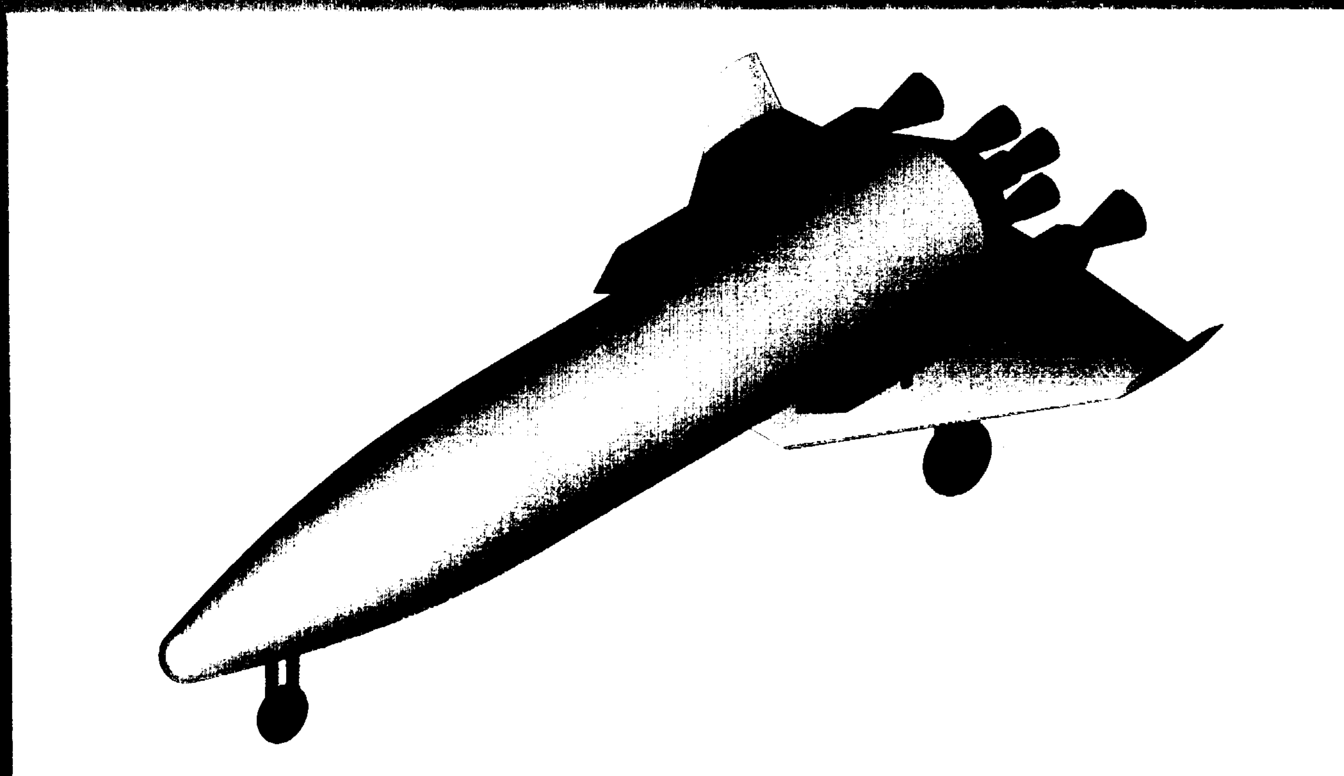
HRST
Rocket/RBCC Options Part 2
A Comparative Study



Space America
October 1997

HRST Rocket/RBCC Options Part 2

This study extension was performed under purchase order # H-28681D from the Marshall Space Flight Center to Space America, Inc.

The HRST study is managed by John Mankins of NASA Headquarters. The principal investigator for this part of the HRST study was Gordon Woodcock. Concept illustrations were prepared by Don Parker. The study Contracting Officer's Technical Representative was Joe Howell of MSFC.

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Relationship to Current HRST Activities

This study was an extension of an earlier study by the same principal investigator, which undertook a comparative study of propulsion options maintaining traceability of weights and performance estimates so that apparent advantages of one concept over another can be traced to their sources.

Relationship to Concurrent HRST Activities

- This study was intended to compare propulsion options on a consistent, traceable basis, beginning with VTOHL all-rocket which benefitted from much definition effort compared to other options
- Comparisons internal to the study difficult to relate to external concepts due to differences in weights conservatism
- Interest in staged options (TSTO) developed late in HRST activity; desired to add at least one staged option
- Accommodated by small follow-on effort reported herein

Study Background

The prior study phase considered four options: (1) Vertical takeoff horizontal landing rocket (VTOHL), single-stage-to-orbit (SSTO), a version of the HRST reference vehicle which in turn was derived from the NASA Access to Space Study recommended SSTO vehicle; (2) An assisted horizontal takeoff, horizontal landing vehicle (HTOHL) SSTO which was presumed to use a mag-lev ground accelerator (no work was done in this study on the ground accelerator itself), to provide takeoff speed 200 m/sec; (3) A modified version of (2) including a kerosene ramjet for acceleration assistance from about Mach 2 to Mach 6; and (4) A modified version of (2) including a LOX-kerosene rocket-based combined cycle (RBCC) engine for acceleration assistance from takeoff to Mach 6. The RBCC was designed to be a rocket ejector device with fixed geometry, without fan augmentation. The rocket switched to ramjet operation about Mach 2, when ram air pressure provides enough cycle thrust by ramjet operation without LOX flow.

Of these, the HTOHL rocket and the HTOHL RBCC were promising. The first option was used as a reference, and the ramjet option did not have attractive performance capability.

For consistency with other HRST data, it was desired to calculate sensitivity of the rocket HTOHL to weights growth margin and payload. Additional configuration and pictorial information was desired. Also, the approach to weights margins was different; the present study built the margins into the subsystems estimates while other HRST studies carried explicit margin. Therefore, a task was needed to evaluate relative weights conservatism and margins between the HTOHL of this study and the NASA HRST reference vehicle. Finally, examination of staged options was desired, to compare staging with the pseudo-staging of a ground accelerator.

These additional needs formed the basis for a statement of work for the current tasks reported herein.

Study Background

What led to the current task?

- Prior study identified two promising options
  - Assisted horizontal takeoff all-rocket (assisted HTOHL)
  - Assisted horizontal takeoff kerosene rocket-based combined cycle (assisted RBCC)
- Open questions remained
  - Sensitivities
  - Configuration layout and geometry
    - The prior study was analysis-oriented
  - Compare with related configurations from other HRST studies
    - Differences in weights conservatism greater than concept differences
  - Staged options as an alternative to takeoff assist
    - Can we preserve benefits of SSTO while reducing sensitivity?

Current Study Objectives

The current study objectives stated on the facing page responded to the stated needs. These objectives were satisfied and are reported in this briefing.

Two new two-stage-to-orbit (TSTO) concepts were introduced for the current study. These were intended to keep the operations simple but gain some of the performance advantages of staging. One is termed "hi-stage" for high staging velocity. The main stage launches and enters a once-around-Earth trajectory, returning to the launch site for landing. The second stage separates at main stage cutoff, inserts into the mission orbit, delivers the payload and returns to Earth for a runway landing. The configuration is tandem, with the second stage in front of the main stage. Both are aerodynamic configurations.

The second new concept, termed "lo-stage", has low staging velocity. It consists of a main stage which goes to orbit and performs the mission, returning to Earth for runway landing, and a pair of boosters which provide takeoff assist. The boosters are low technology pressure-fed or hybrids, parachute/sea recovered a few miles downrange, provide enough thrust for vertical takeoff, and separate between Mach 1.2 and 1.4. During booster burn, the vehicle pitches over to a near-horizontal flight path at about 5 km altitude. The main stage has thrust-to-weight typical of an HTOHL vehicle and flies a horizontal-takeoff-like trajectory after staging.

Current Study Objectives

- Provide configuration definitions and illustrations for assisted HTOHL and assisted RBCC;
- Develop a rationalization approach and compare these concepts with the HRST reference;
- Analyze TSTO configurations which try to maintain SSTO benefits while reducing inert weight sensitivity.

Agenda

This report is organized according to the agenda given here.

Agenda

- Sizing Calculations: Rocket and RBCC Options
- Configuration Descriptions
- Sensitivity to Payload and Inert Weight Margin
- Comparison Approach
- Comparison Results
- TSTO Approaches
- TSTO Analysis
- TSTO Results
- Comparisons

Rocket Option Delta V and Sizing

The facing page summarizes delta V estimates for the rocket options. These are based on the spread-sheet trajectory described in the report of the previous study phase, and on estimates of differences between configurations.

The ideal delta V estimate for the HTOHL is typical for this class of vehicle, assuming the delta V savings for horizontal launch are only the velocity contribution from the ground accelerator. Trajectory losses for a horizontal takeoff vehicle are sensitive to drag. The configurations analyzed here were designed for minimum inert weight and drag minimization was not done. Recent developments in leading edge materials would allow more streamlined leading edges and a smaller nose cap radius, leading to some drag reduction.

Estimates made for the present study, adjusted for vertical takeoff, compared well with those made for the Lockheed-Martin Venture Star vehicle.

A crude glide integral was performed to estimate the insertion delta V required for a once-around trajectory for the "hi-stage". It was estimated that the delta V requirement is about 200 m/sec less than required to achieve low circular orbit. In addition, about 50 m/sec is saved because the second stage provides the delta V to enter the transfer orbit. The main stage also does not need a payload bay or orbit maneuver system.

The boosters for the "lo-stage" system were estimated to reduce ascent delta V for the main stage by about 600 m/sec based on examination of the horizontal takeoff trajectory. It was assumed the ideal delta V required of the boosters is 700 m/sec.

Mass ratios calculated here were used for iterative spread sheet weight estimates, adjusting propellant load until the required mass ratio is established by the ratio (takeoff weight)/(injected weight).

Rocket Option Delta V and Sizing

- Trajectory mass ratio for HTOHL 7.875 to 140 x 400 transfer
- Isp 455; gives ideal delta V = 455*g*ln(7.875) = 9208 m/s
- VTOHL is 200 m/sec more (assumed) ≐ 9408 m/s.
- Mass ratio required is 8.236
- Values from VTOHL/Venture Star comparisons 8.24 to 8.26
- HTO with payload stage (TSTO hi-stage) 250 m/s less; 8958 m/s
  - 200 m/sec less for ascent and 50 m/sec injection
  - Also does not need payload bay or orbit maneuver propellant
- Mass ratio required is 7.445
- TSTO lo-stage is 600 m/sec less (est) = 8608 m/s
- Mass ratio required is 6.884

Sizing the RBCC Installation: 1

The RBCC system includes two separate propulsion systems. The RBCC is a LOX-kerosene system with 32 thrust chambers in a fixed-geometry inlet/mixer/nozzle air duct system. The rockets provide static thrust for takeoff, are augmented by the air system during subsonic and transonic flight, and switch to ramjet operation, by shutting down LOX flow, for supersonic flight to the "staging" point at Mach 6.

No hardware separation occurs at Mach 6, but propulsion operation transitions to LOX-hydrogen full flow staged-combustion engines for ascent to orbit. The LOX-hydrogen thrust-to-weight ratio is half that needed for the all-rocket system; 0.42 versus 0.84.

During the prior phase of study, it was determined that the RBCC system should strive for low inert weight rather than high staging Mach number, and that this design strategy is enhanced by the selected propulsion combination. The LOX-kerosene RBCC is minimum inert weight. Kerosene is carried in integral wing tanks, using existing structure with a minimal weight for accommodation of the kerosene (as compared to a large delta inert weight required to accommodate hydrogen in the case of a LOX-hydrogen RBCC). Separate ascent propulsion engines provide optimized Isp for the vacuum portion of the trajectory where it is most important.

The facing page summarizes sizing of the LOX-kerosene RBCC engines.

Sizing the RBCC Installation: 1

- RBCC installed T/W without augmentation 0.75
- Takeoff weight 10^6 kg = 2205 klbm
- Rocket thrust 1653 klbf
- Jet mixing length 10:1 in 5 m yields 0.5 m (20") exit
- Pc 1000 psia, exit 6 psia, ratio 150, area ratio 15
- Thrust coefficient 1.6; throat area 314/15 = 21 sq in
- Thrust = 21*1.6*1000 = 33,600 lb.
- Number of thrust chambers = 1653/33.6 = 49
- If we increase the mixing length to 6 m we can use 32 thrust chambers at 50 klbf, a more convenient number

Sizing the RBCC Installation: 2

Analyses conducted during the first phase of study indicated a bypass ratio about 3 was appropriate for the RBCC at low Mach number. The facing page summarizes the calculation of capture area at Mach 1.

Capture area as defined in this study is the cross-section area of the air stream flow which actually enters the inlet. If the inlet does not spill air, it is equal to the physical capture area of the inlet. Most supersonic inlets spill air at low Mach numbers and capture all of it at higher Mach numbers. As will be seen, this is appropriate for the RBCC design.

Sizing the RBCC Installation: 2

- Rocket Isp assumed 250; rocket flow 6612 lbm/s
- Bypass ratio 3 at Mach 1 assumed $u = 330$ m/s
- $\rho Au = 3*6612 = 19836$ lbm/s $= 9000$ kg/s
- $\rho u^2/2 = Q = 45000$ Pa; $\rho u = 2*45000/u = 273$ kg/s per m^2
- $A = 9000/273 = 33\ m^2$
- This is flow capture area; physical capture area needs to be larger because the inlet will spill air at Mach 1.

Sizing the RBCC Installation: 3

This page describes calculation of the capture area required at maximum Mach number. Above about Mach 2, the required capture area increases with Mach number so it is appropriate to size the inlet for the maximum Mach number. The other sizing determinant is transonic drag rise, but the RBCC has the rocket on in this speed regime, providing plenty of thrust.

This report includes an appendix, of which one subject is analysis of achievable thrust coefficients, based on consideration of conservation of energy, second law of thermodynamics, and practical efficiency factors.

Sizing the RBCC Installation: 3

- The second criterion for sizing the inlet is adequacy of thrust at maximum Mach number
  - $M_{max} = 6$: u = 1800 m/s
  - Estimated thrust coefficient 0.6 to 0.7
  - Estimated L/D 2.5
  - Mass 350 mt
  - Thrust 1.25* sustain cruise = about 350,000*g*1.25/2.5 = 1.7 MN
  - Q = 45000
  - $A_{min} = 1.7 \times 10^6/(45000 \times 0.6) = 63$ sq m; no spilled air

Concept Illustrations: Assisted HTOHL and RBCC

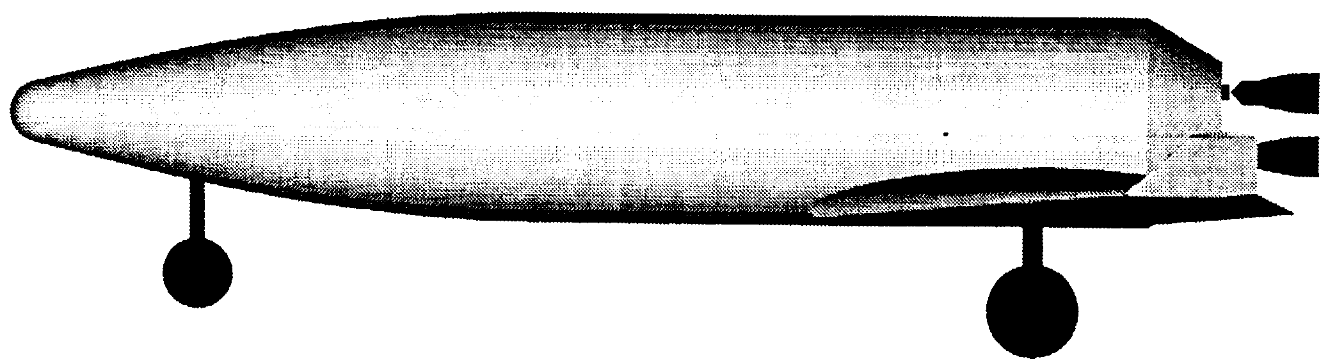
The next several pages present illustrations of the assisted HTOHL and the RBCC option. These illustrations were developed with AutoCad™ and are geometrically accurate with respect to the analytical models used in the previous study phase. The landing gear is notional and does not represent an actual landing gear design. The assisted HTOHL is a simple wing-body shape with a swept trapezoidal wing and tip fins. Wing fore-and-aft location re c.g. and balance was not analyzed. Top-level statistics are presented with the aft views.

The RBCC illustration shows the proper RBCC duct capture area and length, and approximately the correct exit area, but is otherwise notional. The body and wing geometry are the same as for the assisted HTOHL. Inlet location over the wing provides more room for the large capture area required, makes protection of the inlets during reentry easier, and eliminates the need for the extended landing gear needed for ground clearance with under-wing engine installations. The over-the-wing location has a slight disadvantage in that forebody pre-compression of the air is less since the vehicle flies at a slight positive angle of attack.

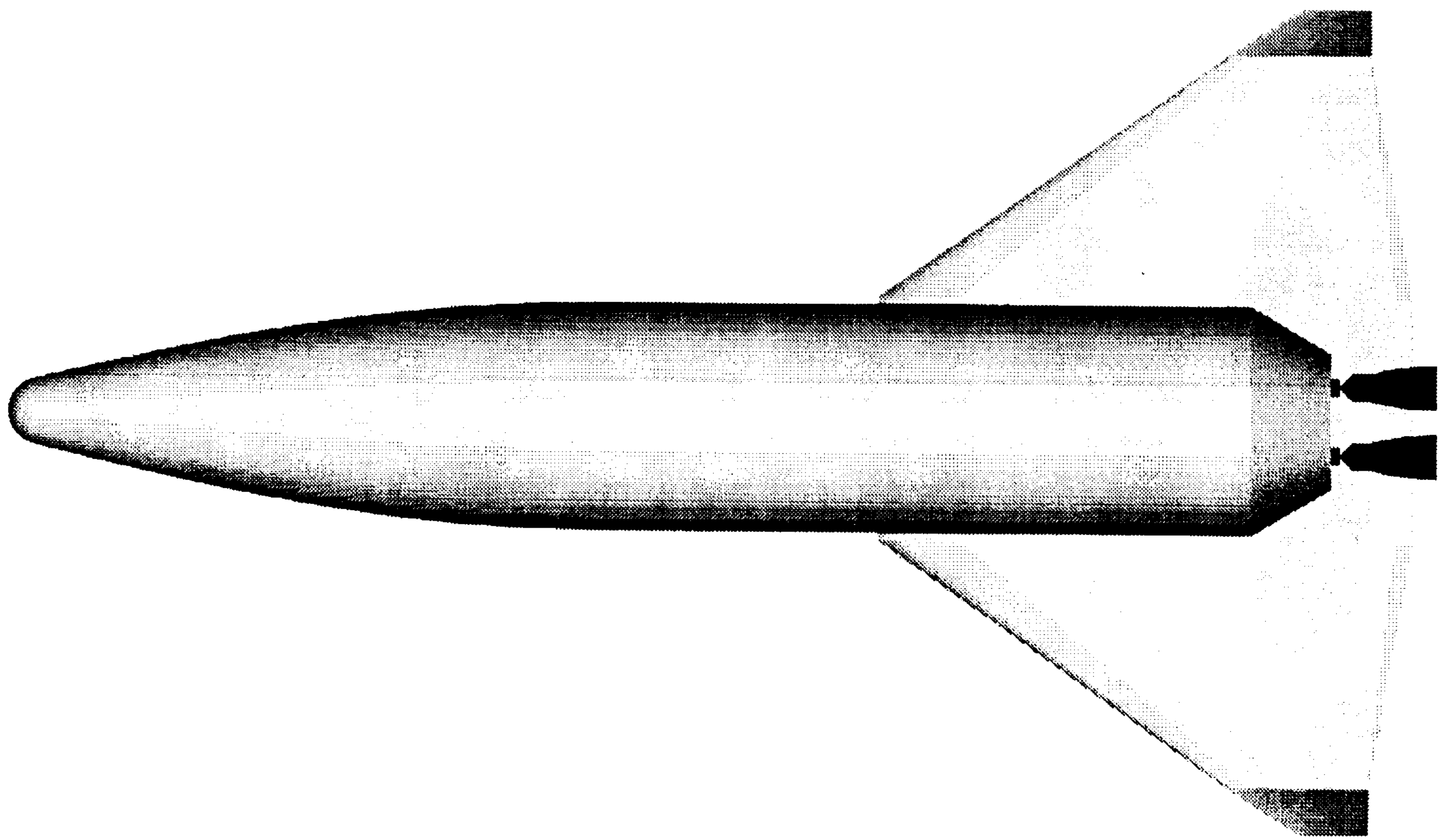
The RBCC is distinctly different than the "spoonbill" forebodies often associated with high-speed airbreather vehicles. This is in keeping with the design approach of minimum inert weight. Non-circular bodies with propellant tanks are substantially heavier than circular ones. Even if the tanks are conformal, they need internal stiffening to maintain a non-circular cross section while pressurized. The lobe tank design used for the X-33, in theory, does not incur added stiffener weight but requires a separate skin outside the lobes for a smooth aero contour. The RBCC design presented here would benefit from a somewhat higher fineness ratio, and smaller radii for the nose cap and wing leading edges. These changes would cost little inert weight and would significantly reduce drag losses.

Concept Illustrations: Assisted HTOHL

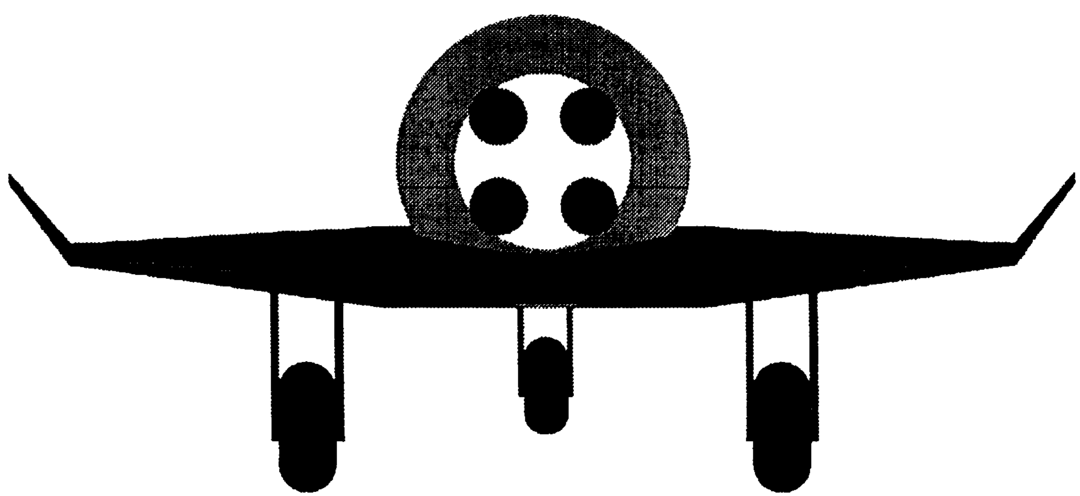
Side View



Assisted HTOHL Top View

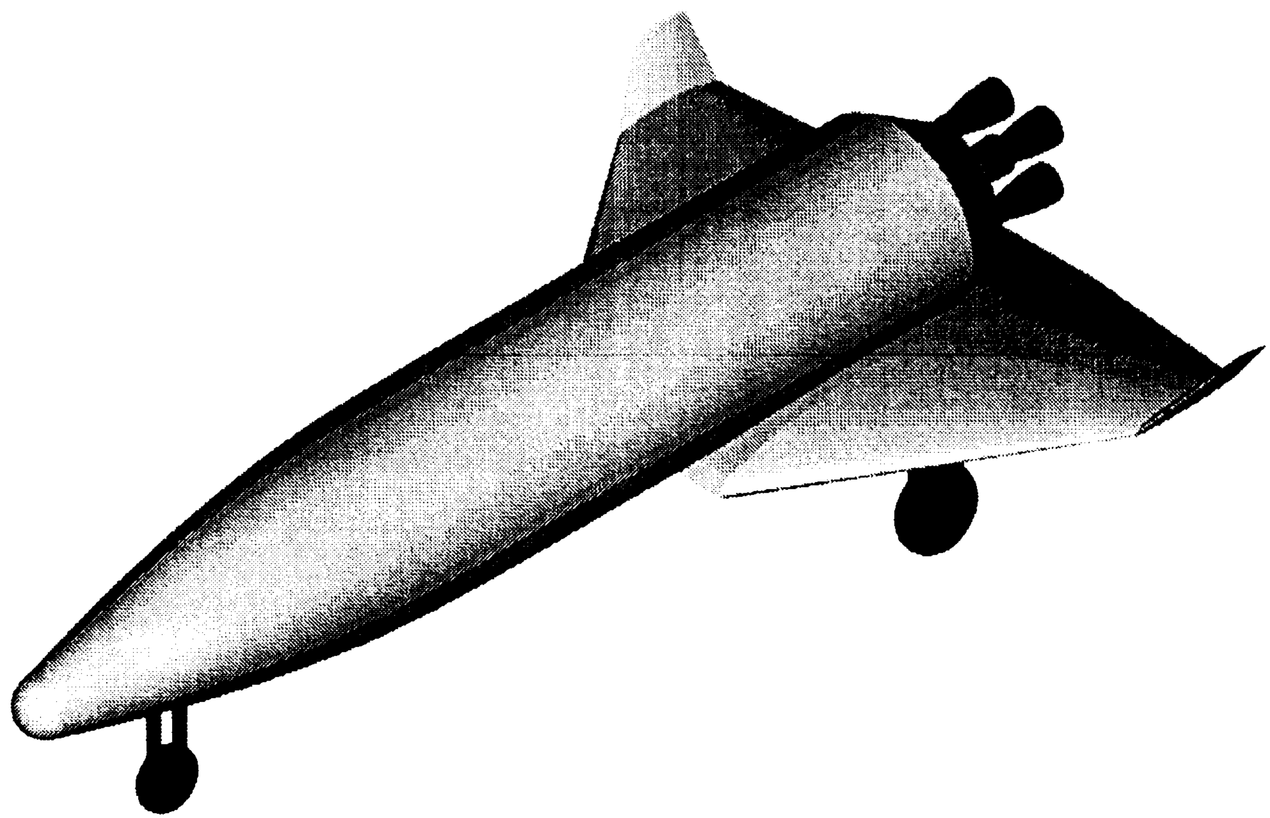


Assisted HTOHL: Rear View and Statistics

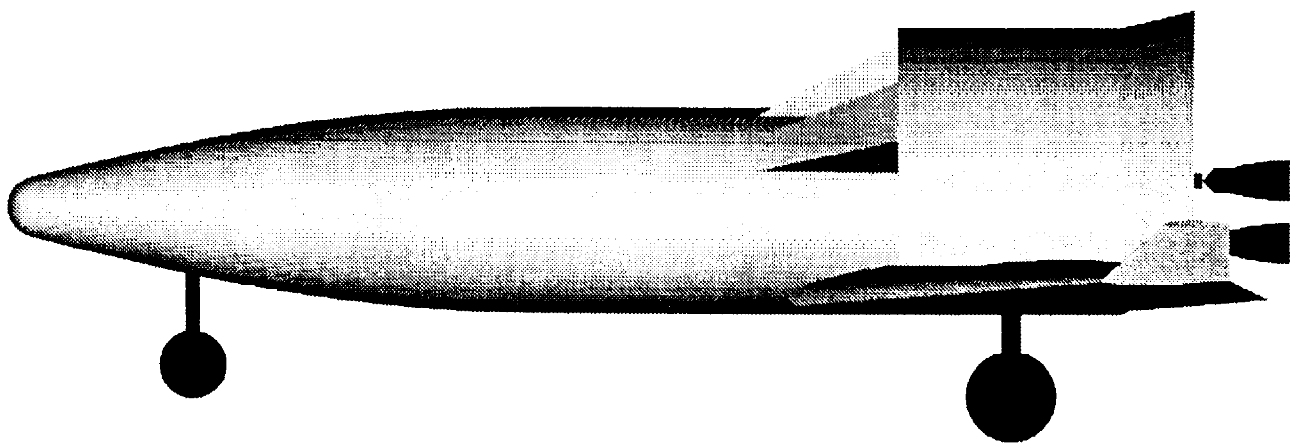


Overall length 211 ft.
Wingspan 122 ft.
Body diameter 34 ft.
Takeoff weight 2.2 M lbm
Installed thrust 1.85 M lbf
Payload 25,000 lb
Payload bay 15 x 45 ft
Ground accelerator
speed Mach 0.6

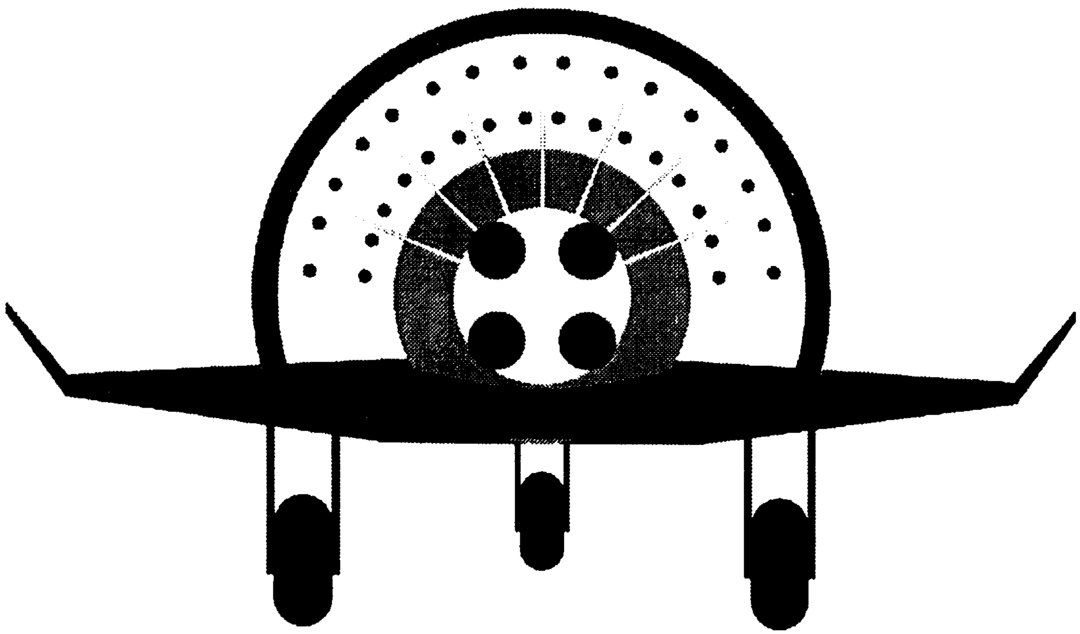
Assisted HTOHL Perspective



Assisted RBCC Side View



Assisted RBCC Rear View and Statistics



Overall length 211 ft.
Wingspan 122 ft.
Body diameter 34 ft.
Takeoff weight 2.2 M lbm
Installed thrust 1.65 M lbf
(RBCC unaugmented),
50 thrusters at 50 klbf
925,000 lbf
Ascent to orbit (SL rating)
Payload 25,000 lb
Payload bay 15 x 45 ft
Ground accelerator
speed Mach 0.6

Sensitivities to Payload and Weights Margin

The original assisted HTOHL design was set up for access to the Space Station with 25,000 lb payload. Weights conservatism was included in subsystems weights estimates rather than being explicitly added. Later, the HRST study selected 20,000 lb and an explicit 15% weights margin as baseline. Sensitivity studies were requested to show how takeoff and dry weights would vary for payload from 20,000 to 40,000 lb and inert weights margins 15% and 25%.

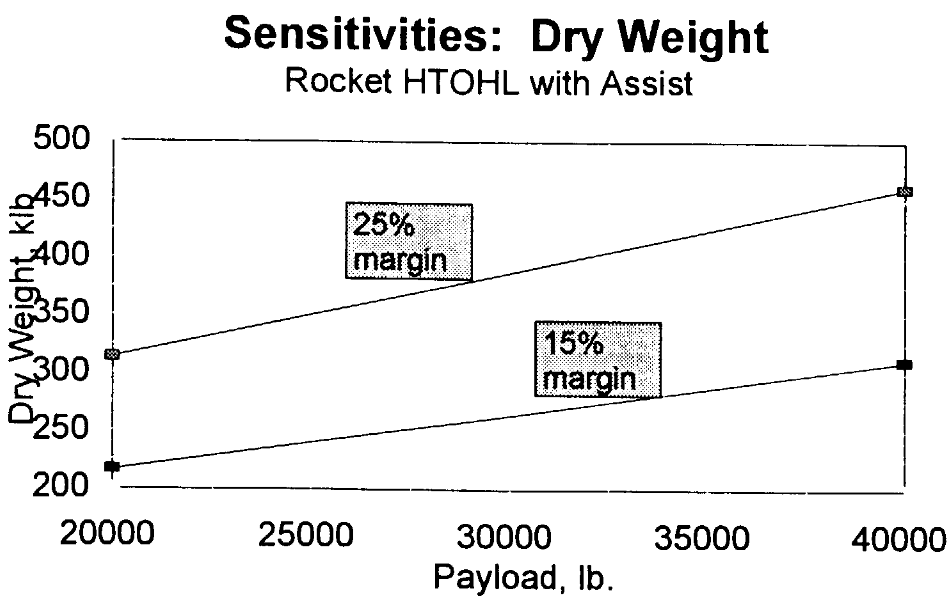
In order to supply the requested data, the assisted HTOHL weights were analyzed for conservatism and some of the built-in conservatism was converted to explicit weights margin. Main areas for changes were structure and TPS, which were quite conservatively estimated compared to related HRST concepts.

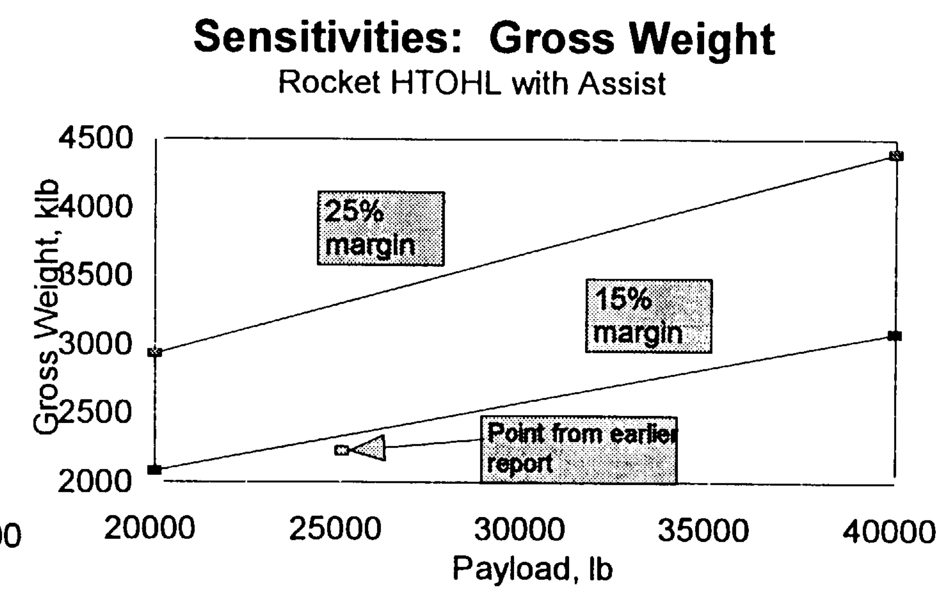
Once these changes were made, the iterative spread sheet for the assisted HTOHL was used to generate the sensitivity data presented on the facing page.

This chart depicts sensitivities for the assisted HTOHL "as weighed", not as adjusted for comparison to the NASA Access to Space reference.

Sensitivities to Payload and Weights Margin

Based on Original Weights Algorithms





Dry Weights Comparison Graphic

Also requested for the HRST comparisons was a comparison of the assisted HTOHL weights of the present study to the NASA Access to Space Study Reference vehicle. This request was made because advantages and disadvantages of the different HRST concepts were obscured by differences in weights conservatism by the various investigators.

For this comparison, the assisted HTOHL weights after extraction of explicit margin were used. Since the two vehicles had somewhat different gross weights and significant differences in engine and wing weights due to the different takeoff mode, the comparison was made on the basis of percentage of main propellant load, and in instances where weights are not a strong function of propellant load (such as avionics) by direct comparison.

The facing page is a graphic of the comparison raw results. The bar lengths are drawn for the percent of main propellant load represented by each subsystem. Numbers on the bars are the actual weights in pounds mass for each element (i.e. do not try to read the numbers as direct measures of the bar lengths).

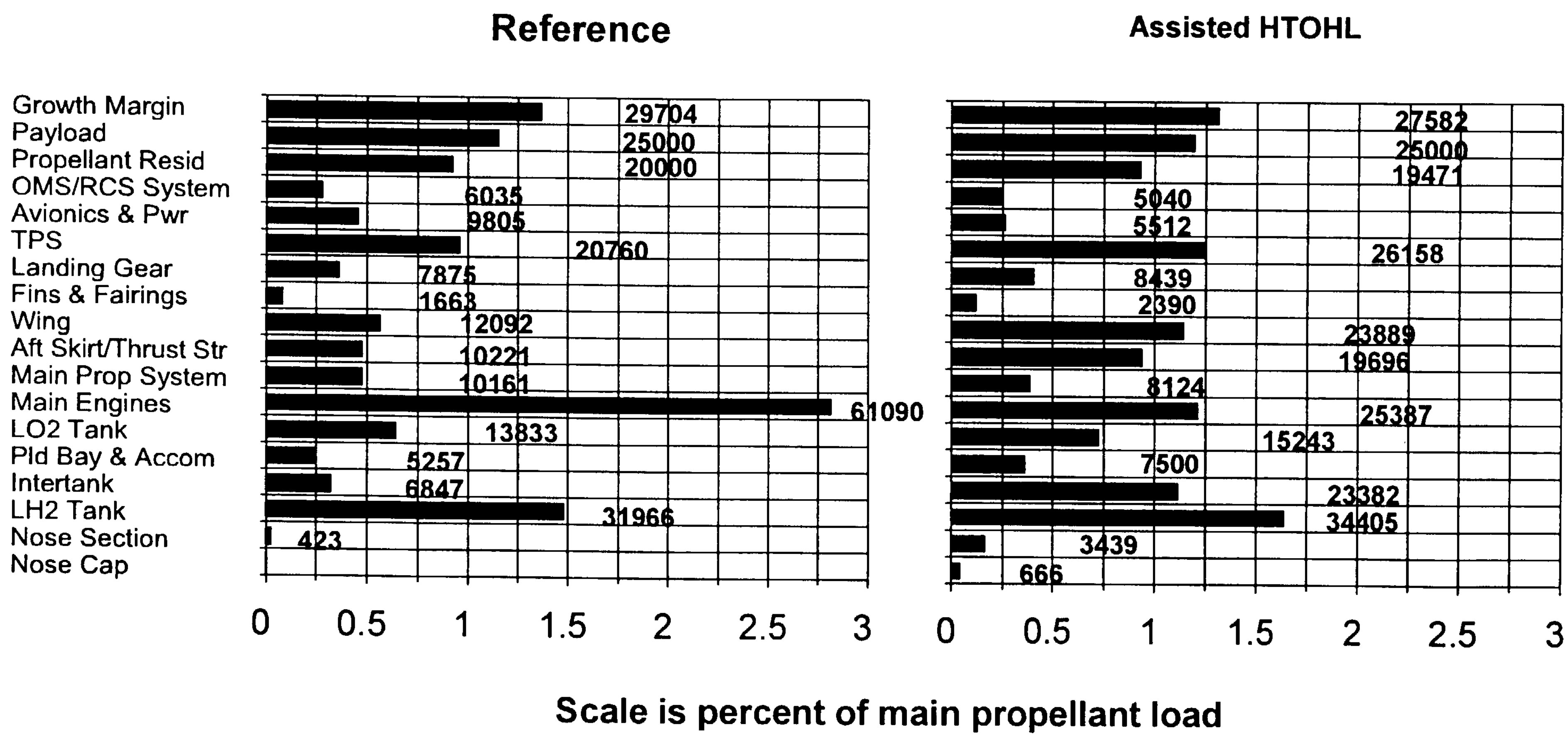
The reference data as provided did not include residual propellants. This value was estimated, using the same general allowance for residuals as for the assisted HTOHL. Engine weights were obtained by telecon with John Mankins of NASA.

Dry Weights Comparison Graphic

Comparison Based on Percent of Main Propellant Load;

Assisted HTOHL is Original Weights Algorithms

Note: Values on bars are weights in lb.



Comparison Preliminary Observations

As these systems were compared, the most striking differences appeared in structures and TPS. For the reference system, the wing is and should be much lighter and the engines are and should be heavier. No adjustments were made to wing weight estimates. The reference system used much more conservative engine weight assumptions, with thrust-to-weight approximately that of the SSME, and in addition has much higher installed thrust-to-weight. For the present study, engine weights about median for the various HRST engine projections were used. These correspond to a next-generation full-flow staged combustion engine fabricated from metallics (i.e. no extensive use of ceramics). The reference main propellant tanks and TPS were somewhat lighter, and the nose section, intertank and thrust structure markedly lighter. Avionics and power for the reference were heavier. The reference analysis of these subsystems was considered to be in greater depth than that for the present study, so assisted HTOHL weights for avionics and power were adjusted upward for the comparison vehicle.

A cursory comparison was also made for a somewhat comparable "Argus" concept which was provided on the same spread sheet. Relatively little weights detail was available for this concept, but it appeared to have less weights conservatism than either the reference or assisted HTOHL concepts.

Comparison Preliminary Observations

- Body weight differences Reference to HTOHL are nose section and thrust structure. Reference looks light, higher T/W and much less thrust structure weight. Reference intertank is lighter in part because transverse payload bay leads to shorter intertank.
- HTOHL has more optimistic engines, vacuum T/W 90 vs 55 for reference.
- HTOHL avionics and power appear light; reference design probably has more depth of analysis in this area.
- Argus body looks very light considering smaller size, high fineness ratio and large H_2 volume (lower tanked mixture ratio).
- Argus main propulsion looks light, considering SERJ cycle (has fan).
- Argus TPS looks light. Smaller size should lead to higher TPS fraction relative to other systems.

Adjustments to Assisted HTOHL

In order to create a more valid comparison of the merits of the adjusted HTOHL concept, adjustments were made as summarized on the facing page, to bring its weight estimating conservatism in line with that for the reference vehicle. Those areas where the present investigator believes these estimates are unrealistically low are noted on the graphic.

In general, the reference vehicle was more conservative on engine weights and less conservative on structural weights. The net was that the reference vehicle overall was less conservative.

A caveat is important for the intertank weights comparison. The reference vehicle, in order to minimize intertank weight, installed its 15 x 30 ft payload bay in transverse orientation. The assisted HTOHL uses a longitudinal installation with a 35 ft long payload bay. This leads to a much longer and heavier intertank region. The intertank estimate for the reference vehicle is not necessarily optimistic for its design. After this reference design was created, deliberations that occurred in the RLV study activities determined that the payload should be installed longitudinally. The reference intertank weight is optimistic for this payload orientation.

Adjustments to Assisted HTOHL

Rationalizing to Reference for Gross and Empty Weight Comparisons

- Reduce nose section from 0.16% to 0.05% MPL (unrealistic)
- Reduce LH2 tank from 1.63% to 1.5%
- Reduce intertank from 1.1% to 0.32% (unrealistic)
- Reduce payload bay & accommodations from 7000 to 5300 lb
- Reduce LO2 tank from 0.73% to 0.64%
- Decrease main engine T/W from 90 vac to 55 vac.
- Reduce thrust structure and aft skirt from 0.94% to 0.47% (unrealistic)
- HTOHL wing is much heavier but should be much heavier
- Reduce TPS from 1.28% to 1%
- Increase avionics and power from 5500 lb to 9800 lb
- Increase OMS/RCS propellant system from 0.24% to 0.28%
- Decrease OMS/RCS propellant from 1.27% to 1.06% (this probably reflects different on-orbit delta V requirements)
- Include engine weights in weights margin basis

Weights Comparison Worksheet

Presented on the facing page is the weights comparison worksheet for the previous 3 pages. Weights are given in kg and lbm and in percent of main propellant load. This worksheet shows the reference weights, the "as-weighed" assisted HTOHL, and the revised assisted HTOHL. Values in bold italics for the reference were inferred and not acquired from the source data. The HTOHL has a lesser takeoff to injection ratio because its delta V to orbit is slightly less. General inferences about engines are also provided on the worksheet. The reference vehicle has a takeoff thrust-to-weight 1.3 and the HTOHLs 0.84.

The revised assisted HTOHL was iterated on the spread sheet to obtain approximately the correct takeoff/injection ratio.

In addition to other differences, there was a difference between the reference and this study in interpretation of the 15% margin. The reference included engine weights in the dry weight basis for the 15%. The present study presumed that the engine weight estimate by the engine contractor already includes a weights growth margin (this was true for the HRST engine presentations) and did not add further margin.

Weights Comparison Worksheet

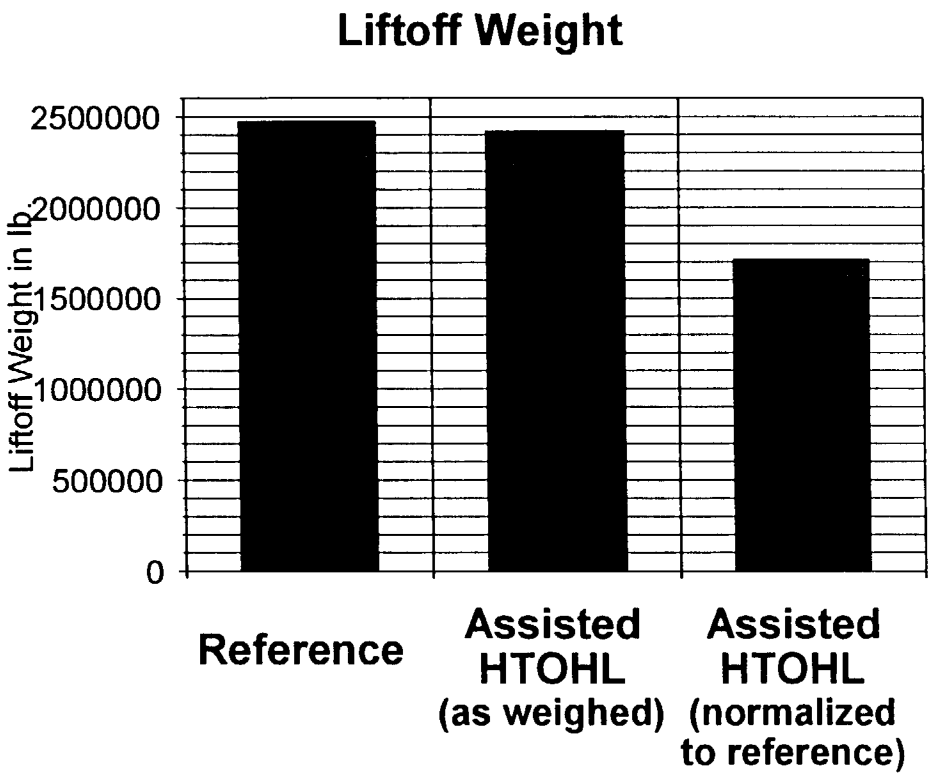
Element	Reference Weights kg	Reference Weights lb	%mpl		HTOHL "as-weighed" Weights kg	HTOHL "as-weighed" Weights lb	% mpl		HTOHL Adjusted Weights kg	HTOHL Adjusted Weights lb	% mpl
Nose Cap	0		0		302	665.789	0.03155		240	529.104	0.03537
Nose Sec	191.872	423	0.01945		1560	3439.18	0.16298		419	923.727	0.06175
LH2 Tank	14499.7	31966	1.4699		15606	34405	1.63038		10478	23099.8	1.54429
Intertank	3105.78	6847	0.31485		10606	23382	1.10802		2465	5434.34	0.3633
Pld Bay & Accom	2384.56	5257	0.24173		3402	7500.05	0.35541		2586	5701.1	0.38113
LO2 Tank	6274.61	13833	0.63609		6914	15242.6	0.72232		4440	9788.42	0.65438
Main Rocket Engines	27710.4	***61090***	2.80914		11515.515	***25387***	1.20304		11865.7	***26159***	1.74881
Main Prop Sys	4609	10161	0.46724		3685	8123.95	0.38498		2612	5758.42	0.38497
Aft Skirt/Thrust Str	4636.22	10221	0.47		8934	19695.9	0.93335		3334	7350.14	0.49138
Wing	5484.9	12092	0.55603		10836	23889	1.13205		7304	16102.4	1.07649
Fins & Fairings	754.332	1663	0.07647		1084	2389.79	0.11325		730.4	1610.24	0.10765
Landing Gear	3572.08	7875	0.36212		3828	8439.21	0.39992		2712	5978.88	0.39971
TPS	9416.67	20760	0.95461		11865	26157.6	1.23955		7408	16331.7	1.09182
Avionics & Power	4447.52	9805	0.45087		2500	5511.5	0.26118		4445	9799.45	0.65512
OMS/RCS Prop Sys	2737.46	6035	0.27751		2286	5039.72	0.23882		2266	4995.62	0.33397
Propellant Residuals	9071.94	***20000***	0.91967		8832	19471	0.92269		6258	13796.4	0.92233
Payload	11339.9	25000	1.14958		11340	25000.2	1.18471		11340	25000.2	1.67133
Weights Growth Margin (15)	13473.8	29704.2486	1.3659		12511.2	27582.2	1.30706		9077.54	20012.3	1.33788
Landing Wt	123711	272732.5726	12.5411		127606.72	281322	0		89980.6	198371	13.2617
OMS/RCS Prop	10491.7	23130	1.06359		11909	26254.6	1.24415		8437	18600.2	1.24348
Injection Wt	134202	295862.5726	13.6047		139515.72	307576	14.5754		98417.6	216971	14.5052
Empty Wt	103299	227732.5726	10.4719		107434.72	236851	11.2239		72382.6	159575	10.668
Main Prop Load	986438	2174702	100		9.57E+05	2110243	100		678500	1495821	100
Takeoff Wt	1120641	2470564.573	113.605		1096715.7	2417819	114.575		776918	1712793	114.505
T/O to Inj ratio		8.350378864				7.86088				7.89409	
lbperkg 2.2046											
Liftoff Thrust		3359967.819				2030968				1438746	
No. Eng		7				4				4	
Thrust per engine		479995.4027	lbf			507742				359686	
Wt per engine		8727.189139	lbm			6346.78				6539.75	
Vac Thrust		560491.5261				592891				420007	
T/W vac		64.22360248				93.4161				64.2236	

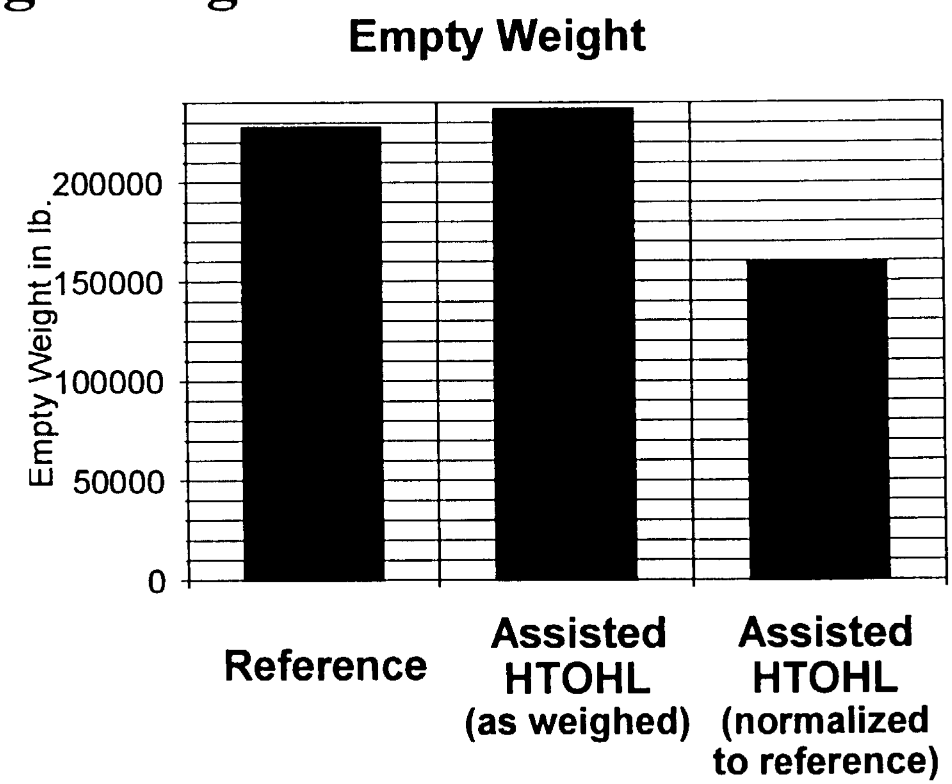
Comparison Results

The comparison results are presented graphically on the facing page. Adjusting the weights conservatism for the concepts to be directly comparable shows a significant advantage in takeoff and empty weight for the assisted HTOHL concept. This concept has the further advantage of reduced installed thrust, leading to less engine initial and operating cost.

Comparison Results

Rationalized Weights Algorithms





Notes: (1) Reference supplied by MSFC
(2) Assisted HTOHL is all-rocket with takeoff assist to 200 m/sec
(3) Normalization process consisted of adjusting inert weights so that assisted HTOHL and reference are directly comparable, and de-escalating HTOHL

TSTO Design Strategies

Staging has been recognized as a way to improve the performance of rocket propulsion systems since the early work of Tsiolkovskii, Oberth and Goddard. Staging, however, complicates the vehicle and its operations. Studies of staged reusable space launch vehicles began with the von Braun “Across the Space Frontier” rocket concept of 1953, and somewhat later, concepts for wings on the Saturn stages. The shuttle is a partially reusable staged vehicle. Most study concepts of staged fully reusable vehicles have been performance-optimized. As a consequence, operational complexities arise, such as booster flyback to the launch site.

For the present study, we chose to examine operations-optimized concepts. Staging provides significant performance advantages even if far from the performance optimum. This is especially true if one wishes to reduce the sensitivity of an SSTO vehicle to weights growth by staging.

Our operations concepts were: (1) Stage late (“Hi stage”) so that the boost stage goes once around the Earth to return to the launch site, while the orbital stage is a much smaller mission-tailored vehicle with relatively little delta V, and (2) stage early (“Lo stage”) so that boosters are recovered only a few miles down range. In this concept the boosters can be simple low-technology and low-cost units with simple (parachute) recovery. Additional information is given on the facing page.

TSTO Design Strategies

Maintain SSTO benefits while reducing inert weight sensitivity

- 1: Design an SSTO-like vehicle which goes once around and delivers a reusable payload carrier vehicle to orbit
  - Payload carrier reusable,winged runway landing
  - SSTO-like booster does not need:
    - Payload bay or long intertank
    - Orbit maneuvering propellant or system
- 2: Provide rocket boosters for HTOHL to eliminate assisted takeoff ground infrastructure
  - Vertical takeoff on booster thrust; pitch over to aerodynamic climbout to preserve HTOHL low T/W
  - Boosters separate between Mach 1.2 & 1.4
    - Through transonic drag rise
    - Minimal down-range recovery point (~ 15 km)
    - Simple hybrid boosters, parachute sea recovery

"Traditional" TSTO

Numerous design studies have been done for fully reusable two-stage-to-orbit launchers, notably the space shuttle Phase A and Phase B studies which examined a wide range of options, and many more recent design studies performed by the NASA Langley Research Center as well as other NASA and contractor organizations..

The spread sheet tools for the present study were used to synthesize a traditional TSTO rocket as described on the facing page, as a check point for evaluating the somewhat unconventional TSTO options considered in the study.

These concepts are usually complicated by the need to fly the booster back to the launch site. This limits the staging velocity to values somewhat less than otherwise optimal. As noted on the chart, liftoff mass can be reduced but at the expense of operational complexity. Further reduction is in fact possible by using parallel burn and cross-feed of propellants, but complexity becomes still greater.

In recent years, people have paid more attention to the economics of reusable launchers. Minimum launch mass is 6th priority, after R&D and unit cost, operational risk, airframe life, turnaround time, and operational complexity. The "traditional" TSTO pays penalties in all these high-priority areas for a modest decrease in launch mass.

A note on aribreathing first stages for TSTO is also in order. An airbreathing first stage pays the same penalties, in greater measure, for a significant reduction in launch mass compared to all-rocket systems. However, when one evaluates economic factors, the airbreathing option has few, if any, redeeming virtues to offset its disadvantages.

"Traditional" TSTO

Fully reusable 2 stage to orbit rocket

- Ideal V stage ~ 3000 m/s (actual about 1500 m/s; Mach 5)
- Flyback booster (jet engines)
- VTOHL
  - Propellant loads are too similar for practical piggyback HTOHL
  - Either tandem or parallel burn design; data shown below assume serial burn and no crossfeed.
- Representative sizing:

	kg		kg
Liftoff mass	**778,950**	**2nd stage sep mass**	**237,330**
Boost propellant	**348,000**	**Boost propellant**	**185,600**
MECO mass	**430,950**	**MECO**	**51,730**
1st stage separation mass	**193,620**	**OMS propellant**	**4,415**
Flyback propellant	**89,400**	**Landing mass**	**47,310**
Landing mass	**103,970**	**Payload**	**11,340**
Empty mass	**99,870**	**Empty mass**	**34,260**

The traditional TSTO is about 9% less liftoff mass than the "lo stage" TSTO to be described, but requires development and operation of two large high-speed airframes. It is far more complex operationally.

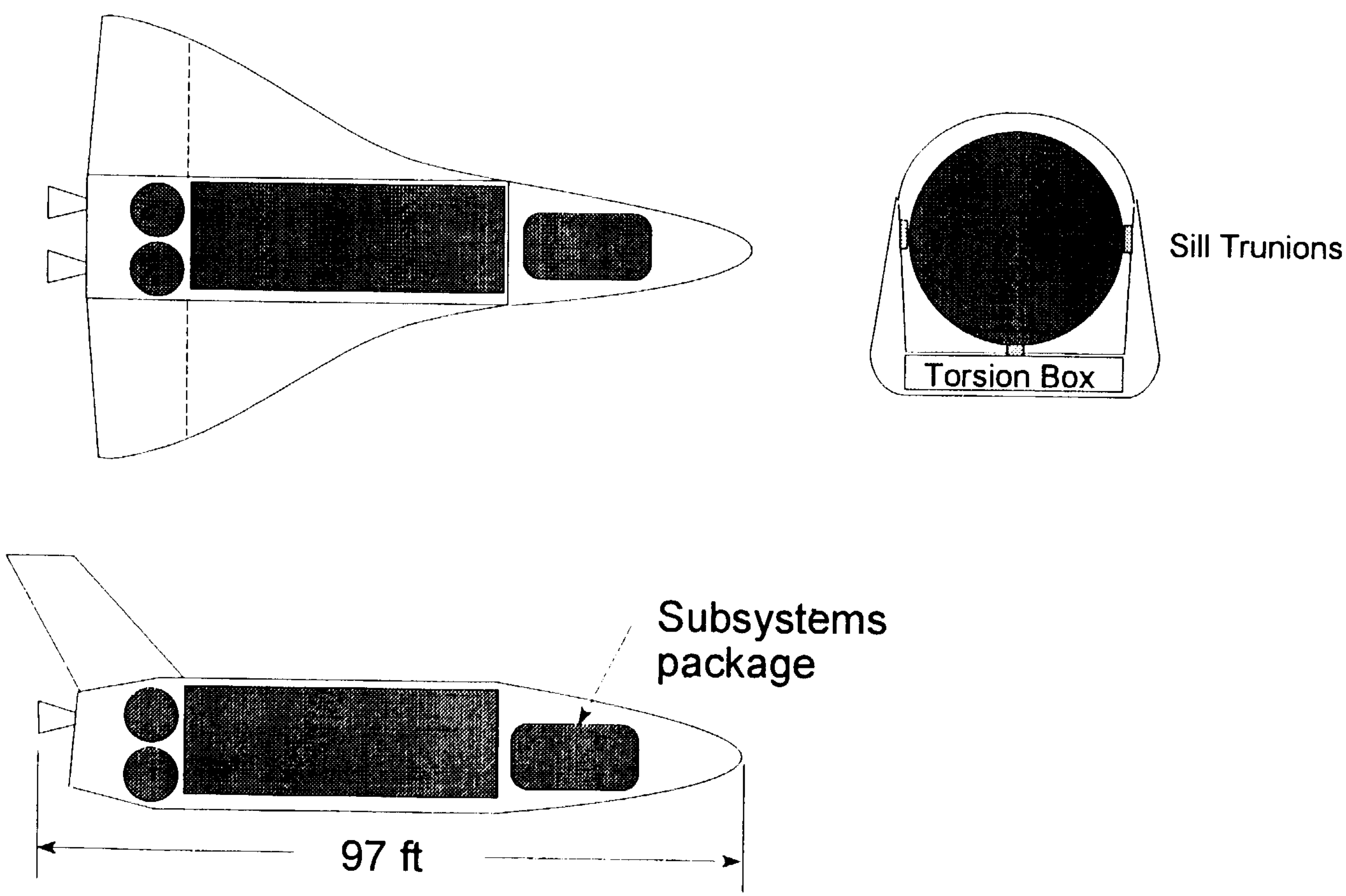
Payload Carrier Stage Concept

The payload carrier for the "hi-stage" concept is a smaller shuttle-like vehicle with kerosene-oxygen propulsion. The relatively low delta V of the stage leads to a preference for a moderate-performance easily handled propellant combination. LOX tank insulation can be a simple foam/vapor barrier system. The fuel tanks do not need insulation. The propellant combination is non-toxic and needs no special handling except cleanliness in the LOX systems. An ignition source is required but the needed technology is available.

The sketch on the facing page was prepared to facilitate weight estimates. Weight estimate details are provided in the appendix to this document.

Payload Carrier Stage Concept

15 x 45 ft payload bay



Mass Estimate for Payload Stage

The mass estimate is summarized on the facing page. Additional information, method and rationale are given in the appendix to this document.

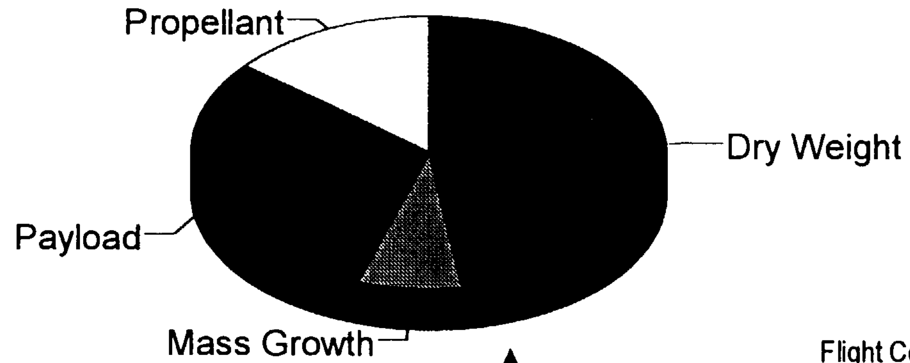
Mass Estimate for Payload Stage

Mass Breakdown Summary		
	kg	lbm
Nose Section	**590**	1300
Body Section	**2022**	4458
Payload Bay	**1730**	3815
Thr Str & Base Ht Sh	**278**	612
Wing	**6102**	13451
Tail	**610**	1345
Landing Gear	**572**	1260
TPS	**2640**	5820
Prop Tanks & Press	**1006**	2219
Propellant Feed Sys	**177**	391
Main Engines	**183**	404
RCS	**40**	88
Elec Power & MPAD	**484**	1066
Distribution	**250**	551
Avionics	**235**	518
Flight Controls	**618**	1362
Subtotal	**17537**	38661
Mass Growth 15%	**2630**	5799
Dry Mass	**20167**	44460
Payload	**11340**	25000
Landing Weight	31507	69461
Residual Propellant	**126**	277
Impulse Propelllant	**5032**	11094
Takeoff Weight	**36665**	80832

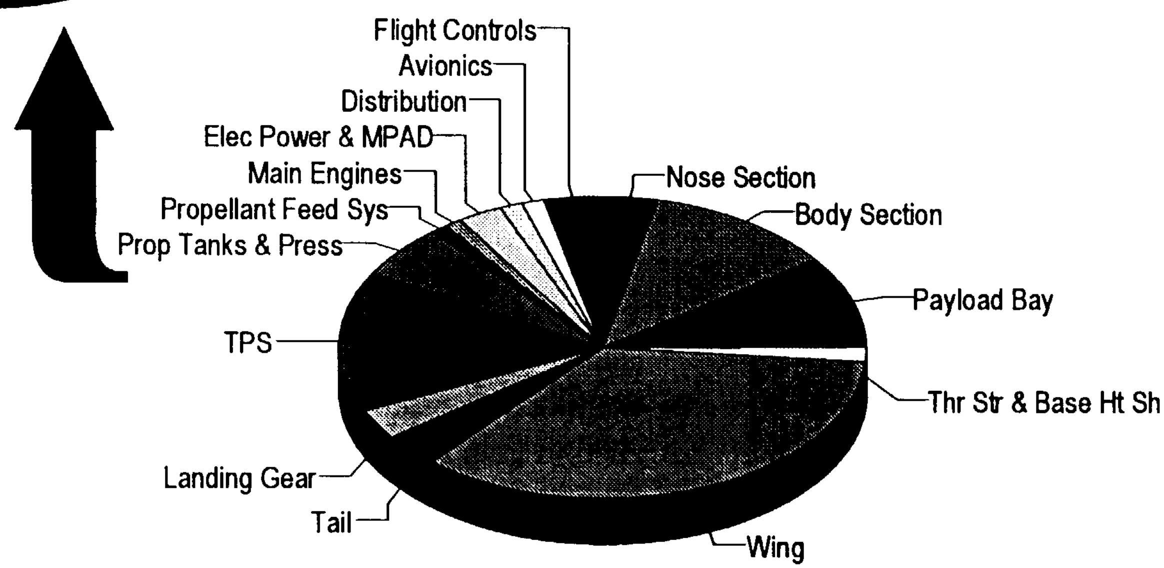
Hi Stage TSTO Payload Stage Weight Estimate

The facing page presents a graphic of the weight estimate.

Hi Stage TSTO 2nd Stage Weight Estimate



Total Weight Distribution



Dry Weight Distribution

Hi Stage Overall Weights Summary

Once a weight was obtained for the payload stage, this value was used as payload for the boost stage. The spread sheet for the assisted HTOHL was modified to eliminate the payload bay, reduce the mass ratio, and eliminate on-orbit maneuver propellant and systems (reaction control was retained as it is needed for reentry). Results are summarized on the facing page. The hi-stage system does not reduce overall weight as compared to the assisted HTOHL. The reason appears to be the substantial overhead required to make the payload stage a fully functional recoverable vehicle. This weight is much more than the weight removed from the boost vehicle, so that an overall advantage is not obtained.

Hi Stage Overall Weights Summary

Payload Carrier Stage	36,600	(80,690)
Once-Around Booster		
▸ Gross Weight	▸ 1,048,800	(2,312,000)
▸ Empty Weight	▸ 97,478	(214,900)
Overall Gross Weight	1,085,000	(2,393,000)

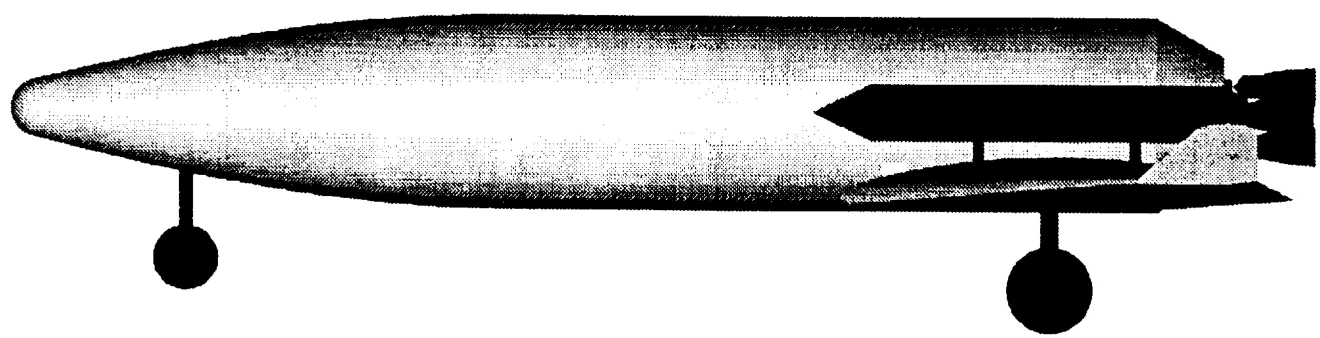
TSTO Lo Stage Concept

The lo-stage concept takes the opposite tack from the hi-stage concept: stage early in the flight. The delta V required of the SSTO-like vehicle is reduced and simple, low-technology boosters with simple recovery can be used. The inert weight penalty of separately accommodating the boost requirement is much less than separately accommodating the payload requirement as in the hi-stage concept, and the penalty is not delivered to orbit. Because the system is de-sensitized for the weight needed to provide boost thrust, vertical takeoff transitioning to horizontal flight offers a means of eliminating the ground accelerator infrastructure while maintaining most of the benefits of horizontal takeoff (some of the refused-takeoff benefit is lost).

The boosters illustrated are LOX-HTPB hybrid motors. LOX-kerosene pressure-fed boosters could also be used. All engines are started before liftoff. Upon boost thrust termination, the spent boost motors are released and separate by sliding aft. The trajectory can be arranged to have a short period of zero lift during the separation event if needed. The orbital stage continues, climbing out on wing lift and LOX-hydrogen rocket thrust. It flies to orbit on a trajectory the same (except for the boosted takeoff) as the assisted HTOHL vehicle.

TSTO Lo Stage Concept

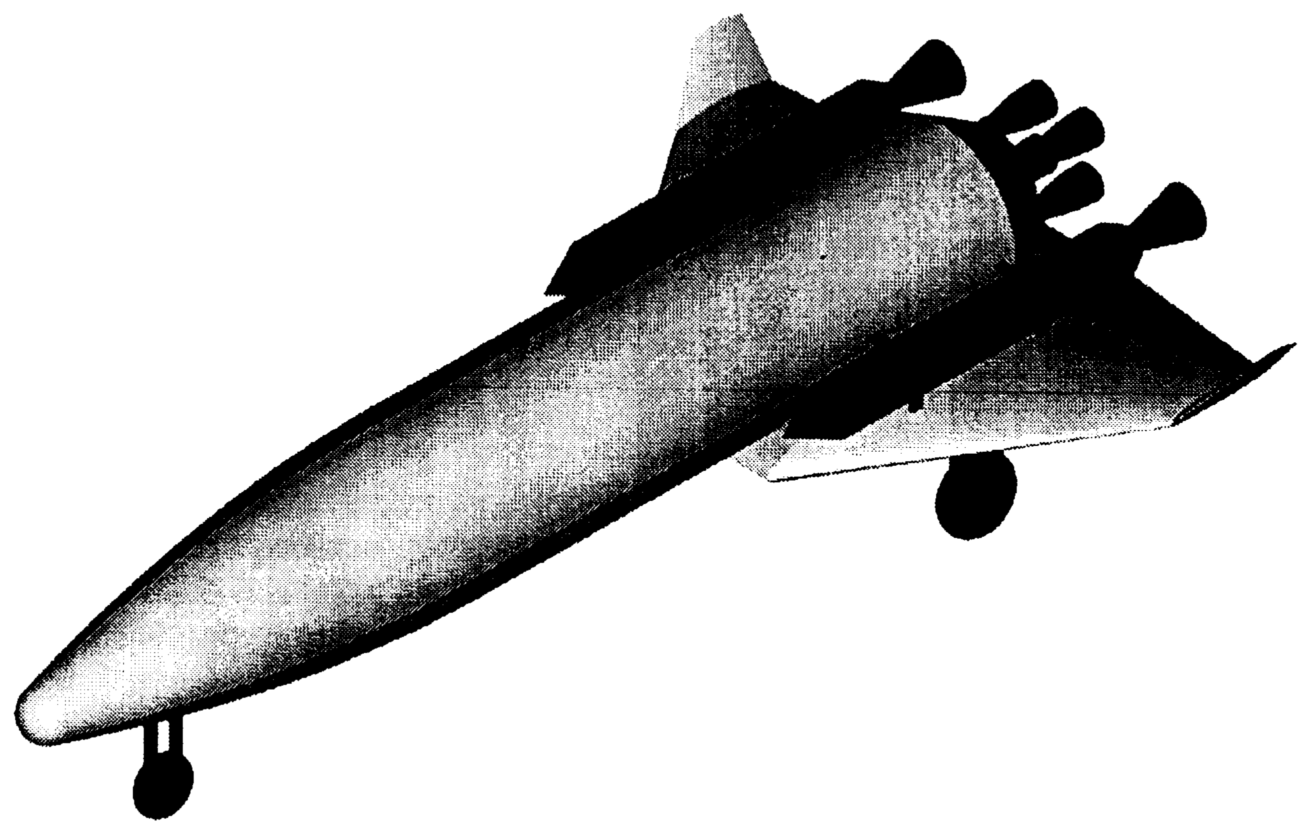
Overall length 193 ft.
Wingspan 104 ft.
Body diameter 29 ft.
Orbital stage takeoff weight 1.38 M lbm
Total takeoff weight 1.88 M lbm
Installed thrust 1.85 M lbf
Payload 25,000 lb
Payload bay 15 x 45 ft
Booster loaded weight 253 klbm each



TSTO Lo Stage Perspective

The facing page presents a perspective view of the TSTO lo-stage concept.

TSTO Lo Stage Perspective



TSTO Lo Stage Weight Estimates

The lo-stage orbital vehicle weight estimate was performed using the assisted HTOHL spread sheet with the appropriate change to overall mass ratio. The delta V provided by the hydrogen-oxygen rocket was assumed reduced by 600 m/sec and the added delta V provided by the boosters as 700 m/sec. Each booster has about 0.5 million lb thrust. The orbital stage engines are throttled to about 75% of their rated thrust during the boost.

The left side of the facing page is the weight estimate for the orbital stage. The weights algorithms are "as-weighed", not adjusted to compare with the reference vehicle, but including the explicit 15% margin. At the right are the sizing estimates for the boosters. Booster acceleration continues through about Mach 1.4 so that transonic drag rise is penetrated at high thrust, reducing drag losses.

OAL stands for overall length.

TSTO Lo Stage Weights Estimate

Orbital Stage

	kg	lb
Nose Cap	204.421	450.667
Nose Sec	916.509	2020.54
LH2 Tank	9165.09	20205.4
Intertank	6901.98	15216.1
Pld Bay & Accom	3402	7500.05
LO2 Tank	4055.46	8940.66
RP-1 Accomm	0	0
Main Rocket Engines	5852.17	12901.7
Main Prop Sys	1872.69	4128.54
Aft Skirt/Thrust Str	5319.88	11728.2
Wing	5710.18	12588.7
Fins & Fairings	571.018	1258.87
Landing Gear	2489.6	5488.57
TPS	8163.83	17998
Avionics & Power	2500	5511.5
OMS/RCS Prop Sys	1661.73	3663.46
Propellant Residuals	4920	10846.6
Payload	11340	25000.2
Weights Growth Margin (15)	7940.16	17504.9
Landing Wt	82986.7	182952
OMS/RCS Prop	7744.88	17074.4
Injection Wt	90731.6	200027
Empty Wt	66726.7	147106
Takeoff Wt	624232	1376181

Booster (hybrid LOX/HTPB)

Isp	295	(Sec vac)
Prop Fraction	0.8	
Delta V	700	(m/sec)
Mass Ratio	1.27375	
Gross Weight	853530	(1.88 M lbm)
Prop Load	183439	(404 k lbm)
Mix Ratio	2.5	
Fuel Dens	1000	
LOX Dens	1140	
LOX load	131028	
Fuel Load	52411	
LOX Vol	120.683	
Fuel load fraction	0.71518	
Fuel Vol	52.411	
Chamber Vol	73.284	
No. of motors	2	
Lox Vol per motor	60.3416	
Fuel Vol per motor	36.642	
Motor Diameter	2.5	(m, = 8.2 ft)
X-sec Area	4.90874	
LOX length	12.2927	
Chamber Length	7.46464	
Nozzle Length	3.5016	
Tare Length	1.5	
OAL	24.7589	(m, = 81 ft)
Chamber L/D	2.98586	
OAL/D	9.90358	

Weights in kg except noted

Note: Wt of 1 loaded booster 252 klbm

Rocket Options Comparison

The facing page compares the rocket options in top-level terms. The assisted HTOHL is from the prior phase of study and the hi-stage and lo-stage options were described in the previous few pages. The hi-stage option offers no particular advantage over the assisted HTOHL except that the large boost stage returns to the launch site after about 2 hours and does not spend time on orbit. The lo-stage option offers less total gross weight and a substantial reduction in the weight ratio total inert/payload, which measures the sensitivity of the system to weights growth.

The lo-stage option can be handled like an SSTO since the boosters are small enough to be attached in horizontal processing. The boosters are entirely inert until tanking with oxygen on the pad. (Depending on the ignition system, a pyro igniter may be present but even if it fired it would not ignite anything without oxygen flow.)

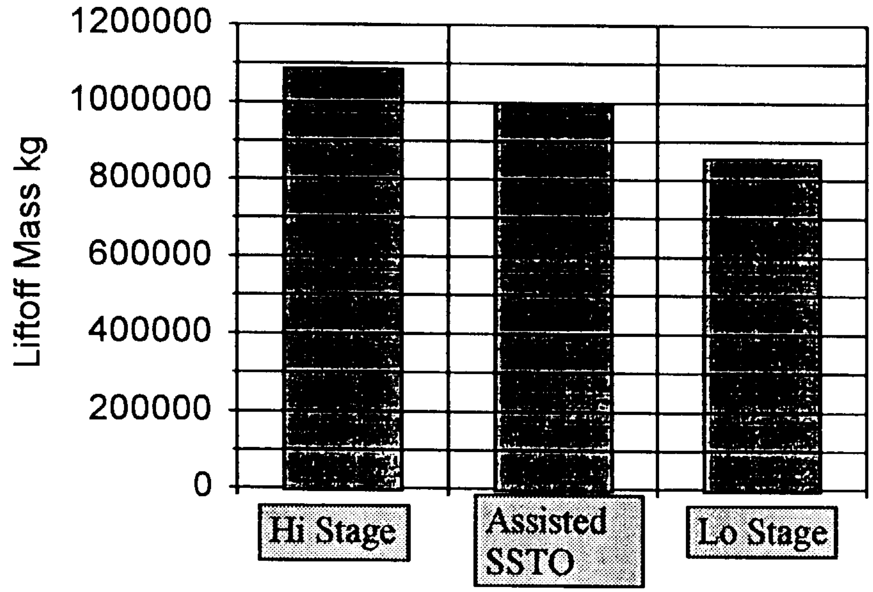
Hybrid or pressure-fed boosters are simple, rugged structures amenable to parachute sea recovery. These boosters can use liquid injection thrust vector control (LITVC) and need no moving parts except valves and regulators. Only rudimentary electronics are required, mainly for operation of the recovery system. It should be practical to design these for no special refurbishment due to salt water exposure. The hybrid fuel grain must be replaced each flight and the nozzle will probably require a new liner. A pressure-fed system could be designed to “gas and go” except for fitting with a re-packed parachute.

Rocket Options Comparison

Based on Original Weights Algorithms

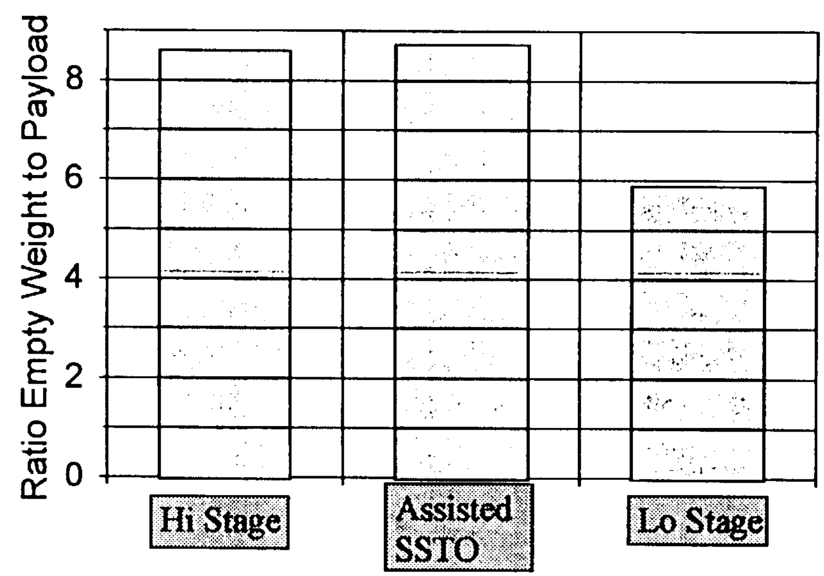
Total Mass Comparison

Rocket Options



Sensitivity Parameter

Rocket Options



Notes:

Hi Stage - SSTO-like booster once around with payload carrier stage

Assisted SSTO - Original assisted horizontal takeoff all-rocket

Lo Stage - SSTO-like vertical takeoff vehicle with booster rockets

Evaluation of TSTO Options

The facing page presents the overall evaluation of the staged options in bullet form. The chart is self-explanatory.

Evaluation of TSTO Options

- Hi stage improves neither liftoff mass or sensitivity parameter (empty weight/payload)
- Hi stage does not eliminate ground accelerator
- Hi stage requires developing two high-speed airframes
- Hi stage not recommended
- Lo stage improves both liftoff mass and sensitivity parameter
- Lo stage eliminates ground accelerator infrastructure while retaining low T/W of orbital vehicle
- Lo stage booster rockets are simple hybrid or pressure-fed units, parachute/sea recovered
- Boosters are inert until LOX loaded, no ground handling problem
- Infrastructure and orbital stage R&D savings estimated to pay for booster R&D and operations
- <u>Lo stage concept is recommended as best "find" of the study.</u>

Most Significant Conclusions

The facing page presents the most significant conclusions for the entire study in bullet form. Since the study was intended to identify new technology paths, the "best buys" were selected on the basis of ability to meet HRST goals and identified new technology options. The assisted HTOHL was also highly rated, but its general features were known before the present study. Ground accelerator technology has been addressed in companion studies.

The new technology to be addressed for the RBCC is clearly the RBCC engine itself, which must be characterized by ground and flight testing.

The lo-stage system requires a very low cost reusable hybrid or pressure-fed technology. The hybrid versus pressure-fed trade may hinge on propellant and turnaround costs. LOX and kerosene are very inexpensive; the loaded cost of HTPB needs to be addressed. Manufacture cost needs to be low, with a reasonable number of reuses and near-zero turnaround cost.

Most Significant Conclusions

Entire Study

- Horizontal takeoff offers significant advantages
  - Less installed thrust; fewer engines; less engine cost
  - Flyable (abort profile) on half of takeoff thrust
  - Less delta V to orbit
  - Less takeoff and empty weight
- Horizontal takeoff requires takeoff assist
  - Ground accelerator, or
  - Recoverable booster rockets, vertical takeoff with pitchover (TSTO Lo-Stage)
  - Boosters offer further weight reduction, eliminates ground accelerator investment
- Kerosene/oxygen rocket based combined cycle may yield improved SSTO performance
  - Horizontal takeoff
  - Mach 6 transition
  - H2/O2 rocket ascent to orbit
  - Requires extensive testing to confirm performance
- "Best Buy" TSTO lo-stage. Runner-up: Kerosene/oxygen RBCC

Appendix: Thrust Coefficients

Energy and Efficiency Considerations

- Thrust capability is limited by the amount of energy added by combustion and by efficiency of conversion of energy to kinetic energy of the exit stream.
- Combustion:
  - Heat of combustion of fuels
  - Maximum fuel/air ratio (stoichiometric limit)
- Efficiency Factors:
  - Inlet compression losses (due to shocks, adverse pressure gradient, friction)
  - Pressure losses of the combustion flow path
  - Combustion inefficiency (usually slight)
  - Expansion losses (usually slight)
  - Limits on expansion pressure ratio due to exit area limitations

Heats of Combustion

- For RBCC operation the lower heat of combustion (no condensation of water) applies
  - Hydrogen: 57.82 kcal/g-mole = 28,680 kcal/kg = 120 MJ/kg.
  - Kerosene: 11,000 kcal/kg (higher); assume $(CH_2)_x$
    - 154,000 kcal/mol CH_2 which yields 1 mol water
    - 540 kcal/kg heat of vaporization; 9720 kcal/mol
    - Kerosene heat is reduced by 9720 kcal/mol
    - Result is 144,000 kcal/mol = 10,300 kcal/kg = 43 MJ/kg

Thrust Coefficient

- Thrust coefficient is thrust/(capture area * q)
- Thrust is V_2*mass flow out - V_1*mass flow in $= V_2(1+r)m_c - V_1 m_c$
- m_c is $\rho A_c V_1$ and q is $\rho V_1^2/2$
- Then $C_T = \{\rho A_c V_1[V_2(1+r) - V_1]\}/(A_c * \rho V_1^2/2)$
  Which simplifies to $2*[V_2/V_1(1+r) - 1]$

Efficiency Factors: Total Energy (1)

Point Calculations at Mach 6: Combustion of Hydrogen

- Air oxygen content 21% by volume
  - Mass of 1 "mol" of air is .21 *32 + .78*28+.01*40 = 28.96, of which .21*32 = 6.72 is oxygen for 22% by mass.
- Combustion of hydrogen:
  - .42 H_2 + .21 O_2 = .42 H_2O
  - Heat is 57.82*1000*0.42 = 24,284 kcal for 1 mol air plus 0.42 mol H_2 = 29.81 kg for 3.486 x 10^6 J/kg of air
- Air energy at Mach 6 (1800 m/s): $V^2/2$ = 1.62 x 10^6 J/kg
- Idealized exit velocity:
  - $\sqrt{2E}$ = 1000*sqrt(2*(1.62+3.486)) = 3195 m/s
  - Velocity change = 3195 - 1800 = 1395 m/s
- Ideal thrust coefficient 2*[(3195/1800)1.028 -1] = 1.65

Efficiency Factors: Total Energy (2)

Combustion of Kerosene

- $0.14\ CH_2 + 0.21\ O_2 = 0.14\ CO_2 + 0.14\ H_2O$
- Kerosene mass is 0.14*14 = 1.96 kg per 28.96 kg air for fuel/air ratio 0.067
- Heat is 10,300 * 1.96 = 20,188 kcal/30.92 kg = 2.733 MJ/kg
- Ideal exit velocity = 2950 m/s

Available Energy (1)

Second Law of Thermodynamics

- Recovery of energy is limited by expansion process
  - Isentropic inlet pressure ratio at Mach 6 is about 1600.
  - Ideal expansion efficiency is $1 - (p_2/p_1)^{(\gamma-1)/\gamma}$ which is $1-(1/1600)^{0.25/1.25} = 1 - 0.23 = 77\%$
- Efficiency applies to expansion process; exit velocity is $\sqrt{h}$ * ideal exit velocity = $\sqrt{0.77}*3195 = 2804$
- Thrust coefficient = 1.2
  - Perfect inlet (isentropic)
  - 2nd law of thermodynamics limit on expansion process

Available Energy (2)

Real Inlets

- Actual inlets operate as compressors with practical efficiency.
  - A typical high-performance inlet efficiency is 60% at Mach 6.
  - Recovery pressure ratio about 350
  - Expansion efficiency about 69%; we may consider friction losses here as well; friction efficiency about 97%.
  - Exit velocity $\sqrt{0.69}\sqrt{0.97}*3195 = 2614$
  - Thrust coefficient = 0.99

Available Energy (3)

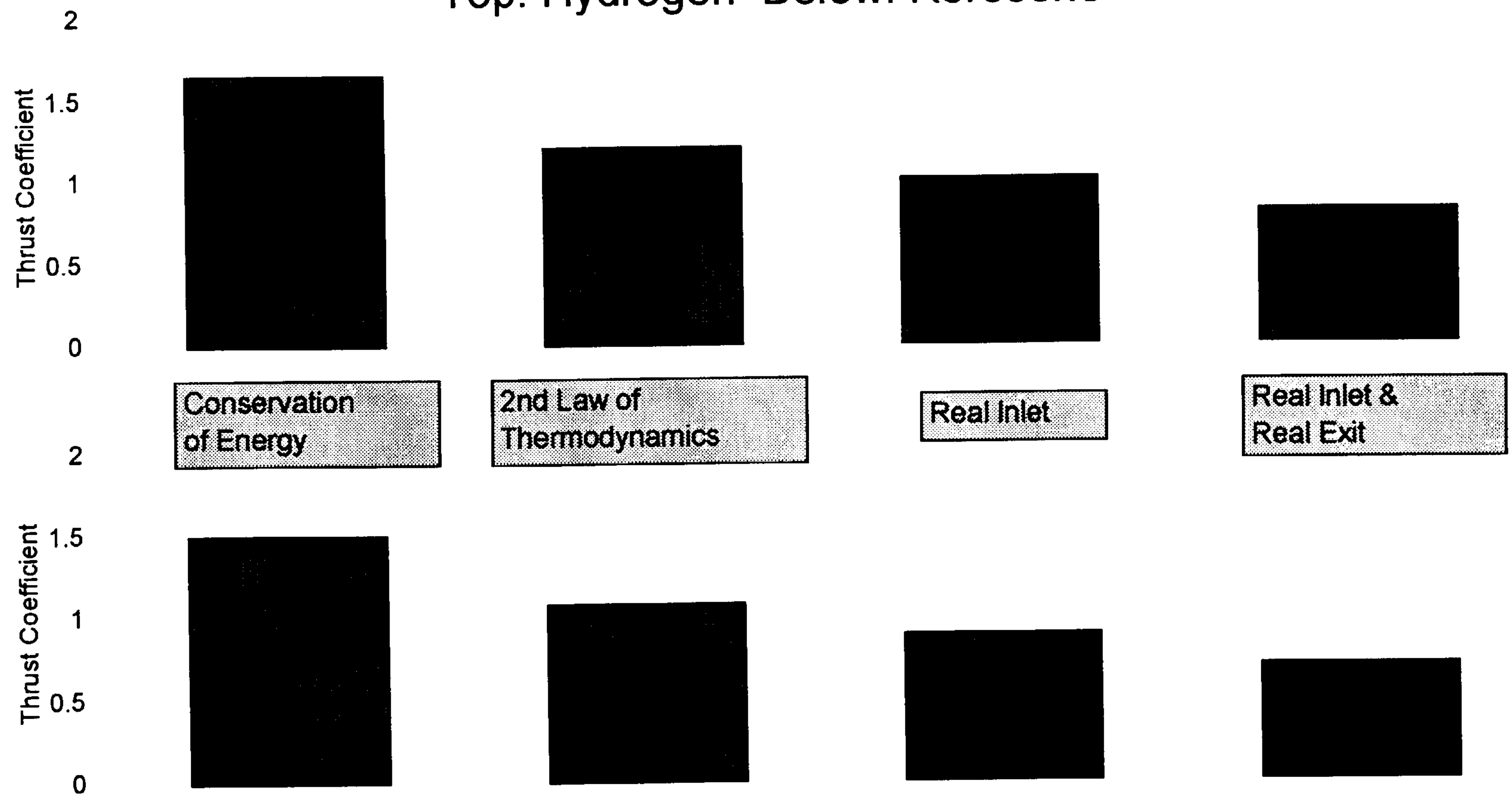
Geometry Limits on Expansion

- Expansion is limited by vehicle geometry which limits the exit area ratio available.
  - At Mach 6, a typical actual pressure ratio is 100.
  - Corresponding efficiency is 60%.
  - Exit velocity is $\sqrt{0.6}\sqrt{0.97}*3195 = 2441$.
  - Thrust coefficient at this velocity would be 0.79
  - Some benefit is obtained from PA force at exit plane; predicted thrust coefficient is about 0.9.
- Predicted thrust coefficient for kerosene about 0.7

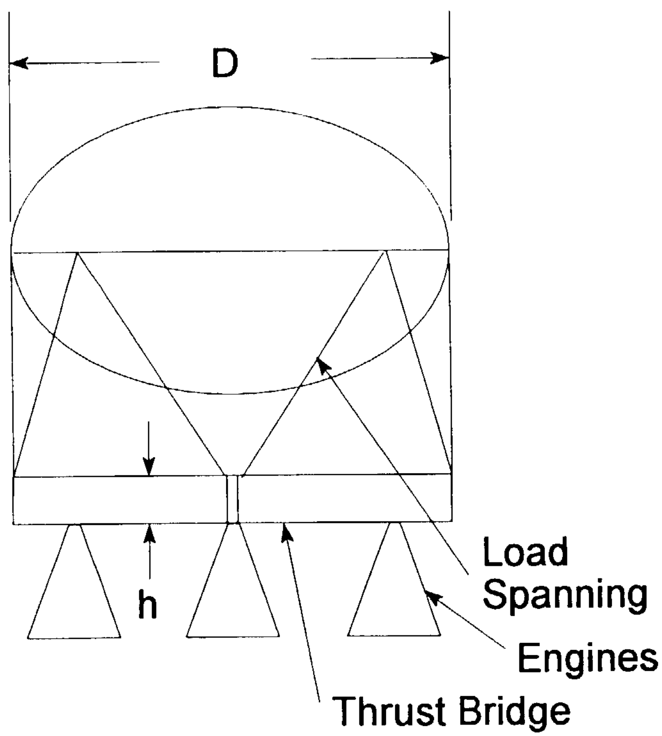
Thrust Coefficients Graphed

Thrust Coefficients at Mach 6

Top: Hydrogen Below: Kerosene



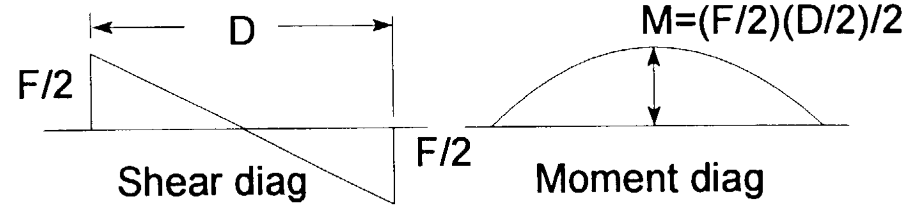
Appendix: Thrust Structure Estimate



For a typical case, ρ = 2850 kg/m2, σ = 200 Mpa, and D = 10 meters.

Thrust Bridge

Evaluate as I-beam.
Assume thrust evenly distributed.



Beam stress eq: $\sigma = Mc/I$

$c = h/2$; $I = 2(X)(h/2)^2$ where X is cross-section area of one cap.

Then $\sigma = \dfrac{(h/2)(F/2)(D/2)/2}{2(X)(h/2)^2} = FD/(8Xh)$

Now, if we let $h = 0.1\ D$, $\sigma = 1.25\ F/X$ and $X = 1.25\ F/\sigma$

Ideal mass is $2XD\rho = 2.5FD\rho/\sigma$ where ρ is material density.

$M_I = (25 \times 2850/2 \times 10^8)F = 0.00035F$ where note that F is in newtons. Then ideal M_I is about 0.3%F if F is in lbf and M_I in lbm. We estimate that ideal mass must be increased by about 20% for shear web and about 50% for practical structural installation.

With adjustment for graphite composite, leads to thrust bridge about 0.3% of thrust. Weight formula is 0.3%(vehicle T/W)(Liftoff Weight)

Small 2nd Stage Weights Methodology

Stage Element	Estimating Approach	Formula & Values
Nose Section	Skin smeared thickness & area	Paraboloidal nose. Surface area $A=\frac{4}{3}\pi r h\left[(1+\frac{r^2}{4h^2})^{3/2}-(\frac{r^2}{4h^2})^{3/2}\right]$ Thickness 2.5 mm GrEp @ 1600 kg/m^3
Body Section	Skin smeared thickness & area plus bending/torsion box	Body skin perimeter around payload bay ~ $3D_p$. Skin perimeter aft ~ $4D_p$. Body length = payload bay length plus propulsion bay length.
Payload Bay, Bay Doors & Payload Support	Internal payload bay skin plus smeared-skin doors plus sill & keel support longerons	Internal skin assume smeared thickness 1.5mm; doors assume 2.5mm GrEp. Longerons sized as Ti-clad GrEp, each longeron capable of supporting half payload mass at 5 g in column loading
Wings	Fighter estimating formula, corrected for composites	See appendix note.
Vertical Tail	(Same)	
Thrust Structure and base heat shield	Base heat shield assumed 3 mm structural support and 1 cm insulator thickness	Heat shield GrEp. Thrust structure weight 1% of thrust; assume T/W = 0.25.
Landing Gear	Based on landing weight with payload	4% of landing weight

TPS	Wetted area except wing upper skin	5 kg/m2 = 1 lb/ft2 + allowance for wing leading edge and nose cap. Assume wing leading edge 5 mm thick by 20 cm wrap, 5 t/m3 density % 35% for attachment and thermal isolation. 6.75 kg per m leading edge length. Assume nose cap 50 kg.
Thermal Control	Assumed build in to subsystems	
Main Propellant Load	Delta V for transfer insertion, circularization, rendezvous and deorbit. LOX-kerosene Isp 350 for main engines.	DV 250 m/s ascent 50 rendezvous & dock 100 orbit maneuver & descent
RCS Propellant Load	Delta V for docking, orbit attitude control, and reentry rate damping. LOX-kerosene Isp 300 for RCS	Assume 100m/s
Propellant Tanks	Aluminum spheres with insulation on LOX. 200 psia provides pressure feed for RCS; main engines pump fed	
Propellant Feed System	Sum of valves and plumbing per schematic	
Main Engines	Assume vehicle T/W 0.25, engine T/W 50, lox-kerosene.	
RCS Thrusters	10 kg per thruster cluster, 4 clusters	

Electrical Power Generation	Fuel cells; assume 3 kWe on orbit 3 days plus 25 kW during 45-minute entry & landing. Total is 235 kWh; assume 1/2 kg/kWh reactants and reactant tanks = 35% reactant mass.	
Electrical Power Distribution	Allowance of 250 kg	
Avionics	"Flat" allowance for boxes, sensors and cabling. Values are installed weights. DFI counted as payload during developmental flights.	Computer 10 kg x 3 GPS 5 kg x 3 IMU 20 kg internally redundant Sun sensors 2 kg x 2 Horizon sensors 2 kg x 2 Rendezvous radar 20 kg Radios 10 kg x 3 Antennas 5 kg x 3 Permanent instrumentation 50 sensors @ 1 kg including RTUs Cabling 25% Total 235 kg
Flight Controls & Actuation	Use fighter aircraft estimating algorithm	106.1(Takeoff Wt, klbm)$^{0.581}$, result in lbm

Appendix Notes

1. Surface Area of a Paraboloid

The paraboloid is revolved around the y axis. The formula for the paraboloid is $y = ax^2$ where x is local radius. If we set the transition from paraboloidal nose to the body cylinder as y=h and x=r, a is h/r^2.

A local differential area is $dA = 2\pi x ds$ where $ds = \sqrt{dx^2+dy^2}$. Integrating for the area, $A = \int dA = \int 2\pi x ds$. Note $dy = 2axdx$ and $ds = \sqrt{dx^2+4a^2x^2dx^2}$, which simplifies to $(1+4a^2x^2)^{1/2}dx$. Then the integral is

$$A = 2\pi \int x\,(1+4a^2x^2)^{1/2}dx$$

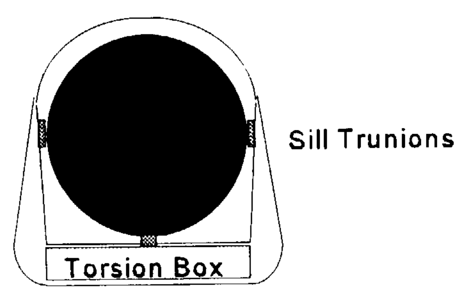
Putting this integral in a standard form,

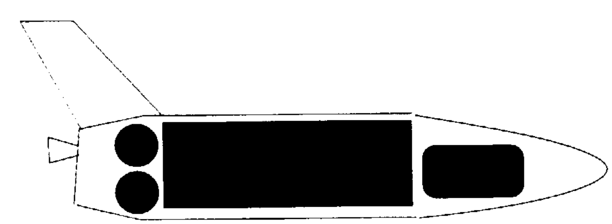
$A = 4\pi a \int x\,(1/4a^2+x^2)^{1/2}dx$, and the integral is evaluated as

$$A = \frac{4}{3}\pi r h\left[(1+\frac{r^2}{4h^2})^{3/2} - (\frac{r^2}{4h^2})^{3/2}\right]$$

2. Body Bending/Torsion Box

Bending in pitch is absorbed by the torsion box acting together with the sill trunnions. Body bending in yaw and body torsion is absorbed by the torsion box. Since the stage is tandem-mounted





to the booster, torsion loads will be small. The torsion box is sized by bending loads. It is assumed that the effective c.g. of the vehicle is 8 m forward of the aft bulkhead of the payload bay and that the mass acting on the torsion box in pitch and yaw bending is 25 t. at 3 g. The bending moment is roughly 6 x 10^6 N-m. The cross-section of the ends of the torsion box is calculated to be 0.003 m^2. Assuming GrEp at density 1600, the mass of the ends is 55 kg each for 110 kg total. The top & bottom of the box are assumed about twice this mass giving a total for the box of 330 kg. Body skin thickness assumed 3mm smeared average, 3 D_p wrap around payload bay and 4 D_p aft of payload bay. (D_p is 5 m.) The body mass around the payload bay is 3*5*11*0.003*1600 = 792 kg, and aft of the payload bay about 4*5*5*0.003*1600 = 480 kg. An allowance of 100 kg is made for secondary structure supporting subsystems. Frames are estimated total 320 kg for 11 frames in the payload bay section (see spread sheet).

The body total is 330+792+480+100 = 1702 kg. This does not include thrust structure, or the payload bay itself, i.e. internal skin, forward & aft bulkheads, trunnions, and payload bay doors.

3. Wing Weight Equations

The following equations are given for wing weights:

USAF fighter aircraft

$$W_W = 3.08\left(K_{PIV} N \frac{W_{TO}}{t/c}\left[\left[\tan\Delta_{LE} - \frac{2(1-\lambda)}{A(1+\lambda)}\right]^2 + 1.0\right] x 10^{-6}\right)^{0.593} [(1+\lambda)A]^{0.89} S_W^{0.741}$$

K_{PIV} is a variable geometry correction factor which is 1.0 for this application.
N is ultimate normal load factor
W_{TO} is takeoff (loaded) weight in lb.
t/c is thickness ratio

Δ_{LE} is leading edge sweep
λ is taper ratio
A is aspect ratio
S_W is wing area in ft^2.

<u>USN fighter aircraft</u>

Uses the same equation except the lead constant is 19.29 and the fractional exponents are 0.464, 0.7, and 0.58 respectively.

Using trial inputs of N=4.5; $W_{TO} = 75{,}000$; $t/c = 0.12$; $\Delta_{LE} = 60$; $\lambda = 0.1$; $A = 2.3$; and $S_W = 1000$: the USAF equation yields about 2750 lb and the USN equation about 4000 lb. The USN equation is deemed more credible since it is about 4 lb per square foot.

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This study is an extention of a previous effort by the Principal Investigator to develop baseline data to support comparative analysis of Highly Reusable Space Transportation (HRST) concepts. The analyses presented herin develop baseline data bases for two two-stage-to-orbit (TSTO) concepts: 1)Assisted horizontal take-off all rocket (assisted HTOHL) and 2)Assisted vertical take-off rocket based combined cycle (RBCC). The study objectives were to: 1)Provide configuration definitions and illustrations for assisted HTOHL and assisted RBCC. 2)Develop a rationalization approach and compare these concepts with the HRST reference. and 3)Analyze TSTO configurations which try to maintain SSTO benefits while reducing inert weight sensitivity.

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