engine failure or payload/vehicle C.G. offset. Therefore, no gimbaling is required for fail safe/fail safe six engine arrangement.

The data presented by the table and graph reveals that the change in weight from the lightest configuration to the heaviest configuration is 185 lbs. Over the spectrum studied this represents only a 7% change from the lightest to heaviest engine grouping. Of more importance is the weight change from one to six non-gimbaled engines. This comparison reveals that the six engine configuration has a weight penalty of only 5% when compared with the one engine configuration. We strongly prefer the six engine configuration for the man rated OTV because of redundancy considerations and packaging advantages.

Figure 3.2-11 **PROPULSION SYSTEM WEIGHT TRADE FOR MULTI ENGINES**

- AOTV MAN-RATED SYSTEMS FAIL SAFE/FAIL SAFE
- MAX Δ WEIGHT $\leqslant$ 135 LB

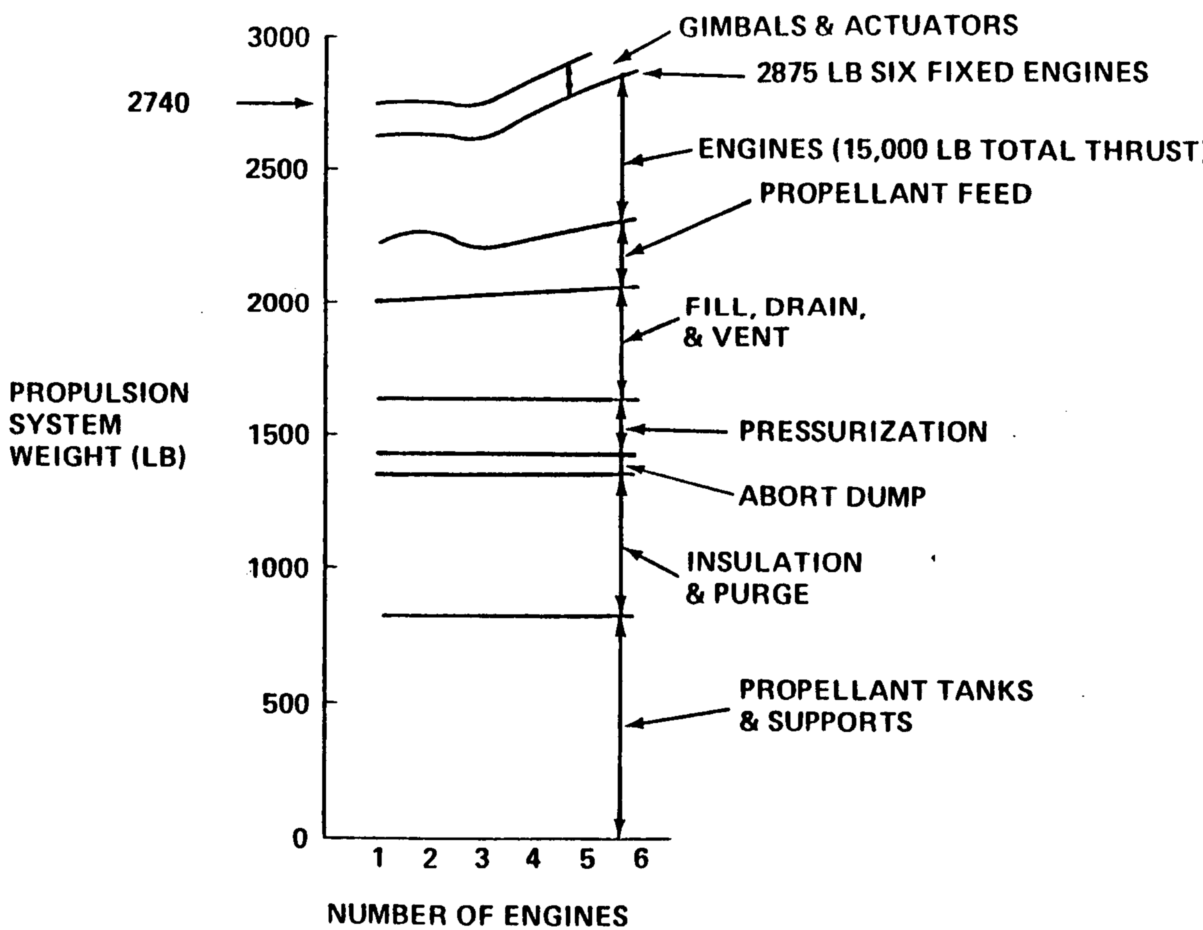


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Figure 3.2-12 **AOTV PROPULSION SYSTEM TRADE**

	WEIGHT (LB)					
NUMBERS OF ENGINES	1	2	3	4	5	6
ENGINE (NO SHUT-OFF VALVES)	408	385	416	467	515	564
PROPELLANT TANKS & SUPPORTS	(824)	(824)	(824)	(824)	(824)	(824)
– LO_2	400	400	400	400	400	400
– LH_2	424	424	424	424	424	424
INSTALLATION & PURGE SYSTEM	(522)	(522)	(522)	(522)	(522)	(522)
– LO_2 (20 LAYERS MLI)	162	162	162	162	162	162
– LH_2 (40 LAYERS MLI)	360	360	360	360	360	360
ABORT DUMP – LO_2 & LH_2	80	80	80	80	80	80
FILL, DRAIN, & VENT	(371)	(381)	(391)	(401)	(411)	(421)
– FILL & DRAIN	98	98	98	98	98	98
– VENT	273	283	293	303	313	323
PROPELLANT FEED	211	231	175	199	227	253
PRESSURIZATION SYSTEM	(207)	(209)	(207)	(209)	(210)	(211)
– TANKS, LINES, GHe	200	200	200	200	200	200
– SOLENOID VALVES (WT/QTY)	7/6	9/7	7/6	9/7	10/8	11/9
TOTAL PROPULSION SYSTEM	2623	2632	2615	2702	2789	$\boxed{2875}$
GIMBALS & ACTUATORS (15°)	118	123	127	132	137	142
TOTALS	$\boxed{2741}$	$\boxed{2755}$	$\boxed{2742}$	$\boxed{2834}$	$\boxed{2926}$	3017

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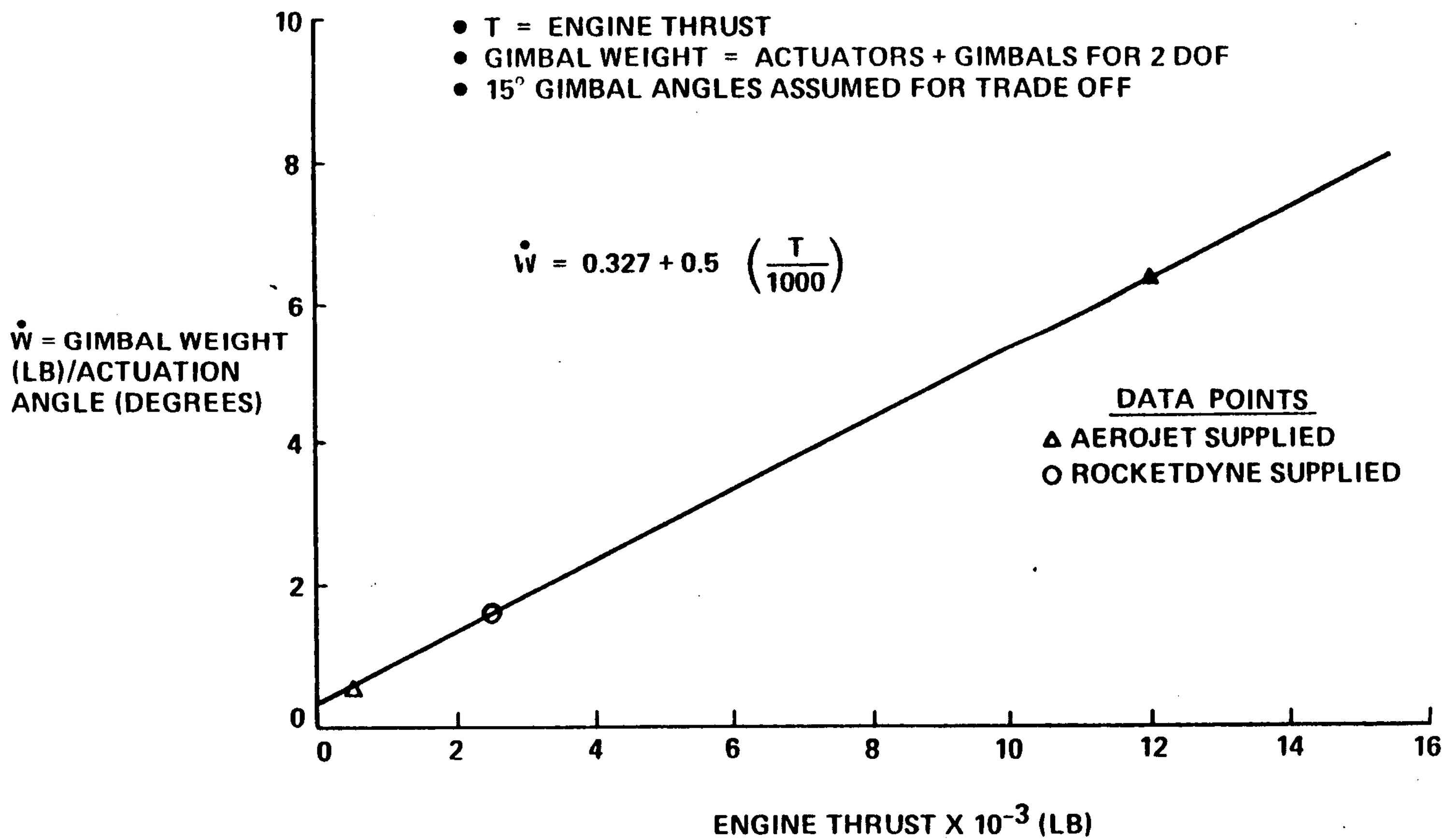
Figure 3.2-13 presents a curve derived from manufacturer's data for engine gimbal weight divided by actuation angle as a function of engine thrust. The derived equation of the curve is used in the previously presented propulsion system trade to define the subsystem value for gimbals and actuators of the engine configurations explored in the tradeoff study.

Figure 3.2-13 GIMBAL SYSTEM WEIGHT FOR PROPULSION TRADE



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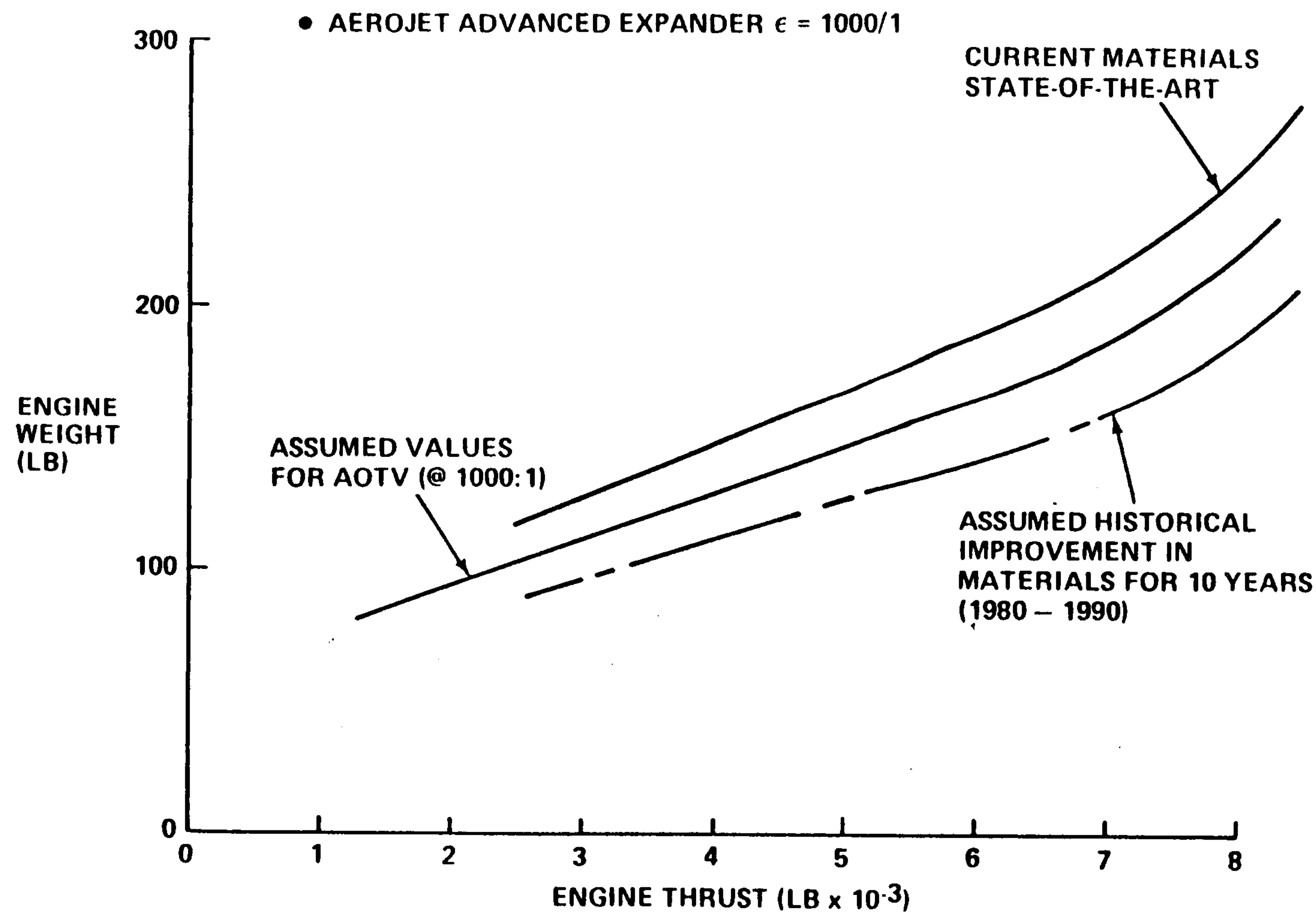
Data for two of the three curves of Figure 3.2-14 were supplied by Aerojet Tech. Systems Company. They represent their estimates for engine weight as a function of thrust for current state-of-the-art materials and for engines made of materials which will be available in 10 years, based on an historical improvement in material properties.

For this engine trade study, we have used the central curve of Figure 3.2-14 for engine thrusts from 2500 lb to 7500 lb. Total engine thrust is 15,000 lb for all systems in this study. Weight data on the single 15,000 lb thrust engine was based on the Pratt & Whitney Advanced Expander engine design.

Figure 3.2-14 ENGINE WEIGHT vs THRUST



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The propellant feed system weight analysis was performed on a representative 40K propellant weight, six to one O/F ratio, LO_2/LH_2 OTV by varying engine location, the number of engines, and holding tank locations, specific impulse, and total thrust constant. Line sizes are based on required mass flow and pressure drop for a one engine configuration and then scaled to maintain the same line velocity in multiple engine configurations. Manifold assembly weights are based upon the weight of the required number of valves, flex sections and tubing length needed to construct the propellant line manifolds that are the functional equivalent of the logic maps in Figure 3.2-16, -17, and -18. The line lengths are based upon a preliminary mapping of the propellant feed lines for the various engine configurations. Flange, valve, and flex section weights are based upon manufacturer data.

The minimum propellant feed subsystem weight occurs on the 3 engine configuration, Figure 3.2-15. The driving factor of the weight loss between the one and three engine configurations is the drop of the line manifold weight. This effect can be understood by studying Figure 3.2-16, -17 and -18. The additional engines, and the feed lines associated with them, reduce the required complexity and size of the line manifolds. This leads to a reduction in number of expansion joints, number of valves, and a shortening of tubing required to construct the required propellant manifolds. It should be noted that the three engine configuration has the lowest number of engines of any configuration incorporating the single flow path manifold and, because of this, has the lowest propellant feed subsystem weight.

The second major factor affecting propellant feed system weight is the increase in the required number of components as the number of engines grows. Although the weight of a component decreases as it beocmes smaller, the total propellant feed system weight tends to increase because more components are required as the number of engines moves upward.

Understanding the shape of the weight curves for the propellant feed system over the entire spectrum of one to six engines requires a superposition of the two factors discussed in the previous paragraphs. From two engines to three engines, the decreasing weight due to a simplification of the manifold cancels the weight increase assoicated with increasing the number of components required in the multi engine configurations. From three to six engines, no further simplification of the valve manifold is possible and weight increases due to increased numbers of components.

The change in weight between the minimum weight 3 engines configuration and maximum weight 6 engine configuration is 77 lbs. This weight difference, although not insignificant when compared to the weight of the propellant feed system, is small when compared to the total weight of the propulsion system taken as a whole.

Figure 3.2-15 PROPELLANT FEED ANALYSIS SUMMARY

NUMBER OF ENGINES	1	2	3	4	5	6
TUBING SIZE, OD (IN.)	3.25	2.34	1.93	1.69	1.53	1.41
TUBING WEIGHT (LB/FT)	2.23	1.59	1.31	1.14	1.03	0.94
NO. OF VALVES/MANIFOLD	6	4	2	2	2	2
MANIFOLD TUBING LENGTH (FT)	10.6	28.1	0.8	0.7	0.64	0.59
MANIFOLD VALVE WT (LB)	7.66	6.11	5.41	5.0	4.73	4.53
MANIFOLD ASSY WT (LB)	69.6	35.6	11.9	10.8	10.1	9.6
ENGINE LINE LENGTH, O_2 (FT)	2.5	8.0	7.6	7.4	6.9	7.2
ENGINE LINE LENGTH, H_2 (FT)	20.7	13.0	12.6	12.4	12.3	12.2
ENGINE LINE WT (LB)	51.8	33.2	26.5	22.6	19.7	18.2
ENGINE FLNG COUP (LB)	5.0	3.6	3.0	2.6	2.4	2.2
TOTAL MANIFOLD WT (LB)	139.1	71.2	23.7	20.6	20.2	19.2
TOTAL PER ENGINE (LB)	195.9	108.0	53.2	45.8	42.3	39.6
TOTAL LINES & MANIF (LB)	195.9	216.1	159.6	183.2	211.5	237.4
TANK OUTLET VALVES (LB)	15.3	15.3	15.3	15.3	15.3	15.3
TOTAL SUBSYSTEM (LB)	211.2	231.4	175.9	198.6	226.8	252.7
Δ WT		55.5				

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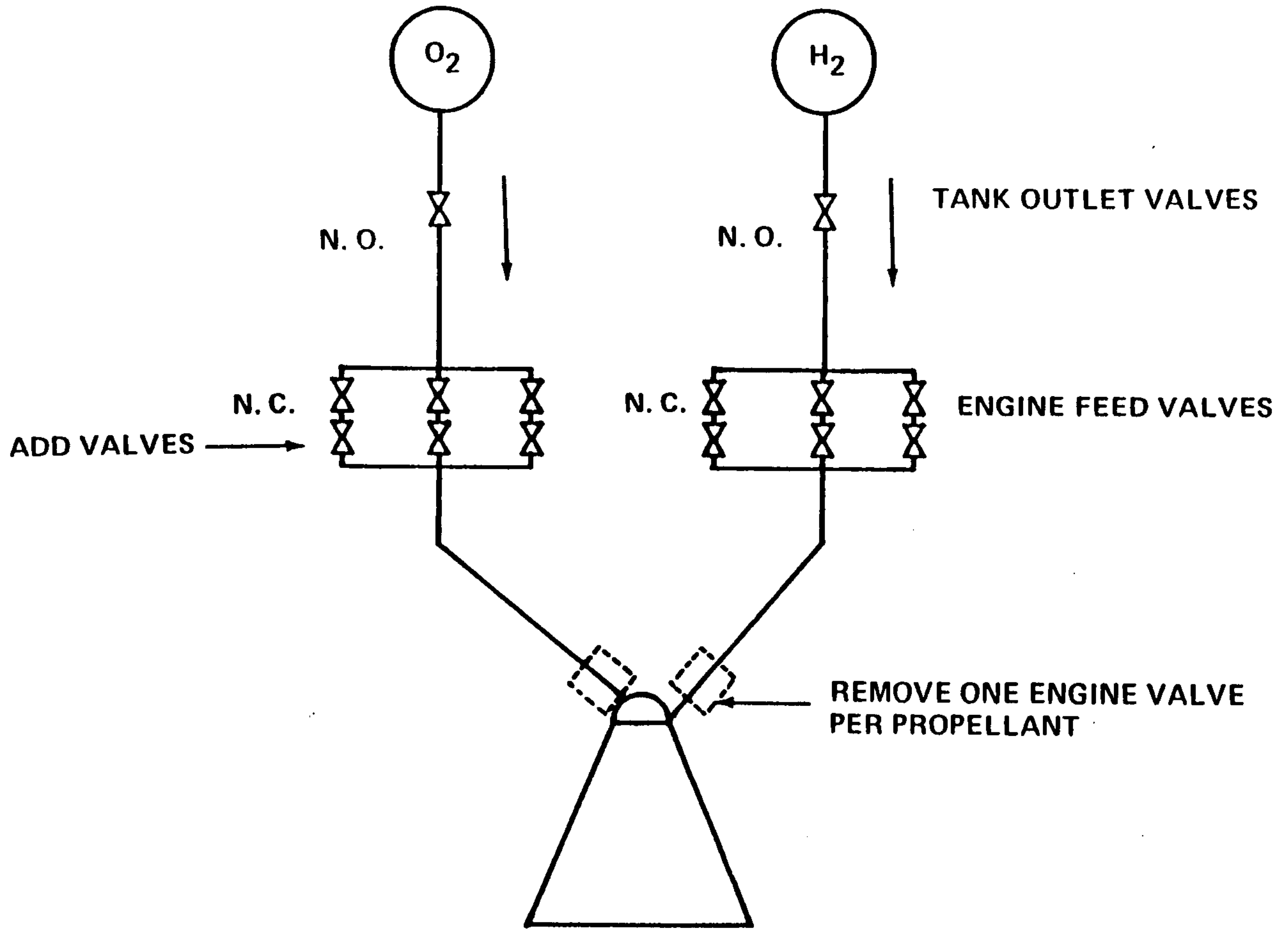
Figure 3.2-16 shows the required plumbing logic for a fail/safe-fail/safe single engine propellant feed system for an AOTV. The assumption made for all propulsion logic diagrams, with respect to electrically related valve failures, is that a valve always fails electrically in the deenergized position. Stated alternately, a normally open valve electrically fails open and a normally closed valve electrically closed. Another mode of failure that must be considered is valve jamming or sticking. In these modes of failure, a valve may fail either in the open or closed position.

The logic presented by the diagram provides for two independent valve failures while maintaining a open flow path for the propellants flowing from the storage tanks to the engine. The purpose of the normally open tank valves is to provide redundancy in the event of a series failure in a propellant line manifold. The normally open valve is energized closed when the OTV is transported in the shuttle cargo bay and provides a third closed series valve. This third valve allows two series valve failures while preventing propellant flow leakage into the cargo bay. The normally open tank valve is provided with a manual override in the event that the tank valve fails closed.

The valve manifold provides three parallel flow paths with two series valves in each flow path. This arrangement insures an open flow path in the event of two independent valve failures and, when used in conjunction with the tank valve, guarantees the ability to stop propellant flow in the event of two independent valve failures.

The diagram as shown relies upon engine redundancy to meet fail/safe-fail/safe criteria to the engine interface. If the engine is not designed to a two failure redundancy level, the type system as shown does not meet a fail/safe-fail/safe criteria. The weight of two additional (small) engines should be added to fairly evaluate this system.

Figure 3.2-16 **PROPULSION TRADE VALVE LOGIC: 1 ENGINE**



LEGEND:

N. O. = NORMALLY OPEN VALVE, FAILS IN OPEN POSITION
N. C. = NORMALLY CLOSED VALVE, FAILS IN CLOSED POSITION

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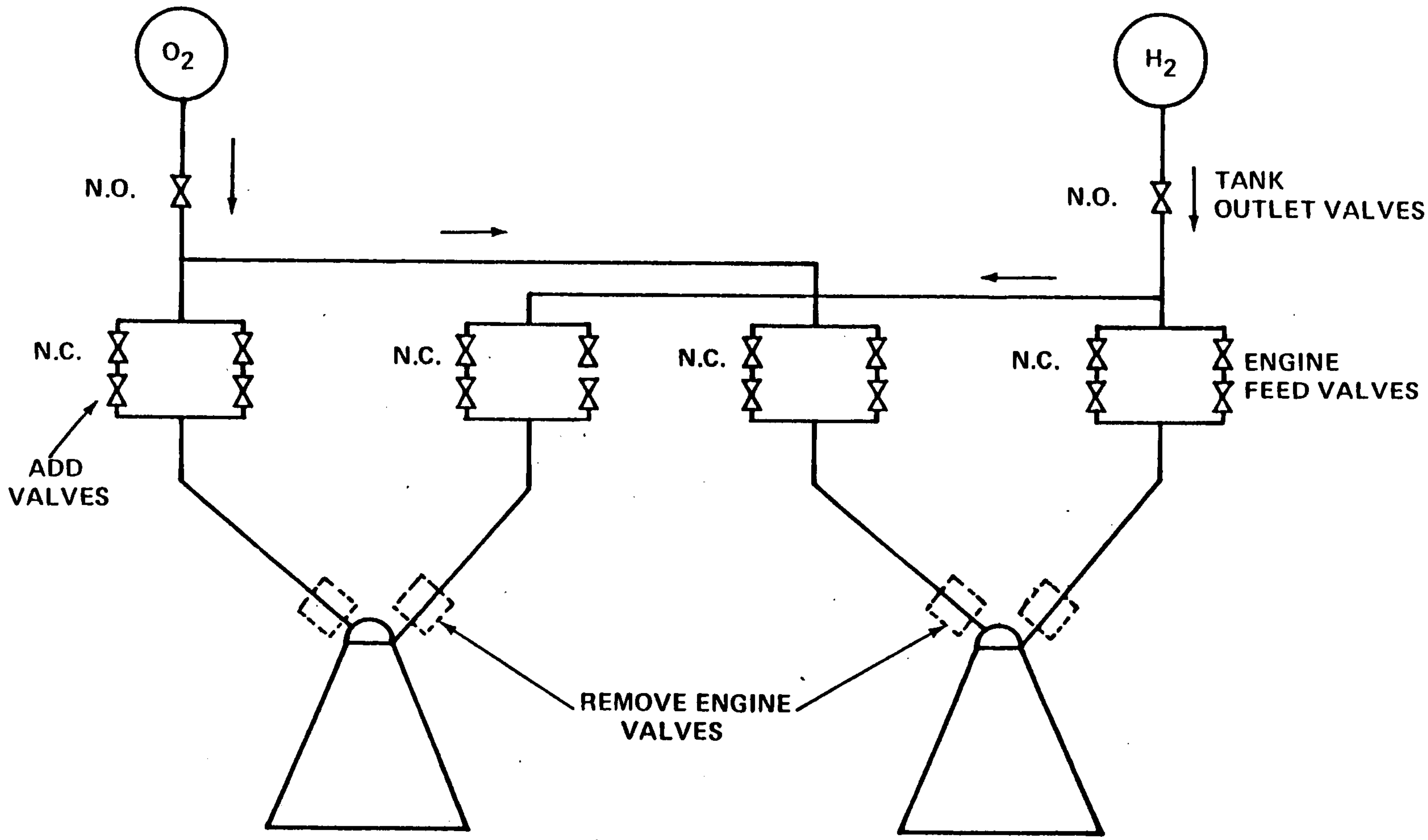
The diagram of Figure 3.2-17 presents the required plumbing logic for a fail/safe-fail/safe propellant feed system for a two engined AOTV. In this configuration, the normally open tank valve serves the same purpose as in the previously discussed single engine configuration.

For the presented logic two independent valve failures can block the flow path to one engine but not two engines. Single engine operation is possible, therefore each engine can act as a redundant engine for the opposing engine. This allows a simplification of the propellant valve manifolds as one of the parallel flow paths in the manifolds may be eliminated.

As discussed in the explanation of the single engine logic diagram, the validity of the claim that the system meet a fail/safe-fail/safe criteria is dependent upon the degree of redundancy in the engines. Failure of two critical engine components, one in each engine that render the engine inoperable, would constitute two independent failures that negate the logic of the propulsion subsystem. Therefore the engines need redundant moving components in order for the system to meet the required fail/safe-fail/safe criteria.

Figure 3.2-17 **PROPULSION TRADE VALVE LOGIC: 2 ENGINES**



LEGEND:

N.O. = NORMALLY OPEN VALVE, FAILS IN OPEN POSITION
N.C. = NORMALLY CLOSED VALVE, FAILS IN CLOSED POSITION

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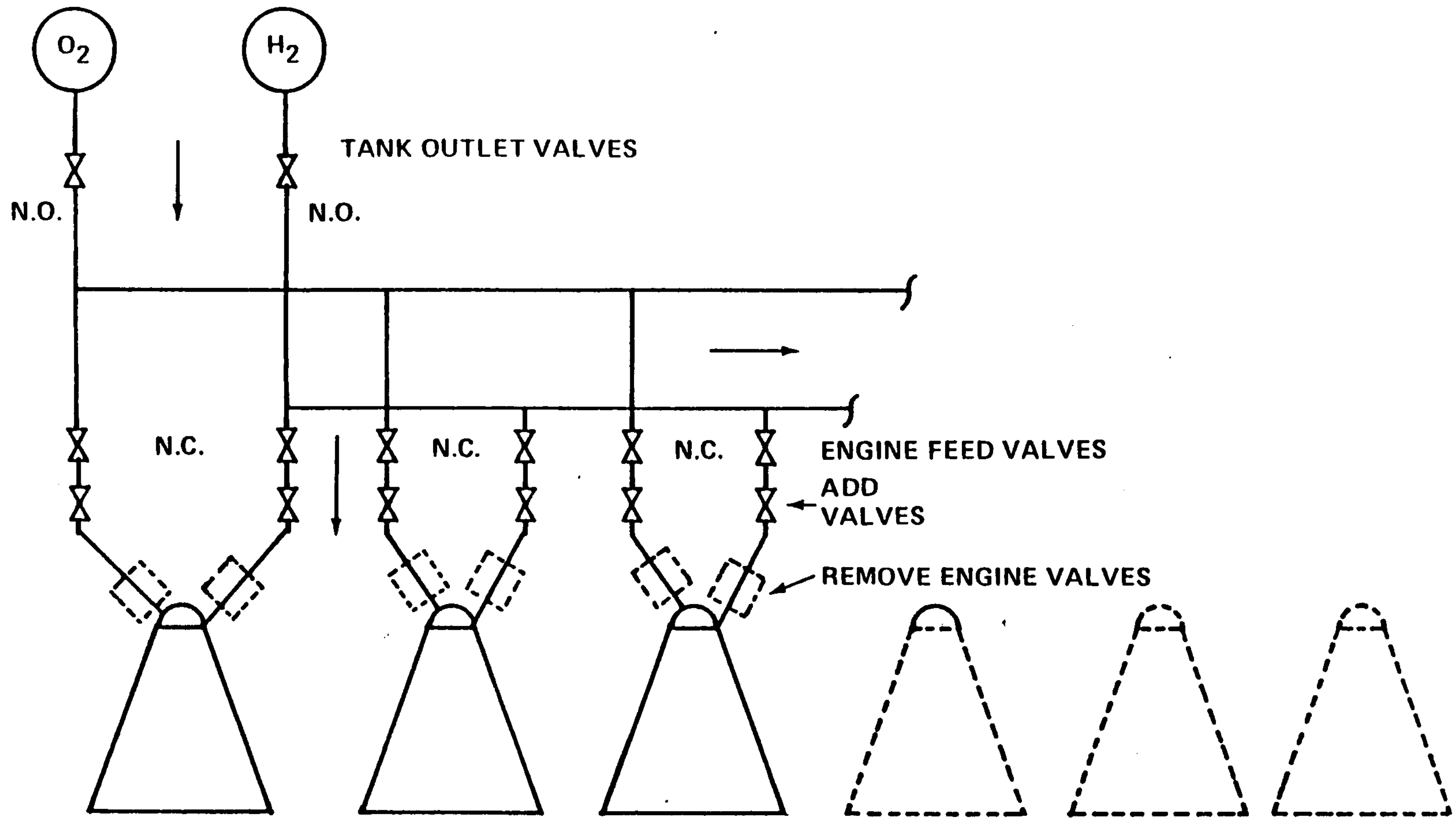
The fail/safe-fail/safe propellant feed system manifolds for three to six engine configurations is further simplified from the one and two engine configurations, Figure 3.2-18. The simplification comes from the fact that the additional redundancy inherent in using 3 or more engines make the event of loss of propellant flow to an engine, or critical engine component failure occurring in one or two engines, a non-critical failure. If the remaining engine(s) have sufficient capability to align the thrust vector with the vehicle system center of gravity, the failure of one or two engines can be tolerated and compensated for by the other engine(s) throttling or gimbaling. Thus only one flow path per propellant is required for each engine.

The normally open tank valve is used to maintain the three series valve arrangement required to insure no leakage into the shuttle cargo bay in the event of a two in line valve failures in a propellant feed line.

The three to six engine configurations meet the fail/safe-fail/safe criteria without redundancy requirements on the engine. The additional redundancy provided by a large number of engines make these configurations preferable to single engine OTV's for man rated missions. The six engine configuration offers the greatest flexibility in compensating for component failures.

Figure 3.2-18 **PROPULSION TRADE VALVE LOGIC: 3 TO 6 ENGINES**



LEGEND:

N.O. = NORMALLY OPEN VALVE, FAILS IN OPEN POSITION
N.C. = NORMALLY CLOSED VALVE, FAILS IN CLOSED POSITION

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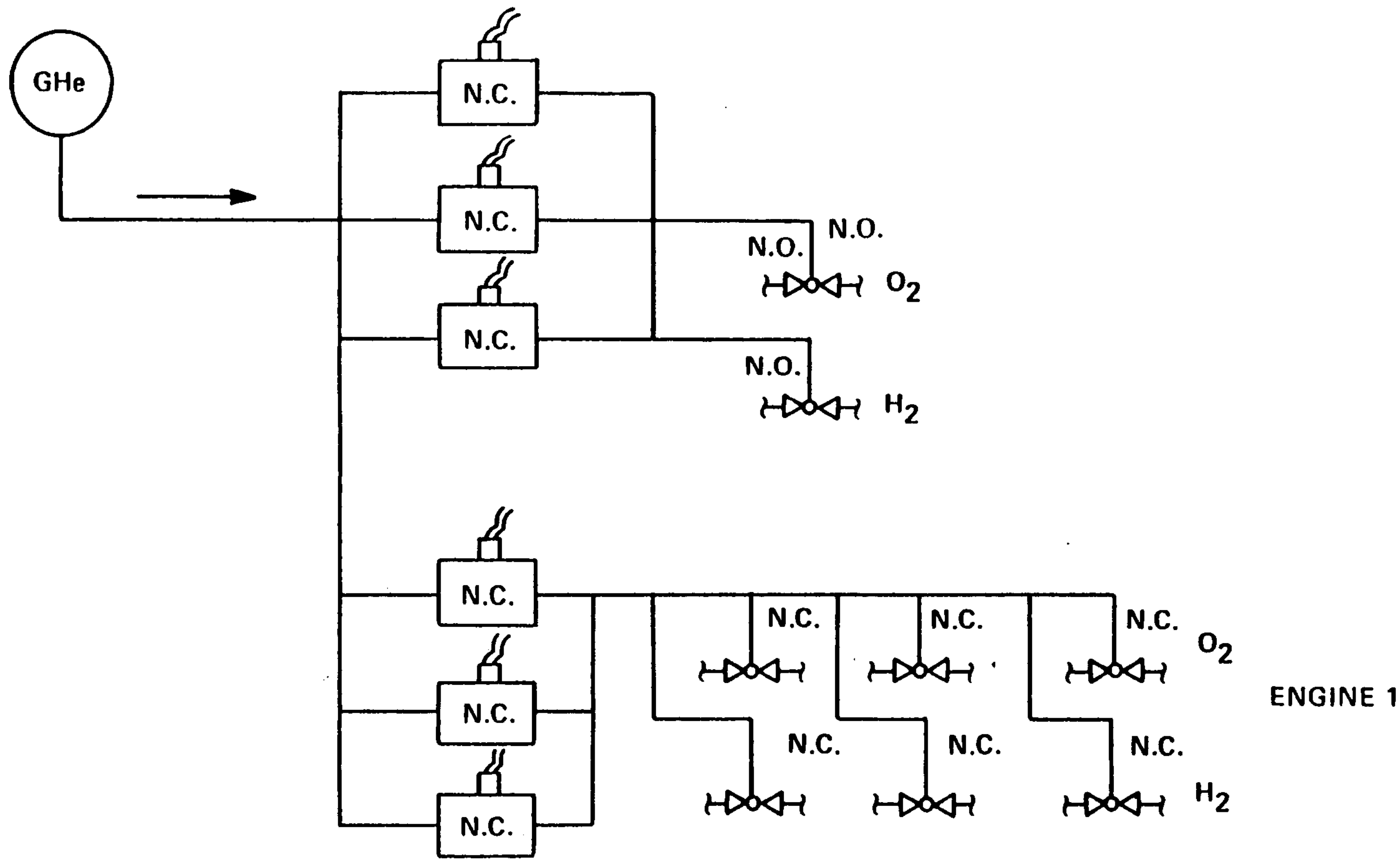
Figure 3.2-19 illustrates the required logic for a helium pressurization system for a fail/safe-fail/safe single engine OTV. The helium gas is used to actuate the engine feed valves, the propellant tank outlet valves, and to maintain pressure level in the propellant tanks are the propellant is depleted.

The solenoid valves provide three independent paths for helium to the stop T valves shown in the pressurization diagram. The upper half of the figure represents the required flow path of the high pressure helium to the propellant tanks. In the upper half of the figure there are two stop T valves. The flow into a stop T valve branches into two flows; one path goes to the helium actuated tank outline valve while the other branch flows to the propellant tank pressurization inlet manifold.

The lower half of the diagram shows the flow path of the helium to the helium actuated valves in the propellant feed line manifolds. The helium flows through the fail safe/fail safe solenoid valve manifold to the stop T valves. In the lower half of the figure, each side of a stop T valve routes flow to a helium actuated propellant manifold feed valve. Opposing sides of the T valves channel flow to valves that are in series with each other in the propellant feed line manifold. This insures that the pressurization system can still provide valve actuation after two independent failures.

The required manifolding to the storage tank for pressurization is not shown in this diagram. Redundant shutoff valves and regulation valves are required to maintain the double level of redundnacy in the system.

Figure 3.2-19 **PROPULSION TRADE PRESSURIZATION LOGIC: 1 ENGINE**



- GHe PRESSURANT SYSTEM FOR ENGINE FEED VALVE ACTUATION

LEGEND:

N. O. = NORMALLY OPEN VALVE, FAILS IN OPEN POSITION
N. C. = NORMALLY CLOSED VALVE, FAILS IN CLOSED POSITION

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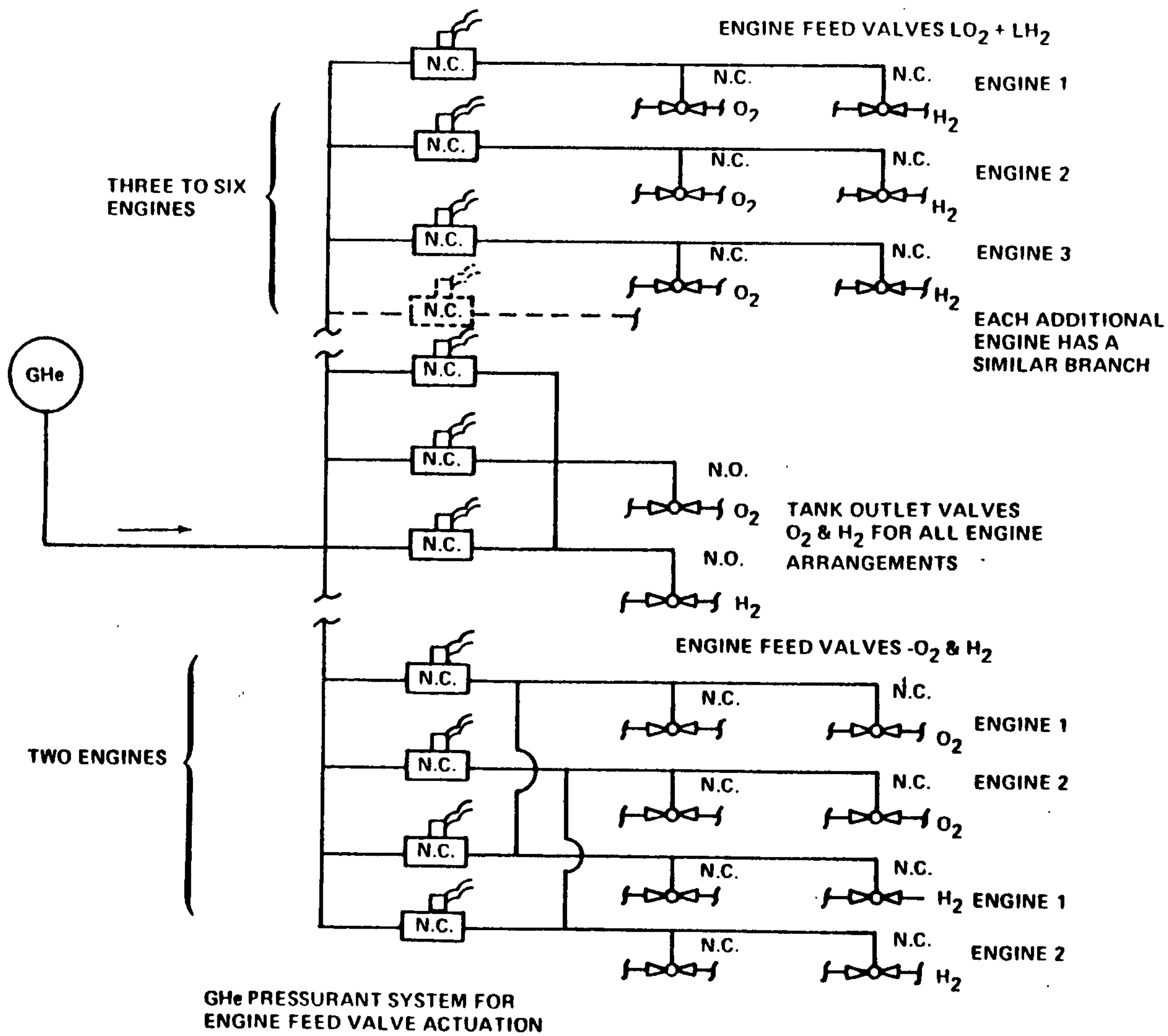
The pressurization system logic for multi engine configurations differs from that of a single engine configuration because, as the numbers of engines increases, the number of T stop valves required per engine branch to actuate the shut-off valves in the propellant feed line manifolds decreses. The reduction of the number of T stop valves is made possible by the simplification of the propellant feed line manifolds as number of engines grows. The simplification of the manifolds was discussed on Pages 183, 184 & 185 (Figures 3.2-16, -17, and -18).

The helium feed logic for flow to the propellant tanks in all multi engine configurations is identical to the tank branch subsystem described for the single engine configuration shown in the previous diagram. Figure 3.2-20 presents the logic for a pressurization system for an OTV over the spectrum of two to six engines.

Figure 3.2-20 **PROPULSION TRADE PRESSURIZATION LOGIC: MULTI-ENGINES**



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C-3

The remaining portion of the engine trade study deals with the interaction of the engine and the AOTV aft end. A vehicle design ground rule (for Phase I) that effects this area is that all reuseable AOTV components must be protected by the vehicle aeroshell during hypersonic passage through the atmosphere. This assumption will be explored in detail in Phase II of this study. This groundrule implies that the exit plane of the engines not extend beyond the end of the AOTV's aeroshell (Figure 3.2-21a). Note also (in Figure 3.2-21a) that an engine on the vehicle centerline forces a spherical (or elliptical) tank farther away from the end of the aeroshell than does an engine which is off the vehicle centerline. This implies that engines on the AOTV centerline will create longer AOTVs when minimum weight tanks are used.

The second sketch (b) on the facing page illustrates that a fixed location aeroshell will be burned off by operation of an off centerline, gimbaled engine with fixed nozzles, and, that damage will probably result from a gimbaled on centerline engine. For this reason, fixed nozzle gimbaled engines have been eliminated from the engine quantity trade study.

The third sketch (c) illustrates the potential for reduced length AOTV from large, deployable nozzle engines. The engines shown in (c) are the same size, and are in the same location as those in (a). The resulting AOTV is significantly shorter (as measured by the distance from the tank to the end of the aeroshell) in (c). However, this advantage disappears when engine size is small enough so that a fixed nozzle engine can fit into the gap between aeroshell and tank without increasing the length of the aeroshell beyond that required for other subsystems. This condition did occur for the 6-2500 lb thrust engines which were considered in the engine trade study.

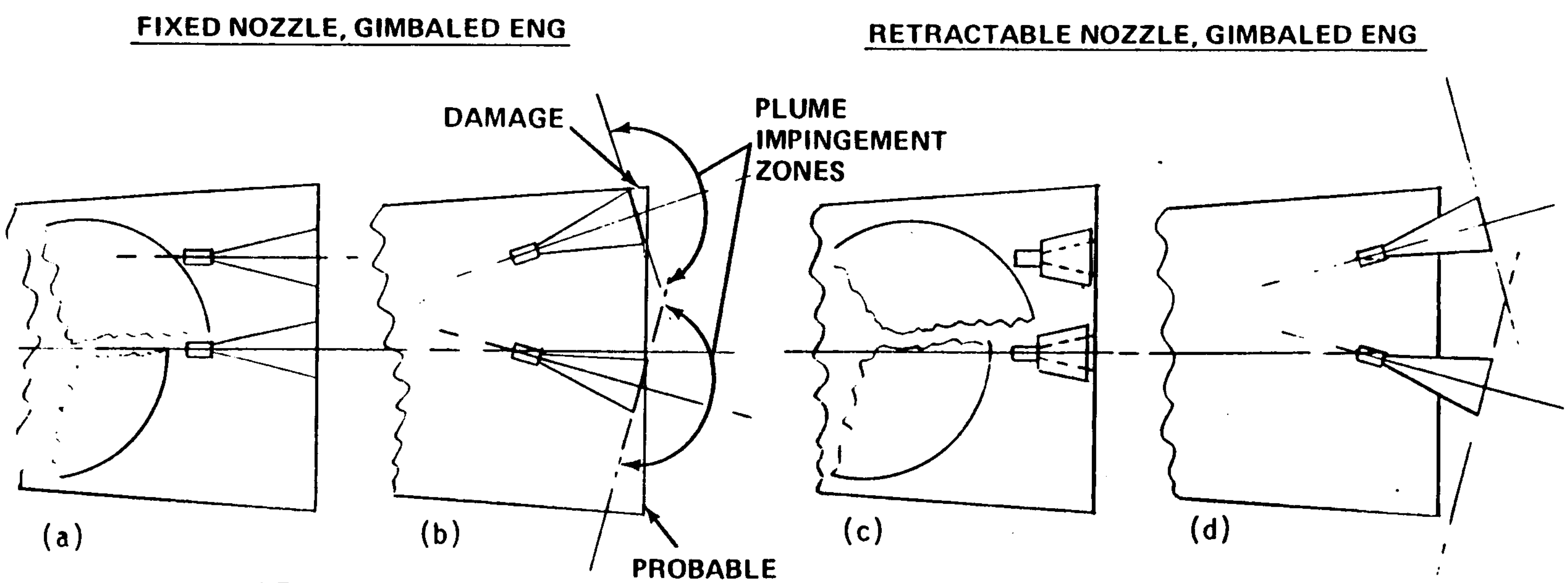
Figure 3.2-21d indicates that gimbaling engines with deployable nozzles does not threaten the AOTV with damage from the engine plume, which has been represented for this study as covering the entire region aft of the nozzle exit plane. Consequently, for this engine trade study, all gimbaled engines (for 1 through 5 engines) have deployable nozzles.

Figure 3.2-21 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

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The major assumptions and results of the trade study examining the number of main propulsion engines versus AOTV weight, for a man rated AOTV, are displayed in Figure 3.2-22. Study results are shown relative to a single engine AOTV (which does not completely meet the man rating redundancy requirements in the engine area).

One or two large engines produce the highest weight, and longest, AOTV. Three or four engines produce AOTVs which are close in weight (Δ = 26 lb), but the lower thrust engines in the 4 engine version permit an AOTV which is about 1 foot shorter. The AOTVs which result from 5 or 6 engines are about the same size (5 foot shorter than a single engine AOTV), but the 6 engine version weighs 100 lb less since it does not have deployable nozzles or gimbals and gimbal drive actuators.

The startling conclusion of this study, that 6 fixed nozzles, non-gimbaled engines will produce a lighter AOTV than 1 (or 2 or 3) engine(s) with the same total thrust, has two explanations. The first has to do with the propulsion system. When a fail/safe-fail/safe criteria is applied, there is not significant weight difference between 1, 2 or 3 engine systems and the propulsion weight penalty for 6 fixed engines is small (135 lb). The second explanation for the major engine trade conclusion is that a biconic AOTV pays a structural and TPS weight penalty for additional length. The extra 5′ of vehicle required by the single engine design adds nearly 400 lb of vehicle weight. This greatly exceeds the propulsion system weight advantage of a small number of engines. Consequently, 6 small, fixed nozzle, non-gimbaled 2500 lb thrust engines save 260 lb of AOTV weight when compared with a single, gimbaled, deployable nozzle 15,000 lb thrust engine. In addition, the 6 engine configuration produces an AOTV which is 5 feet shorter than the single engine version. This length reduction is a very significant advantage for ground based AOTVs.

Figure 3.2-22 **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

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Recommendations for engines on a man rated AOTV are displayed in Figure 3.2-23. During Phase I of this study some extremely attractive vehicle and mission combinations have been identified. For example, the H-1M vehicle can perform several of the design reference missions of the MOTV study (Ref. 7 - for example, Emergency Repair 1:ER-1) at substantially lower cost. For example, ER-1 required 3 shuttle launches to build up the MOTV. H-1M requires only 1 shuttle launch. Some of this savings of >$100M per ER-1 mission should be allocated to the advanced engines which help enable the H-1M mission. If the appropriate number is $\approx$ $30M, then 10 flights of H-1M will have offset the cost of developing new engines.

However, other possibly less expensive alternatives warrant investigation. Some of these are outlined in Figure 3.2-23.

Figure 3.2-23

ENGINE RECOMMENDATIONS FOR MAN RATED BI-CONIC AOTV

- AT THIS TIME, SIX FIXED, LOW THRUST ($\approx$2000 TO 3000 LB), ADVANCED EXPANDER, LOX-HYDROGEN ENGINES ARE STRONGLY PREFERRED
  - NEW ENGINE ROM DEVELOPMENT COSTS ARE $\approx$ \$300M
  - IS PERFORMANCE GAIN (MAN TO GEO ON ONE 65k STS FLT) WORTH THE EXPENSE?

- SOME ALTERNATIVES TO NEW LOX-HYDROGEN ENGINE DEVELOPMENT THAT SEEM WORTHY OF FUTURE STUDY:
  - THREE RL 10A-3-3A ENGINES ON H-1M TYPE VEHICLE INCORPORATE "IMPROVED TECHNOLOGY" SUBSYSTEM WEIGHTS
    - EXPLORE EFFECTS OF ENHANCED STS CARGO CAPACITY (65k, 75k, 100k) ON THIS VEHICLE
  - ALTERNATIVE LOX-HYDROCARBON FUELS (MMH, PROPANE, METHANE, KEROSENE)
    - EXPLORE AOTV SYSTEM IMPLICATIONS
    - INCLUDE EFFECTS OF VARIABLE STS LAUNCH CAPACITY.
  - STORABLE PROPULSION (MMH, N_2H_4) AOTV IN CONJUNCTION WITH HEAVY LIFT LAUNCH CAPABILITY (100k STS OR SDV)

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3.3 Thermal Protection Subsystem

The baseline state-of-the-art thermal protection subsystem is of the Reusable Surface Insulation type (RSI and FRCI) bonded to a Strain Isolation Pad (SIP) where required with a silicone based adhesive. The baseline TPS thickness (weight) requirements have been determined for a design criteria of 1) initial temperature of 100^{o} and 2) maximum bond line temperature of 350^{o}. Thickness requirements were determined, Figure 3.3-1 through use of the Reaction Kinetics Ablation Program (REKAP) code (37). Thermal conductivity as a function of both temperature and pressure was employed. Thermal soak-out of the AOTV TPS occurs during the lift out of the atmosphere at near vacuum pressures in contrast to the Space Shuttle Orbiter where soak-out occurs at near one atmosphere pressure.

The total weight of the thermal protection subsystem can be reduced significantly through three separate means: 1) thermal conditioning of the TPS prior to entry by cold-soaking it to -100^{o}, 2) reduction in the weight of the protective coating, 3) permitting the structure/bond line to soak out to 600^{o}.

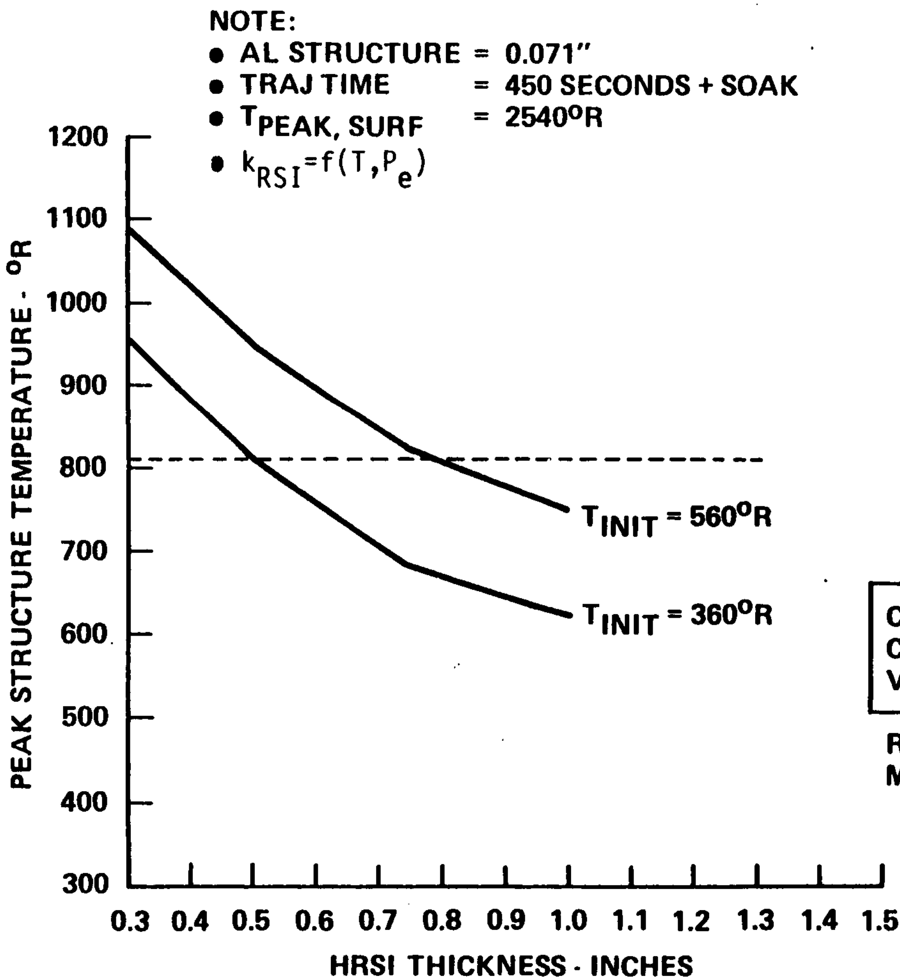
It has been estimated that thermal conditioning prior to entry could reduce the RSI weight requirement by as much as 23%, reducing the protective coating by 50% could result in a reduction of the total TPS weight by 9% and increasing to 600^{o} the allowable maximum structure bond line temperature could result in a TPS weight reduction of 37%.

Estimates have been made of the maximum surface temperatures expected on the nose and aft frustum areas of this class of vehicles, Figure 3.3-2 and 3. The convective heating rates are for equilibrium flow and a fully catalytic surface. The non catalytic nature of the TPS coating and the non-equilibrium nature of the flow are expected to produce maximum temperatures $100-200^{o}$ lower than those quoted for the aft frustum and up to 400^{o} lower for the nose. Estimates of eqilibrium hot gas radiation, using Pages results (13) demonstrated it to be a second order effect, so it has been neglected in this study.

Current state-of-the-art, normal growth and accelerated growth increases in peak allowable surface temperatures have been specified by Goldstein and Curry (34 and 35), Figure 3.3-4. These have been compared to the predicted maximum surface temperatures for these AOTVs.

It can be observed that most of the aft frustum maximum surface temperatures fall within the current capability of FRCI,

Figure 3.3-1 AOTV WINDWARD MERIDIAN RSI THICKNESS REQUIREMENTS



PEAK STRUCTURE TEMPERATURE °F	350		600
$T_{INITIAL}$ (°F)	RSI + SIP WT (PSF)		
	WITH COATING	WITHOUT COATING	WITH COATING
+100	0.94	0.78	0.60
−100	0.73	0.56	

COLD SOAK, HIGHER STRUCTURE ALLOWABLE, AND COATING WEIGHT REDUCTION/ELIMINATION CAN PROVIDE SIGNIFICANT WEIGHT PAYOFFS.

RECOMMEND: AOTV ADVOCATE MATERIALS DEVELOPMENT AND TPS CONCEPT EVALUATIONS TESTS.

Figure 3.3-2. Effect of Mission on Maximum Aft Frustum Surface Temperature

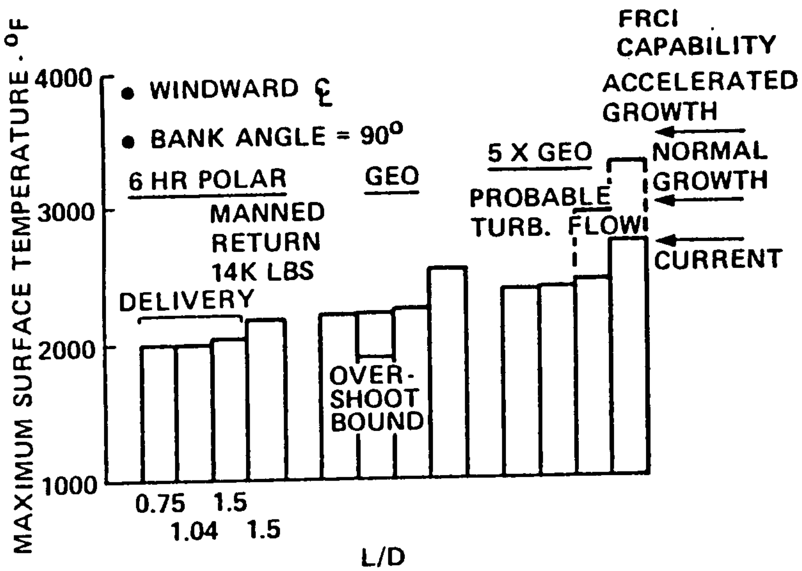


Figure 20. Effect of Mission on Maximum Aft Frustum Surface Temperature

Figure 3.3-3. Effect of Mission on Maximum Nose Surface Temperature

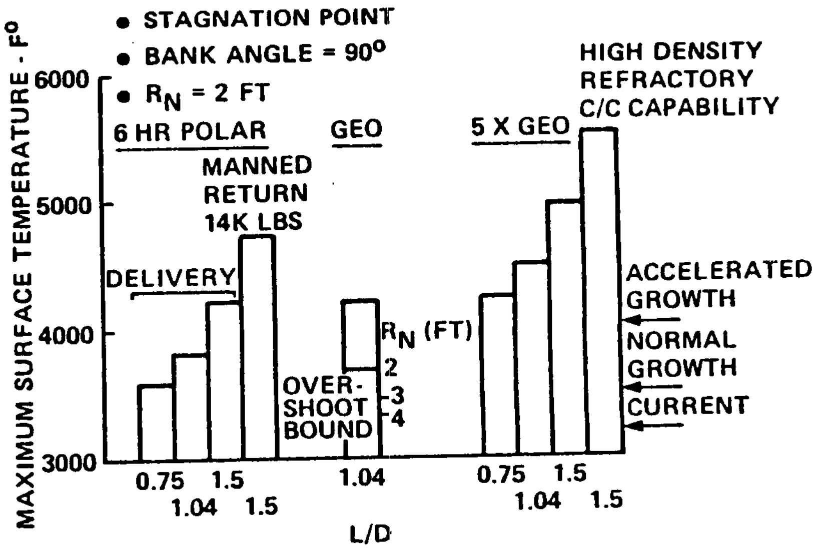


Figure 21. Effect of Mission on Maximum Nose Surface Temperature

AOTV

Figure 3.3-4 TPS Technology Projections

MATERIALS TEMPERATURE CAPABILITY (°F) - 100 REUSES

1990 TECHNOLOGY (1995 IOC)

MATERIAL	CURRENT	NORMAL GROWTH	ACCELERATED GROWTH
HIGH DENSITY REFRACTORY (CARBON/CARBON)	3200 (1) 3 2800 4	3500 3800-4000 5	4000
RIGID SURFACE INSULATION (FRCI)	2700	3000	3500
FLEXIBLE SURFACE INSULATION (AFRSI)	1500 - 1800	1800 - 2500 (2)	2500 - 3000

(1) CURENTLY LIMITED BY COATING

(2) ADDITIONAL DATA WILL BE PROVIDED BY W.T. TESTS TO BE CONDUCTED AT ARC IN FEBRUARY '84

(3) SINGLE USE, PER DON CURRY

(4) 100 USES, PER DON CURRY

(5) PER DON CURRY, NASA JSC

Approved by H. Goldstein, NASA Ames
17 JAN, 1983

with the exception of L/D = 1.5 vehicles returning from 5 x GEO, where the 2500° temperature capability is exceeded due to probable turbulent flow if single pass capture is used. On the nose, due to the relatively small nose radius, maximum surface temperatures are expected to exceed even the 4000° accelerated growth TPS capability for most of the missions involving one pass capture.

Four different approaches to rescue the nose temperature have been investigated and include: 1) alternate steering laws, 2) higher angles of attack, 3) multiple rather than single atmopsheric passes, and 4) a larger nose radius. Evaluations to date have considered entry at (L/D) maximum only for the purpose of producing maximum aerodynamic plane change. The convective heat flux reductions possible by opeorating toward the one pass capture bound rather than going for maximum plane change, are illustrated in Figure 3.2-2.

Significant reductions in peak nose surface tempeorature can also be obtained by flying at an angle of attack greater than that for (L/D) max. and accepting less plane change. By flying at this larger angle of attack, larger lift and drag coefficients are obtained and thus the vehicle can operate at a higher minimum altitude while reducing the peak heat transfer rate experienced. The net result is a decrease in maximum nose temperature and a small increase in maximum windward surface temperature due to the higher angle of attack.

It has been demonstrated in the AMOSS Studies (6) that use of multi pass rather than single pass capture significantly reduces the maximum heat transfer rates experienced and hence maximum surface temperatures, while the total plane change obtained aerodynamically remains the same as for a single pass. These results can be combined to determine those combinations of steering law and number of passes to limit the nose cap to temperatures acceptable for reuse. These results are summarized for return from 5X GEO and 6 hr polar in Figures 3.3-5 and 6 where it can be seen that for a given allowable temperature limit, the magnitude of aerodynamic plane change achievable (and hence, delivered payload) increases with number of atmospheric passes permitted. An L/D = 1.5 AOTV, flying at a 15° angle of attack, will require 4 atmospheric passes to achieve nearly all of its maximum aerodyanamic plane change (~26°) capability if nose temperature is limited to 4000°.

The effect of increasing the nose radius has also been investigated. For a given steering law and angle of attack, the required nose radius to reduce the maximum nose temperatures to those required for reusable passive TPS for a one pass capture have been estimated. Increasing the nose radius also reduces the vehicle L/D due to increased drag, thus allowing the vehicle to operate a a higher altitude, again reducing the peak heat transfer rate experienced, but reducing the magnitude of the plane change obtained. For example, if two passes are required

Figure 3.3-5. Limiting the Nose Cap to Acceptable Temperatures for Reuse

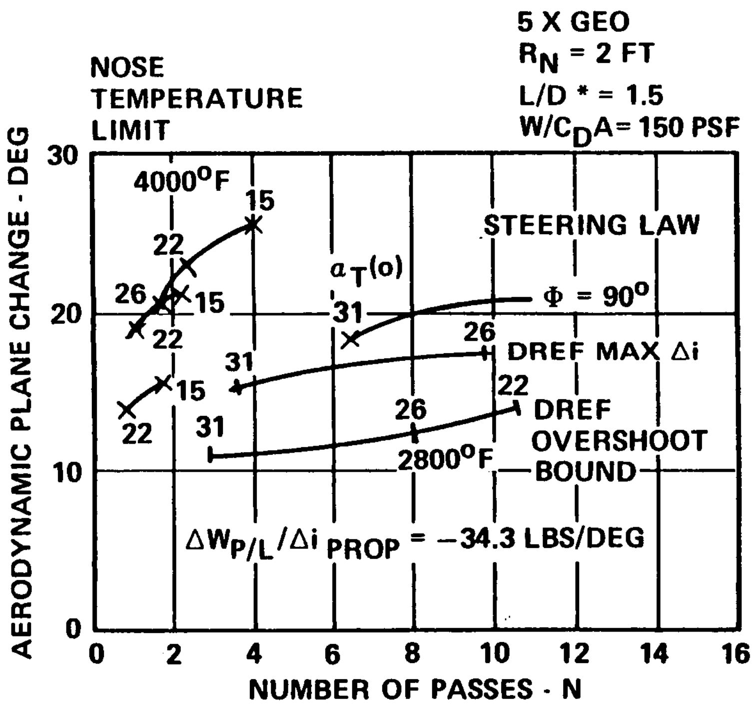
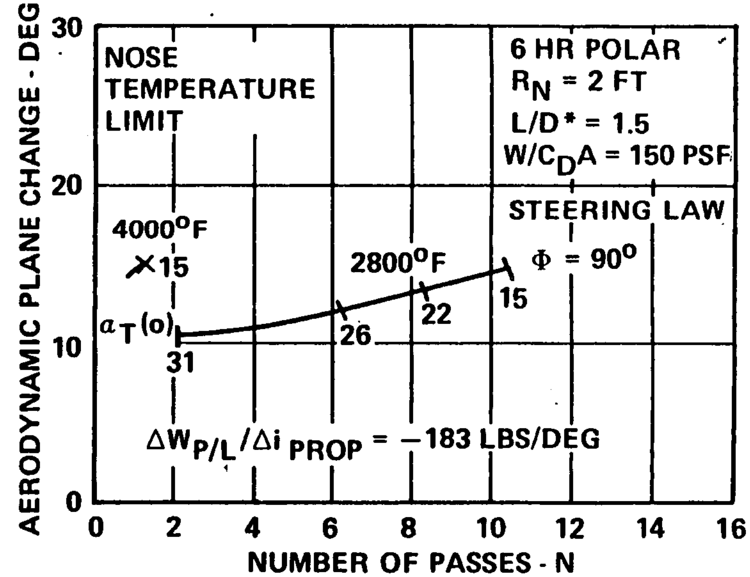


Figure 3.3-6. Limiting the Nose Cap to Acceptable Temperatures for Reuse



in order to meet temperature limits for the reference vehicle with a nose radius of 2 feet, then by doubling the nose radius to about 4 ft., the number of passes may be reduced to one.

If the combination of aerodyanamic plane change and number of atmopsheric passes required is unacceptable, then alternate approaches of either transpiration cooling, heat pipe, or use of a refurbishable thermal protection material will be required. Transpiration cooling using either a liquid or gas injectant has been flight qualified on small nose radius (<2 inches) vehicles for some time. Development effort is currently underway on much larger systems. Heat pipe designs for noses and leading edges have been developed and ground test in previous studies (38, 39, 40).

For a ground based AOTV, an alternate approach to a transpiration or heat pipe cooled nose or a multipass reusable RCC nose is the use of a refurbishable state-of-the-art low density ablator. The potential for downstream TPS surface contamination from ablation products would have to be considered.

3.4 Structure Subsystem

The AOTV baseline state-of-the-art structural subsystem consists of the following elements: 1) a graphite epoxy structure, 2) aluminum cryogenic fuel and oxidizer tanks, and 3) split body flaps for use in center-of-mass offset management.

Several recent trade studies and surveys have been conducted to determine the value of structural material advances and design/property improvements for advanced space vehicles. An organized activity is underway, spearheaded by the "AIAA Composite Structure Subcommittee" composed of members from NASA, Government agencies, and manufacturing firms. Their recent survey revealed composite structure design allowables were obtained from a wide variety of test methods and many different failure criteria were employed. The committee is striving for:

1) Unified testing to determine allowables - consensus is to use uniaxial data and analysis to determine composite properties.

2) Unified failure criteria for design of composite structure.

We recommend that AOTV beocme an advocate for this organized activity. It is projected that a 10 to 30% weight reduction of AOTV structural weight is possible with an improved definition of design properties.

Numerous advanced structural materials are under investigation/development and include graphite composites, metal matrix composites - and doped and undoped aluminum lithium alloys. It is projected that additional structural weight reductions are possible by use of metal matrix materials reinforced with graphite whiskers (14). The higher temperature limits (e.g., $600^{o}F$-$1000^{o}F$) of these materials will provide additional reduction in TPS weight.

3.5 Avionics Subsystem

The reference state-of-the-art avionics subsystem weight of 600 pounds is driven primarily by degree of autonomy and redundancy. The future OTV technology study, (15), projected 50% weight reduction due to normal growth technology employing laser gyros and data bus. Our evaluation of an expandable fly-by-wire delivery stage design suggests up to 70% weight reduction is possible if avionics autonomy could be sacrificed.

Numerous hardware and software advanced development activities are underway, e.g., spacecraft flight computers are currently under development that can process 450,000 instructions per second with up to one million words memory (16 bit).

Hence, significant reductions in avionics subsystem weight appear possible.

3.6 Flight Control Subsystem

Atmospheric roll control using a hybrid combination of Reaction Control System (RCS) and aero surfaces for trim control or additional control authority is expected to provide the most attractive Flight Control actuation approach. Attitude sensing and acceleration information will be provided by inertial sensors which may also serve as instrumentation sensors or in some cases, provide data during other phases of the mission.

These conclusions are based on previous studies (36) which evaluated the mid L/D aeroassist vehicle response to radial and axial c.g. offsets. It was found that for a general vehicle split windward flap subsystem weight was relatively insensitive to c.g. offset magnitude; whereas the mass of RCS propellant consumed was directly proportional. Six-degree-of-freedom simulations in previous studies (36) also demonstrated the use of the RCS to quickly damp angle of attack and yaw angle oscillations. It is expected that the RCS will provide additional motion damping during atmospheric exit, where typically undamping tends to occur due to decreasing dynamic pressure.

The information from the inertial measuring sensors will provide inputs for the flight path/attitude algorithm which will provide vehicle position, velocity, attitude and attitude rate which are used for atmospheric flight path control and analytic drag guidance.

Flight path control is provided by the attitude control system through the command of either gas jets or aerodynamic control surfaces for roll modulation of the lift vector. The RCS is required for exoatmospheric attitude contol and is therefore part of the AOTV, but attempts to adapt for atmospheric roll control, using two level thrustor or pulse width modulation have resulted in heavier systems without comparable payoff in accuracies. Aerosurfaces can provide the roll control for additional weight penalty, but also augment pitch and provide damping.

Additional benefits resulting from the combined reaction and aero torque approach include: (1) the potential to design with some margin for redundant contingency control modes, (2) potential to extend control authority to the pitch and yaw axes to provide damping, or if found necessary, full 3-axis control and (3) potential for tighter attitude controls during atmospheric operation and entry/exit transition regimes.

The roll control subsystem provides required bank-reverse-bank roll command logic and torques to modulate lift in the plane to maintain Dref, while minimizing the out-of-plane components of lift error, to minimize exit state errors.

Bank angle commands provide the desired in-plane lift with modulation for the in-plane magnitude resulting in out-of-plane components which are nulled by commanding roll attitude sweeps of the fixed trim in both plus and minus directions.

Figure 3.6-1 shows a block diagram of a possible hybrid flight control using combined RCS and Split Windward Flap (SWF). The in-plane lift is modulated for altitude adjustments to maintain constant Dref. Primary lift magnitude is derived from basic aeroconfiguration with a percentage, or Δ-lift, lift control authority available from the flaps.

Roll control must cause the vehicle to follow commands from the guidance law in a stable manner. Studies have shown the strong sensitivity of control sizing to upsetting roll disturbance torques which arise primarily from mass property asymetries resulting from loads, equipment or manuacturing tolerances. Out-of-plane angle of attack can result from asymmetries which contribute to error build-ups in out-of-plane lift.

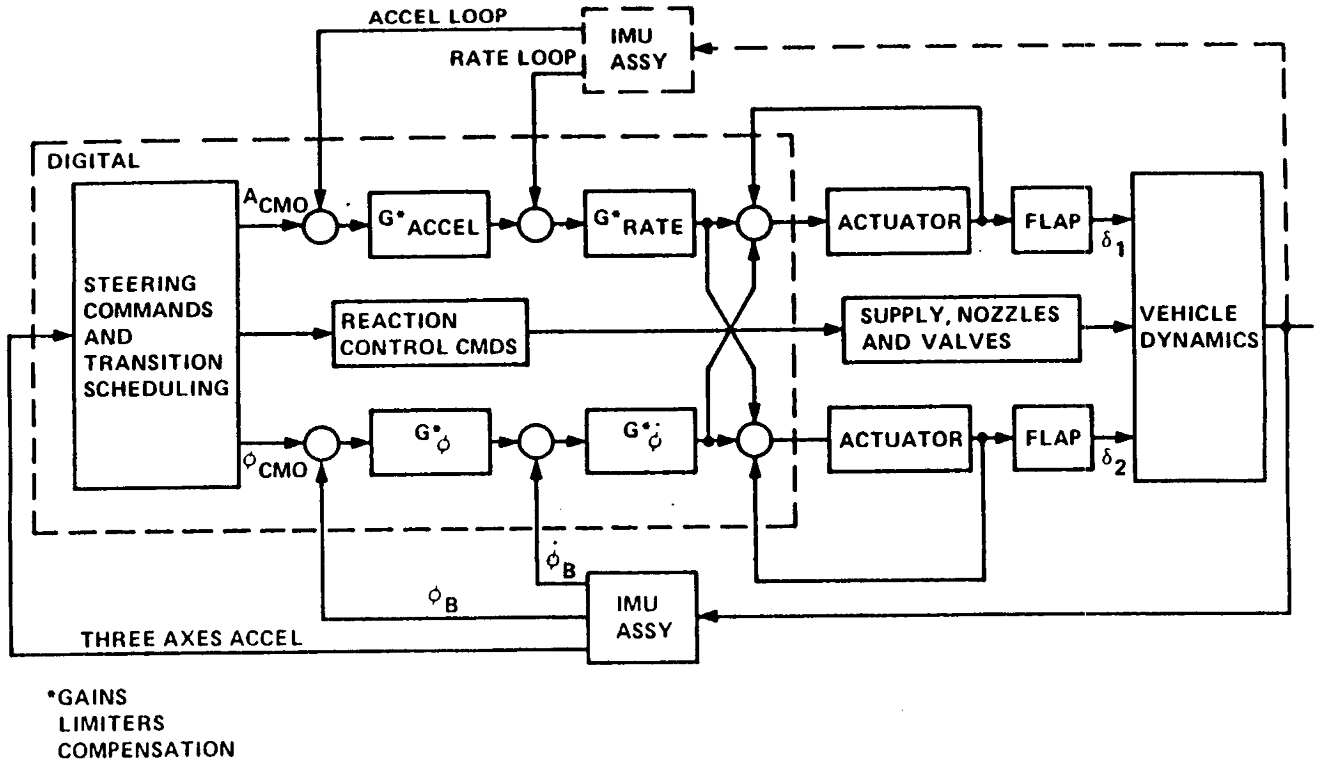


Figure 3.6-1. Hybrid RCS/SWF Control

In addition to cg offsets, variations in the aerodynamic center-of-pressure (CP), from design, can affect the aero performance and resulting control authority. Combination of CG and CP variations and the resulting static margin changes also impact control authority.

Advantages of a hybrid roll control approach are: 1) improved accuracy in attitude control 2) blending during approach or entry dynamics 3) blending during exit or departure dynamics, and 4) augmented lift control authority.

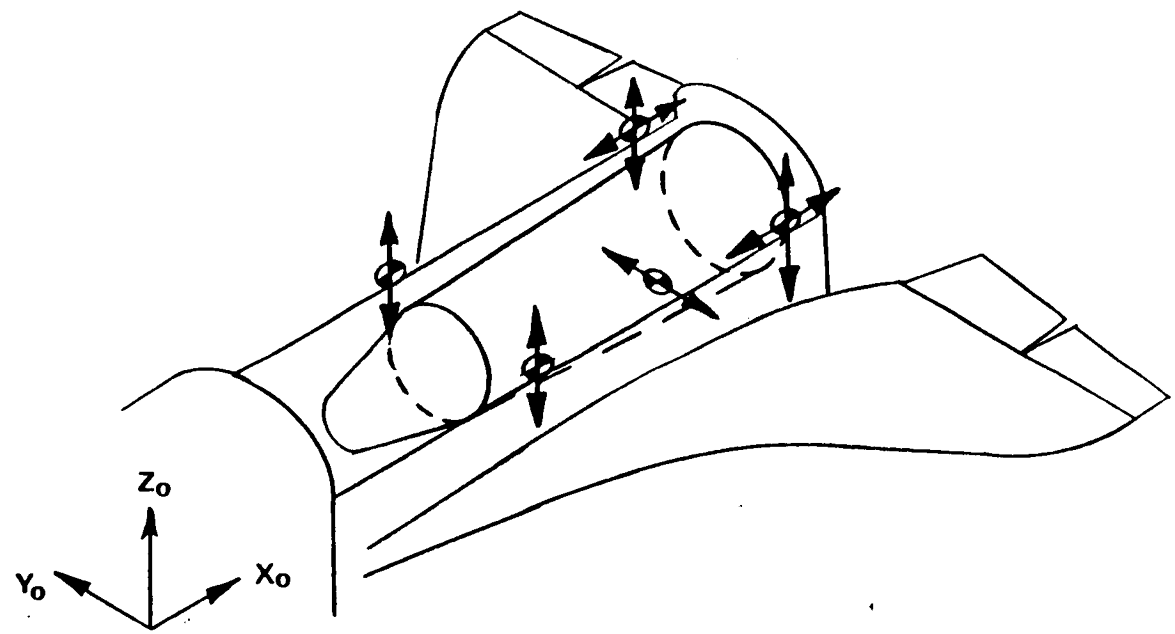
Transition scheduling from full RCS roll control in the rarefied regime to predominant SWF atmospheric roll control, i.e., blending, will be controlled in the digital computer.

3.7 Aerospace Support Equipment

The preferred general arrangement for transmitting loads between the Orbiter payload bay and AOTV is illustrated in Figure 3.7-1. Vertical (±Zo) loads are removed at 4 locations on the Orbiter longerons. Lateral (+Yo) loads are removed at an orbiter keel fitting at the bottom of the bay. Fore (-Xo) and After (+Xo) loads are removed at the same 2 aft longeron locations that carry vertical loads.

AOTV deployment in space is a two phase operation. First, the two +Yo longeron fittings and the keel fitting release the AOTV. The AOTV is then rotated by the orbiter RMS about the xo axis on the remaining -Yo longeron fittings until it is outside the payload bay (about 90° of rotation). In this position, the AOTV can be checked out. For manned AOTVs like Figure 2.4-21, half the aeroshell can be opened to expose the crew capsule. The crew capsule can be rotated out of the AOTV to permit attachment of a "transfer tunnel" from the orbiter cabin. This allows shirtsleeve transfer of the crew to the AOTV. The second phase of deployment occurs after the AOTV is closed and prepared for flight. At that time, the two -Yo longeron attachments release the AOTV and springs at the attachments propel the AOTV away from the Orbiter. Another mode of deployment could use the orbiter RMS for a separation impulse.

Figure 3.7-1 **AOTV—ORBITER ATTACHMENT**



- 5-POINT PAYLOAD RETENTION SYSTEM (INDETERMINATE)
- X_o & Y_o LOADS REMOVED NEAR LO_2 TANK
- RMS POWERS AOTV REMOVAL FROM ORBITER
- UMBILLICALS ENTER NEAR KEEL FITTING

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A ROM preliminary design of the Airborne Support Equipment (ASE) structure, which attachs the AOTV to the orbiter, indicates that a very lightweight structure is possible, Figure 3.7-2. The baseline structural element is a hollow steel tube with a diameter/thickness ratio of 20. Applying the utlimate load factors shown on the facing page to a 61,300 lb AOTV (H-1M when fully loaded), and using modest strength steel (FTu = 180,000 psi),the four longeron tubes and one keel tube weigh about 110 lb. Adding weight for mechanisms to permit deployment motions and control, the total structural system weight for holding (and releasing) a large AOTV within the Orbiter is 270 pounds.

Figure 3.7-2 SOME AOTV ASE WEIGHTS - STRUCTURAL

	WEIGHT ALLOWANCE
AFT LONGERON STRUCTURE:	100 LB
• ULTIMATE LOAD FACTORS	
– $N_x = 6.75$ g	
– $N_z = \pm 4.5$ g	
• MAX LOADS: 211,500 LB & 2.1×10^6 IN. LB	
• (2) 10" LONG STEEL TUBES, 6-5/8" DIA, t = .33"	
– F_{TU} = 180,000 PSI	
– WEIGHT (FOR 2) = 37 LB	
• WEIGHT ALLOWANCE FOR MECHANISMS, MOTORS, EMERGENCY RELEASE, ETC = 63 LB	
FORWARD LONGERON STRUCTURE:	120 LB
• (2) 40" LONG STEEL TUBES, 5.7" DIA, t = .28"	
– WEIGHT (FOR 2) = 54 LB	
• WEIGHT ALLOWANCE FOR MECHANISMS = 66 LB	
KEEL STRUCTURE	50 LB
	Σ = 270 LB

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A review of orbiter longeron bridge fittings (from JSC07700) shows 14 different sizes, with an average weight of 141 pounds. Since this AOTV installation has the aft fittings near the maximum density region of the AOTV (the LOX tank), the aft fittings are highly loaded. Since the aft are highly loaded, the forward longeron fittings are lightly loaded. Since the number of highly loaded fittings is equal to the number of lightly loaded fittings, an average fitting weight can be assumed for all. Four average longeron bridge fittings weigh 564 pounds.

The 12 different keel fittings of JSC 07700 have an average weight of 201 pounds, and a standard deviation of 53 pounds. Since the keel fitting is very highly loaded in this AOTV application (H-1M), a keel fitting weight of 254 pounds was chosen. This is very close to the heaviest keel fitting in the list.

After adjusting the total of all longeron bridge and keel fittings weights for the 675 lb allowed all payloads (at no charge), the net bridge fitting weight penalty charged to the ASE for support of a 61,000 lb vehicle is 143 pounds, Figure 3.7-3.

Figure 3.7-3 **SOME AOTV ASE WEIGHTS - ORBITER FITTINGS**

	WEIGHT ALLOWANCE
LONGERON BRIDGE FITTINGS:	
• 14 DIFFERENT FTGS	
– AVERAGE WT = 141 LB	
– 1 σ VARIATION = 24 LB	
• 4 FTGS REQ'D FOR AOTV	
– 2 HIGHLY LOADED	
– 2 LIGHTLY LOADED	
• CHOOSE 4 AT MEAN VALUE	564 LB
KEEL BRIDGE FITTINGS:	
• 12 DIFFERENT FTGS	
– AVERAGE WT = 201 LB	
– 1 σ VARIATION = 53 LB	
• 1 HIGHLY LOADED FTG REQ'D	
– CHOOSE "MEAN + 1 σ"	254 LB
Σ =	818 LB
FREE BRIDGE WEIGHT ALLOWANCE =	675 LB
NET BRIDGE FTG =	143 LB

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A weight estimate was prepared for the hardware elements which are necessary to dump 45,000 lb of LOX/LH propellants, from the AOTV to discharge ports on the orbiter, within 5 minutes, Figure 3.7-4. The major weight advantage of this system, when compared with currently planned, systems, is the single large (5' diameter) tank for storage of high pressure helium gas. The current state-of-the-art filament wound tank saves many hundreds of pounds over multiple small bottles.

Figure 3.7-4

AOTV ASE WEIGHTS - ABORT PROPELLANT DUMP

DESCRIPTION	WEIGHT (LB)		SUBSYSTEM WEIGHT (LB)
3000 PSI GHe TANK (STAINLESS STEEL WOUND WITH KEVLAR FILAMENT)		893	
GHe TANK – HDW + INSULATION		48	
GHe TANK – SUPPORT IN PAYLOAD BAY		115	
	Σ TANK & SUPPORTS =	1056	1056
COUPLINGS		5	
VALVES		38	
INSTRUMENTATION		12	
ELECTRICAL WIRING & CONNECTORS		44	
SAFETY DEVICES		16	
REGULATORS		14	
FILTER		2	
INSULATION		36	
	Σ COMPONENTS =	167	167
LINES + FITTINGS + MISC – O_2 BRANCH		74	
LINES + FITTINGS + MISC – H_2 BRANCH		86	
LINES + FITTINGS + MISC – He MANIFOLD		27	
	Σ LINES & FITTINGS =	187	187
		SYSTEM TOTAL =	1410 LB

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Subsystem elements of the Airborne Support Equipment for two programs are compared in Figure 3.7-5. Centaur G' (a preliminary definition from the Centaur Project Office) and a large biconic AOTV (typically, H-1M).

The first four lines of the table have been discussed previously (for AOTV). Residuals are virtually the same for both vehicles. A large weight difference occurs in Avionics. Centaur G' has extensive instrumentation to monitor the pressure stabilized bulkhead, as well as a number of batteries. The AOTV is intended to be nearly autonomous, with a significant on-board health monitoring system. Consequently, some of the Centaur ASE Avionics are on board the AOTV as part of the vehicles Avionics weight. The 100 lb allocated to AOTV ASE Avionics was estimated sa satisfactory for an autonomous vehicle. the last entry in the AOTV table is for an orbiter stored collapsable tunnel to permit shirtsleeve transfer of personnel from the orbiter cabin to the crew capsule of a manned AOTV.

The substantially lighter weight of the AOTV ASE (-6500 lb) allows more propellant to be loaded in ground based AOTVs. This allows higher payload weights delivered to GEO.

Figure 3.7-5 **PRELIMINARY ASE WEIGHT SUMMARY**

	ASE WEIGHT (POUNDS) CENTAUR G[1] (GALILEO MISSION)	AOTV
STRUCTURE	2045	270
NET BRIDGE FITTINGS	883	143
FLUID SYSTEMS (ABORT DUMP)	1937	1410
FLUID SYSTEMS MISSION KIT	268	268
RESIDUALS		
– PROPELLANTS	43	43
– HELIUM	117	120
AVIONICS	1474	100
SMCH CABLE (+ TRAYS, BULKHEAD WIRING)	750	200
PAYLOAD RECORDER & TIMING BUFFER	63	63
AFT FLT DECK WIRING & SWITCH PANELS	46	46
VEHICLE/MISSION UNIQUE	1751	240*
	Σ = 9377 LB	2900 LB

*PERSONNEL TRANSFER TUNNEL & AIR

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3.8 Electrical Power Subsystem

The AOTV baseline state-of-the-art electrical power subsystem consists of the following elements: 1) two 3.5 KW STS orbiter fuel cells with reactants, 2) back-up batteries, 3) tanks and plumbing, 4) power conditioning equipment, electrical harness and equipment supports.

The reference state-of-the-art electrical power subsystem weight is 600 pounds. It is projected that use of non-metallics like graphite in the power section to replace magnesium and nickel and continuing improvement in battery technology and other parts of the subsystem can result in 20 to 38% weight saving. Currently the NASA sponsored OAST "NASA Regenerative Fuel Cell Program" at JSC and Lewis has this task in their program, but it is unfunded. It is recommended that AOTV become an advocate for task funding.

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