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In Defense of External Tanks

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External tanks can reduce the specific cost of reusable, partially reusable, and even certain types of fully expendable launch vehicles. Adding external tanks to a single-stage-to-orbit launch vehicle reduces development cost by easing the difficulty of an otherwise very challenging design problem. At flight rates that will be likely in the near future, this reduction in development cost more than makes up for the increased hardware and operation costs associated with the external tank. Using identical stages, such as the bimese and trimese designs, can reduce the development cost of a new launch vehicle by allowing an entire vehicle to be developed at the cost of developing only one stage, but they suffer from suboptimal staging velocities. Adding an external tank to these identical stage designs can improve performance by allowing better staging velocities for the main vehicle components. Other forms of expendable propellant tanks can allow limited reusability to be introduced to lower stages of multi-stage launch vehicles when a fully reusable design is not economically justified.

Nomenclature

a	=	amortization factor
c_F	=	specific cost of facilities and other non-vehicle, non-recurring items
c_H	=	cost of vehicle hardware per unit weight
c_L	=	cost of labor per man-hour, including overhead
c_P	=	cost of propellant per unit weight
c_{PL}	=	cost of payload per unit weight
c_T	=	specific launch cost = C_{TOT}/M_{PL}
C_{TOT}	=	total launch cost
f	=	fraction of launch vehicle expended in one mission
GLOW	=	gross liftoff weight
Isp	=	specific impulse
L	=	labor intensity
LEO	=	low Earth orbit
M_P	=	vehicle propellant mass
M_{PL}	=	vehicle payload mass
M_S	=	vehicle structure mass
P_{FAIL}	=	probability of a mission failure
q	=	propellant-structure mass ratio = M_P/M_S
r	=	structure-payload mass ratio = M_S/M_{PL}
r_D	=	developed structure-payload mass ratio
T/W	=	engine thrust-to-weight ratio
η	=	propellant mass fraction = $M_P/(M_P+M_S)$
η_T	=	tankage propellant mass fraction
λ	=	structure cost = C_{TOT}/M_S
λ_D	=	vehicle development structure cost
λ_H	=	vehicle hardware structure cost
λ_K	=	risk structure cost

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λ_L = flight operations structure cost
λ_P = propellant structure cost
λ_R = recurring structure cost

I. Introduction

The addition of external drop tanks to the P-51 Mustang gave it sufficient range to be an effective fighter escort for Allied bombers and helped win World War II. External tanks (ETs) are still used today to extend the range of military aircraft. Studies for an economical launch vehicle to replace the Saturn V regularly made use of external tanks with reusable or partially reusable vehicles, and a large external tank is one of the major components of the Space Shuttle system that eventually resulted. The use of expendable tanks to improve performance and reduce cost continued in launch vehicle studies like the National Launch System and Boeing's original EELV proposal (although their physical location sometimes made the term "external" inappropriate). External tanks are noticeably absent from current popular launch vehicle proposals. This paper will demonstrate that the external tank still has a useful role to play in the next generation of reusable or partially reusable launch vehicles, and that in special circumstances it can even reduce the cost of fully expendable launch vehicles.

II. Cost Model

The cost model used in this analysis is based on the equation:

$$c_T = r\lambda \tag{1}$$

where

c_T = specific launch cost = total launch cost / payload mass
r = structure-payload mass ratio = structural mass / payload mass
λ = structure cost = total launch cost / structural mass.

This cost model is similar to previously published models, such as those by Griffin[1], Claybaugh[2], Kalitventzeff[3], Carton[4], and Taylor[5]. It has the benefit of dividing the problem of launch cost into two variables, r and λ, that reflect different areas of study. The value of the variable r is driven primarily by technology and the physics of the orbit desired. The calculation of r can be done from historical trends, simple rocket equation analysis, or detailed engineering studies depending on the level of accuracy desired. Sample values of r for several current launch vehicles are shown in Table 1. The value of the variable structure cost, λ, is driven by economics and program management. Determining the value of structure costs will be discussed in below.

Table 1. Values of *r*

Vehicle	r (to LEO)
Atlas V 400	2
Proton M	2.2
Ariane 5	5.2
Space Shuttle	12

A more detailed analysis of cost can be achieved by rewriting the cost model to consider recurring and amortized non-recurring cost separately:

$$c_T = r\lambda_R + r_D \frac{\lambda_D}{a} + c_F \tag{2}$$

where λ_R is the recurring launch cost per unit of vehicle structure, r_D is the developed structure-payload mass ratio, λ_D is the cost of developing the launch vehicle per unit mass of structure, a is an amortization factor that determines how much of the non-recurring development costs are charged to each launch, and c_F is the specific cost of new launch facilities and other non-vehicle-related, non-recurring costs. The amortization factor, a, is typically proportional to the launch vehicle flight rate. The developed structure-payload mass ratio, r_D, is identical to the structure-payload mass ratio, r, for completely new launch vehicles with non-identical stages. The first term on the right side of Eq. (2) is the recurring launch costs. The second term is the amortized vehicle-related development costs. The third term covers the amortized, non-vehicle related, non-recurring costs (such as the cost of constructing a new launch facility.) The "back of the envelope" comparisons done in this paper assume that all the launch vehicle concepts analyzed make use of existing space launch infrastructure, and so the c_F term will be omitted in future discussion. Omitting c_F allows Eq. (2) to be rewritten as:

$$c_T = r\lambda_R + r_D \frac{\lambda_D}{a} \tag{3}$$

Recurring structure costs can be further broken down into the categories of vehicle hardware cost, operations cost, risk cost, and propellant cost.

$$\lambda_R = \lambda_H + \lambda_L + \lambda_K + \lambda_P \tag{4}$$

Vehicle hardware cost per unit of structure mass is[2] the product of the fraction of the vehicle expended in a launch, f, and the cost of vehicle hardware, c_H.

$$\lambda_H = fc_H \tag{5}$$

For a fully expendable launch system f would have a value of 1. For a reusable system f would represent the fraction of the vehicle that is worn out during each mission. Some researchers[3,4,6] prefer to separate the hardware cost into more detailed categories such as "engine" and "tankage" costs. This technique will be used on some of the concepts analyzed in this paper.

Operations cost per unit of structure mass is[2] the product of the cost of labor including overhead, c_L, and a labor intensity parameter, L.

$$\lambda_L = Lc_L \tag{5}$$

For expendable launch systems L is defined as the man-hours of launch operation labor divided by the structure mass of the vehicle. For a reusable launch system L would also need to include recovery and refurbishment labor.

It is difficult to determine the cost of risk per unit of vehicle structure because launch vehicles have many different failure modes, each with their own potential cost. For private launches the price of insurance must also be a consideration. For the simple "back of the envelope" analysis done in this paper, the cost of risk is assumed to be:

$$\lambda_K \approx P_{FAIL}\left[(c_{PL}/r) + (1-f)c_H\right] \tag{6}$$

where P_{FAIL} is the probability of a mission failure resulting in vehicle and payload loss, and c_{PL} is the value per unit mass of the payload. This equation will likely produce a low estimate of the risk cost because it only estimates the direct cost of the lost vehicle and payload. A real mission failure would also incur indirect costs resulting from schedule delays, public relations problems, accident investigation, and similar factors.

The cost of propellant per unit of structure mass is a product of the propellant-structure mass ratio, q, and the cost per unit mass of the propellants, c_P.

$$\lambda_P = qc_P \tag{7}$$

where

$$q = \eta/(1-\eta) \tag{8}$$

A computer spreadsheet named Rocketcost and based on this cost model is available free on Jupiter Research and Development's website at http://www.jupiter-measurement.com/research/rocketcost.xls.

III. Current State of Launch Costs

Table 2 lists values typical of the economic characteristics for current generation space launch vehicles. The range of values for c_H is from Worden[7], the range of values for L is from Claybaugh[2] and Griffin[8], and the range of P_{FAIL} is from information presented by Chang[9]. The value for c_P assumes a liquid fuel vehicle. Rocket propellant prices are available from the Defense Energy Support Center, and are provided online at http://www.desc.dla.mil/Static/ProductsAndServices.asp under the heading "missile fuels." The range of values for λ_D is from Claybaugh[2] supplemented by information given by Isakowitz[10] and at the website http://www.astronautix.com. In order for the data in Table 2 to be useful in calculating the current state of launch costs, the amortization factor, a, and values for the launch vehicle parameters r, r_D, and η must also be known. For this analysis a will be assumed to be 27. This value for a is based on a 10 year payback of the non-recurring costs with a 4 year development program followed by 27 flights over the next 6 years, a flight rate typical of current American space launch vehicles[8], and neglecting interest or inflation. Wertz[11] presents a description of the effect of interest and inflation on launch costs that may be useful to researchers wanting a more detailed treatment of amortization. For the current launch cost estimate described in this section the value of structure-payload mass ratio, r, will be assumed equal to 2, r_D will be assumed equal to r, and η will be assumed equal to 0.9. These values are not conservative, in order to represent the best case for current space launch costs.

Table 2. Current Generation Launch Vehicle Economic Characteristics

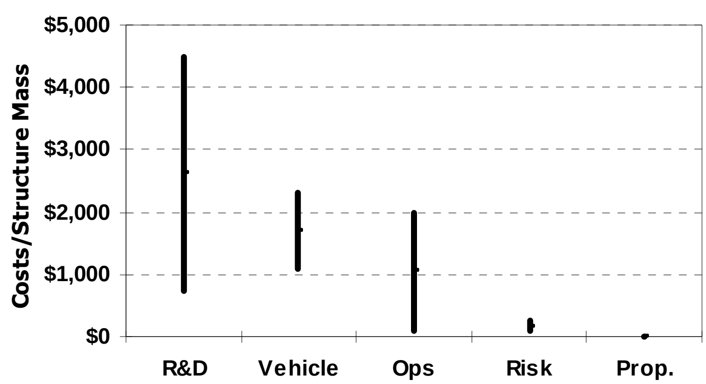
Characteristic	Values
f	1
c_H	\$1100/lb.to \$2300/lb.
L	1 to 20
c_L	\$100/hr
P_{FAIL}	2% to 5%
c_{PL}	\$10,000/lb.
c_P	\$0.1/lb. to \$0.25/lb.
λ_D	\$20,000/lb.to \$120,000/lb.

Table 3 gives the estimated current state of launch vehicle structure costs from this cost model using the above information and assumptions. This estimated cost information is also presented graphically in Figure 1.

Table 3. Estimate of Current Space Launch Structure Costs to LEO

Cost	Value
Amortized Vehicle Development	\$750/lb. to \$4,500/lb.
Vehicle Hardware	\$1,100/ lb. to \$2,300/lb.
Operations	\$100/lb. to \$2,000/lb.
Risk	\$100/lb. to \$250/lb.
Propellant	\$1/lb to \$2.5/lb.

Figure 1. Estimate of Current Space Launch Structure Costs to LEO



The costs given in Table 3 and Fig. 1 are in dollars per pound of vehicle structure mass. To convert this to launch cost per pound of payload mass you must multiply these costs by the value of r, which in this example is 2. The resulting estimate of current space launch costs to LEO (ideal ΔV = 30,000 ft/s) suggests that the two largest components of space launch cost are the amortized non-recurring cost, which in this analysis is limited to R&D costs, and the cost of vehicle hardware. Flight operation costs may not even be the third biggest contributor to space launch costs with the current generation of launch technology. Using the best current practices the cost of operations and the cost of risk were both estimated at approximately \$100/lb. As discussed above, however, the cost model used probably produces a low estimate of the cost of risk because it only includes direct costs. Once indirect costs, such as accident investigations and schedule delays, are considered the third largest contributor to cargo space launch costs would likely be the cost of risk, with the cost of operations close behind as the fourth largest cost component. For manned launches, where the cost of failure is much higher than for cargo, the cost of risk would certainly overshadow the current state of operations costs. The cost of propellant is insignificant and can be ignored in a "back of the envelope" analysis.

IV. External Tanks to Reduce Specific Cost

The results in Table 3 and Fig. 1 contradict the "conventional wisdom" that the two biggest components of space launch cost are vehicle hardware and flight operations. Many proposals for developing an economical space launch vehicle attempt to lower the cost of space access by having a fully reusable vehicle with "aircraft-like operations." These advanced concepts are technically challenging, and therefore significantly increase the amortized development cost of the system. These vehicles seek to reduce the second and third biggest component to space launch cost at the expense of increasing the biggest component. Cheap access to space cannot be achieved in this fashion. Even if all other costs were eliminated completely, the estimates in Table 3 and Fig. 1 show that the amortized development costs alone are enough to make the specific launch cost with a typical new launch vehicle exceed $1000 per pound of payload to LEO. Some way must be found to reduce amortized development costs if cheap space access is to be achieved.

One way to reduce amortized development costs is to increase the amortization factor, a. Typically this requires increasing the vehicle's flight rate, although it can also be achieved with multi-use technologies that spread the amortization over other applications. Considerable work is being done to find missions for launch vehicles, such as space tourism or fast package delivery, that will allow the high flight rates needed to keep amortized development costs low for technically ambitious vehicles.

Another way to reduce amortized development costs is to reduce developed structure-payload mass ratio, r_D. This can be accomplished by having an evolutionary vehicle where only part of the total system needs to be developed and the rest can reuse previously developed hardware. It can also be accomplished by reducing the vehicle's overall structure-payload mass ratio, although this may result in an even more expensive vehicle if the development structure cost is increased too much by the effort put into weight savings. Developed structure-payload mass ratio for an entirely new vehicle can be reduced without reducing the vehicle's structure-payload mass ratio, by using identical stages or stage components so that hardware can be developed once and used in multiple places on the system. An example of this would be to use many identical solid rocket motors as lower stage boosters. The solid rocket motor only has to be developed once regardless of how many are used.

A third way to reduce amortized development costs is to reduce the development structure cost, λ_D. Good program and organizational management can result in large development structure cost savings[12], but such non-technical approaches are outside the scope of this paper. Technical approaches to reducing development structure cost start with designing the system for economic performance from the beginning of the program. Cost optimization affects configuration and sizing in the conceptual design stage, and cannot be added as an afterthought. Reducing the technical difficulty of the development effort reduces the development structure cost. This requires selecting a launch vehicle configuration that does not have overly ambitious performance requirements relative to the number of paying flights that the development cost can be amortized over.

Putting expendable external tanks on a reusable launch vehicle is a way to potentially improve the launch system's economics by accepting increased hardware cost to gain a greater reduction in amortized development cost. Using external tanks allows the vehicle to gain some of the effects of staging by discarding unneeded tankage without incurring the high cost of throwing away more expensive components like engines and avionics. It also reduces development costs even further because the recovery system does not need to be sized to bring back the mass of the expendable tanks. Reusable external tanks could be utilized if the cost of reusing them and the cost of the reduced system performance caused by the increased weight of reusable tanks is less then the cost of manufacturing new tanks for each launch. Reusable external tanks would still be a compromise on operational costs since they would add another component to be integrated with the launch vehicle, but they would not be as heavy or have as many system interfaces as a complete stage would have.

V. Single-Stage-to-Orbit and External Tanks

Single-Stage-to-Orbit (SSTO) vehicles are very good examples of the a technically demanding launch vehicle concept that produces a high development cost in an attempt to chase low hardware and operations costs. SSTOs require both high engine performance and very low structural mass fractions, resulting in designs that are very susceptible to weight growth during development. To keep weight growth from spiraling out of control a lot of engineering effort is required to find every performance advantage possible. Design problems that arise must often be solved by lengthy investigations to eliminate the root causes instead of simply increasing the factor-of-safety on any questionable components. A fully reusable SSTO with airplane-like operation looks great when examined from a standpoint of recurring costs but might be unaffordable when amortized development costs are included. Adding an external tank to an SSTO vehicle could greatly reduce the technical difficulty of the vehicle development effort by granting some of the benefits of staging. This would come at the expense of some increase in recurring costs, but not as much as adding an additional full stage. The SSTO+ET configuration might also be called a "stage and a half, drop tanks with reusable core," a "reusable integrated launch and re-entry vehicle (ILRV) with tip tanks," or a "reusable all system (RAS) with expendable tanks."

In order to illustrate the benefits of altering the SSTO configuration to include an external tank (or tanks), the economic model described in Section 2 will be used to produce "back of the envelope" cost estimates of a launch system comprised of an SSTO-like reusable core with an expendable external tank. When launched, the rocket motors on the core vehicle are fueled entirely from the external tank. Once the external tank fuel is expended, the tank is discarded and the core vehicle proceeds to orbit on internally stored fuel.

Using the Rocketcost spreadsheet, the cost and mass properties of a baseline vehicle were calculated for various external tank separation velocities. The Rocketcost input characteristics for the reusable core and expendable external tank of this baseline SSTO+ET vehicle are listed in Table 4. The performance and propellant cost for this vehicle were selected to approximate a next-generation LOX/LH_2 launch vehicle. The hardware and development costs for the reusable core are from the high end of the range for existing launch vehicles, while the ones for the expendable external tank are from the lower end of the range of existing launch vehicles. Probability of mission failure is lower than for current launch vehicles. The labor intensity for the vehicle with no external tank (ET separation velocity = 0) is half of present best-case launch vehicles, and

Table 4. Baseline SSTO+ET Vehicle Characteristics

Characteristic	Values
f_{core}	0.02
f_{ET}	1
$c_{H,\,core}$	\$2500/lb.
$c_{H,\,ET}$	\$1000/lb.
L	0.75 hr/lb. (with ET) 0.5 hr/lb. (without ET)
c_L	\$100/hr
P_{FAIL}	0.5%
c_{PL}	\$10,000/lb.
c_P	\$0.24/lb.
$\lambda_{D,\,core}$	\$100,000/lb.
$\lambda_{D,\,ET}$	\$25,000/lb.
a	27
Isp	450 s
T/W	50
η_T	0.96

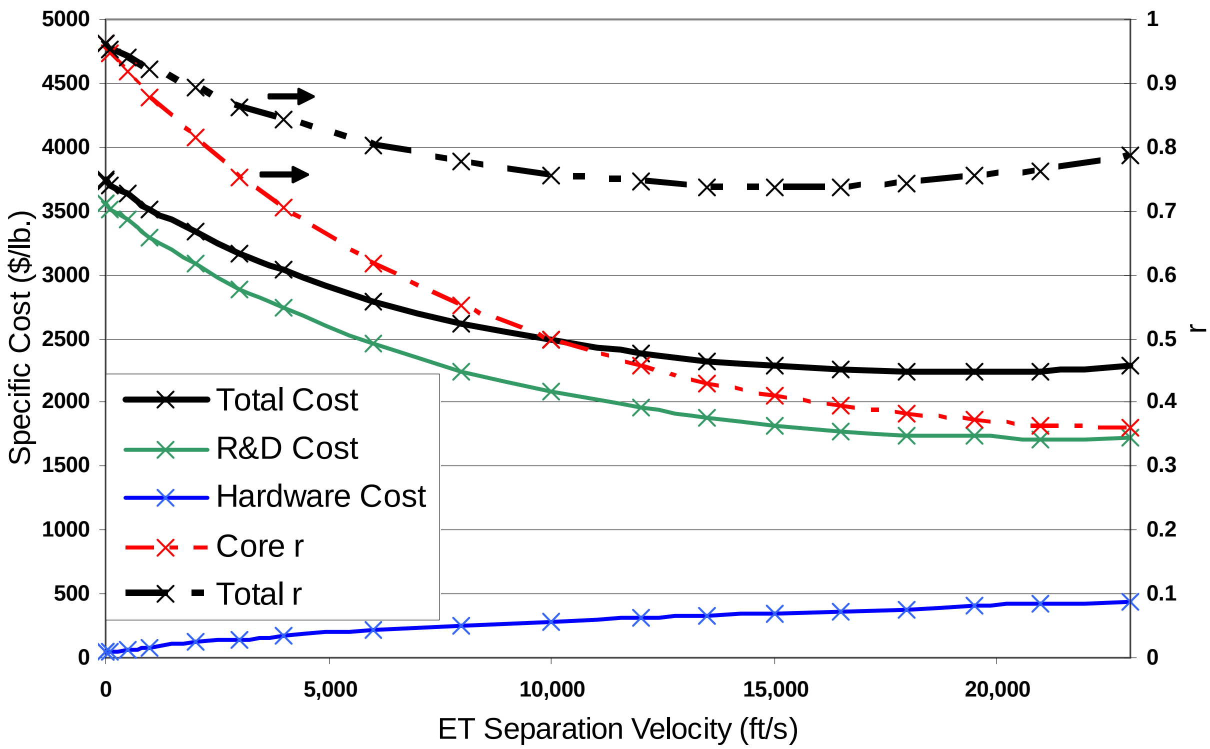


Figure 2. SSTO+ET Baseline Specific Cost and Mass Ratio Estimates vs. ET Separation Velocity

adding an external tank is assumed to increase this by 50%. The mass of the vehicle was calculated assuming a 10,000 lb. payload but because the sizing and costing equations used were linear, the specific cost and mass ratio results would be the same for any payload size. The vehicle mass was calculated by separating the structure mass into tankage-related mass and engine-related mass. Tankage-related mass was calculated from the amount of propellant and the tankage propellant mass fraction, η_T. Engine-related mass was calculated using the GLOW and engine trust-to-weight ratio, *T/W*, to achieve a vehicle thrust-to-weight ratio of 1.25 at liftoff.

The results of the Rocketcost analysis of the baseline SSTO+ET vehicle are plotted in Fig. 2. The structure-payload mass ratio, *r*, of the launch system was also calculated and is represented by the dotted black line (read on the secondary Y axis.) The structure-payload mass ratio of the reusable core vehicle is shown as the dotted red line (also read on the secondary Y axis), and the gap between the two lines shows the size of the external tank. The black solid line represents the total specific cost for the launch system in dollars per pound of payload delivered to LEO, and shows that the minimum cost configuration occurs when the propellant in the external tank provides approximately 18,000 ft/s of ideal ΔV before it is dropped. This minimum cost occurs at a higher external tank separation velocity than the minimum structure-payload mass ratio, which occurs at approximately 15,000 ft/s of ideal ΔV. This happens because the hardware structure cost and amortized development structure cost of the external tank are lower than those of the core vehicle. The blue line represents the hardware specific cost, and shows the increasing hardware cost as the expendable external tank becomes larger. The green line represents the amortized development specific cost and shows how the use of external tanks can greatly reduce the amortized development cost of the launch system. Figure 2 clearly illustrates that adding an external tank can significantly reduce the cost and size of the baseline SSTO-like launch vehicle.

Table 5. Low and High Performance SSTO+ET Vehicle Characteristics

Characteristic	Low Performance Vehicle	High Performance Vehicle
f_{core}	0.025	0.001
f_{ET}	1	1
$c_{H,\,core}$	\$2500/lb.	\$2500/lb.
$c_{H,\,ET}$	\$1000/lb.	\$1000/lb. (monthly flights) \$300/lb. (weekly flights)
L	1 hr/lb. (w/o ET) 1.5 hr/lb. (w/ ET)	0.01 hr/lb. (w/o ET) 0.015 hr/lb (w/ ET)
c_L	\$100/hr	\$100/hr
P_{FAIL}	1%	0.1%
c_{PL}	\$10,000/lb.	\$10,000/lb.
c_P	\$0.24/lb.	\$0.24/lb.
$\lambda_{D,\,core}$	\$100,000/lb.	\$100,000/lb.
$\lambda_{D,\,ET}$	\$25,000/lb.	\$25,000/lb.
a	27	72 (monthly flights) 312 (weekly flights)
Isp	430 s	460 s
T/W	40	60
η_T	0.95	0.965

To demonstrate that the cost savings achieved by adding external tanks on SSTO concepts occurs over a broad range of design conditions, two additional designs were analyzed using the cost model described in Section 2 with the Rocketcost spreadsheet. The results of these calculations are graphed in Fig. 3. The first additional design is a lower performance vehicle. The characteristics for the reusable core and expendable external tank of this lower performance system are listed in Table 5. The structure-payload mass ratio for this lower performance system is represented in Fig. 3 by the dotted red line (read on the secondary Y axis). The specific cost for the lower performance design is represented by the solid red line. The external tank provides an even greater specific cost savings with this vehicle. This is not unexpected, since the lower performance vehicle is generally more expensive and so there are more potential savings to be had.

The baseline vehicle's specific cost and structure-payload mass ratio are shown as the black solid line and black dotted line (read on the secondary Y axis), respectively. The second additional design is a higher performance vehicle that is flown more frequently. The characteristics for the reusable core and expendable external tank of this higher performance vehicle are listed in Table 5. These values are estimated characteristics of an ambitious next-generation vehicle with a low probability of failure and a labor intensity similar to those of supersonic aircraft. The dotted blue line (read on the secondary Y axis) shows the structure-payload mass ratio of the higher performance system. The solid blue line represents the specific cost of this higher performance system with an amortization factor of 72. This amortization factor would occur with a ten year payback period comprising a four year development program followed by monthly flights over six years with no interest or inflation[†]. Even flying monthly with a higher performance vehicle, this design still shows that an SSTO-like vehicle with an expendable external tank has a lower specific cost than a pure SSTO launch system. The minimum cost configuration for this higher flight rate and higher tech system does come with a smaller external tank than the baseline vehicle that separates at

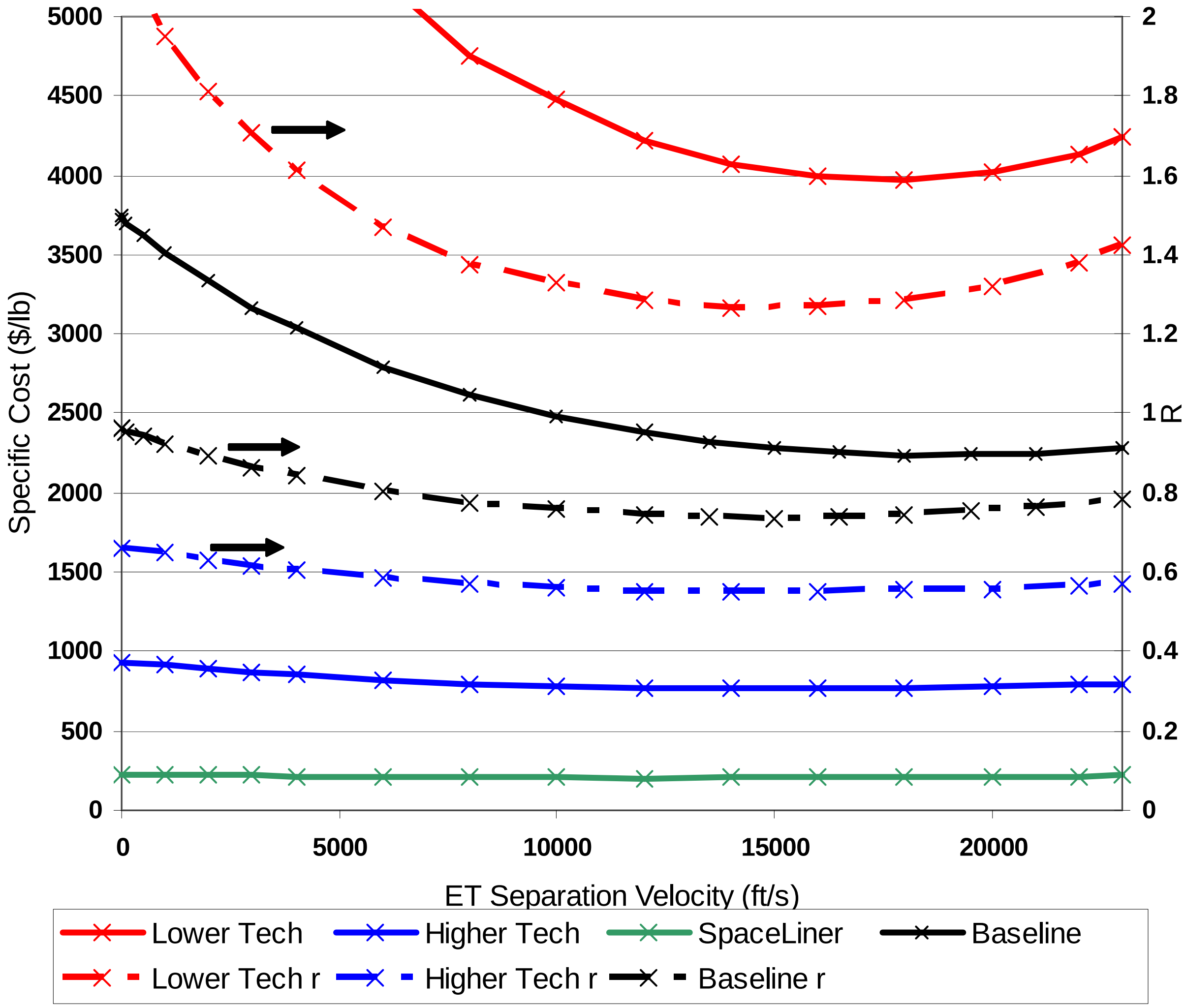


Figure 3. Specific Cost and Mass Ratio Estimates vs. ET Separation Velocity for SSTO +ET Launch Vehicles of Various Performance Levels and Flight Rates

an ideal ΔV of about 14,000 ft/s.

As amortization factor is increased, the minimum cost configuration of an SSTO+ET occurs with a smaller and smaller external tank. Weekly flights, over six years and neglecting interest or inflation, would give an amortization factor of 312. With that amortization the higher performance vehicle has its lowest cost configuration as a pure SSTO with no external tank (which occurs at an external tank separation velocity of 0 ft/s) when the hardware structure cost of the external tank is still $1000/lb. Weekly flight rates would, however, require higher production quantities and higher production rates for the expendable external tank, and that would reduce the manufacturing

[†] Neglecting interest and inflation is not realistic for private space launchers. Private space launchers would, however, hope to achieve an increasing number of launchers as the system matures. This analysis could also be described as assuming the initial launch rate is one flight per month and that the increase in additional flights as the system matures is enough to offset the net effects of interest and inflation.

costs of the tanks considerably[13]. The solid green line in Figure 3 represents the specific cost of a "weekly spaceliner" system composed of the higher performance vehicle with an amortization factor of 312 and the external tank hardware structure cost reduced to $300/lb. Once the lower production costs of external tanks are considered, even the "spaceliner" has a lower specific cost when expendable external tanks are added to the reusable SSTO-like core vehicle. The minimum cost for the "spaceliner" is about $205 per pound of payload delivered to LEO and occurs at an external tank separation velocity of about 12,000 ft/s. This is in contrast to the estimated cost of $228 per pound without any external tanks. Perhaps space launch flight rates will eventually be high enough that a pure SSTO vehicle will provide lower space launch costs than one with an expendable external tank, but that condition seems unlikely to arrive in time to affect the design of next-generation space launch vehicles.

Using expendable external tanks requires dropping the empty tanks downrange after launch. If dropping vehicle components downrange is acceptable, then that also opens up the possibility of using solid rocket boosters (SRBs) to augment the core vehicle instead of using external tanks. The key advantage of using SRBs to reduce the technical difficulty of the reusable core is that they provide thrust in addition to propellant. When using external tanks, the engines of the core vehicle must be large enough to lift both the core vehicle and the full external tanks. Dropping the tanks reduces the tankage structure that must be carried to orbit but the extra engine mass needed to lift the tanks at takeoff are carried all the way to orbit. With a design where SRBs are providing thrust, the core vehicle engines can be sized for the vehicle weight after the SRBs burnout and separate instead of being sized for the vehicle weight at liftoff. Key disadvantages of SRBs relative to external tanks are that solid propellant is typically much more expensive than liquid propellant and that the solid rocket motors will typically have a lower *Isp* than the main engines, where the propellant from the external tanks will be burned.

In order to compare SRBs to external tanks as a method of augmenting an SSTO-like reusable core, an estimate was made of the optimum SRB size for the baseline vehicle. The economic and performance estimates used for the core vehicle in this analysis are the same as those given in Table 4. The engine-related structure mass in this analysis was sized to achieve a vehicle thrust-to-weight ratio of 1.25 after SRB separation. The SRB's were assumed to have an *Isp* of 274 s, a propellant mass fraction of 90%, a propellant cost of $100/lb. and a hardware structure cost of $500/lb. Figure 4 shows the specific cost and structure-payload mass fraction of this SSTO+SRB launch system for varying sizes of SRB. The SRB size is indicated by the ΔV attributable to the SRBs (not the velocity at SRB burnout). The solid blue line represents the specific cost of the system in dollars per pound of payload delivered to LEO assuming that the SRBs must be developed at a development structure cost of $20,000/lb. The solid red line represents the specific cost of the system assuming that there are no development costs associated with the SRBs, and shows a minimum cost of about $3,100/lb at an SRB ΔV contribution of 5000 ft/s. The dashed green line represents the structure-payload mass ratio of the launch system.

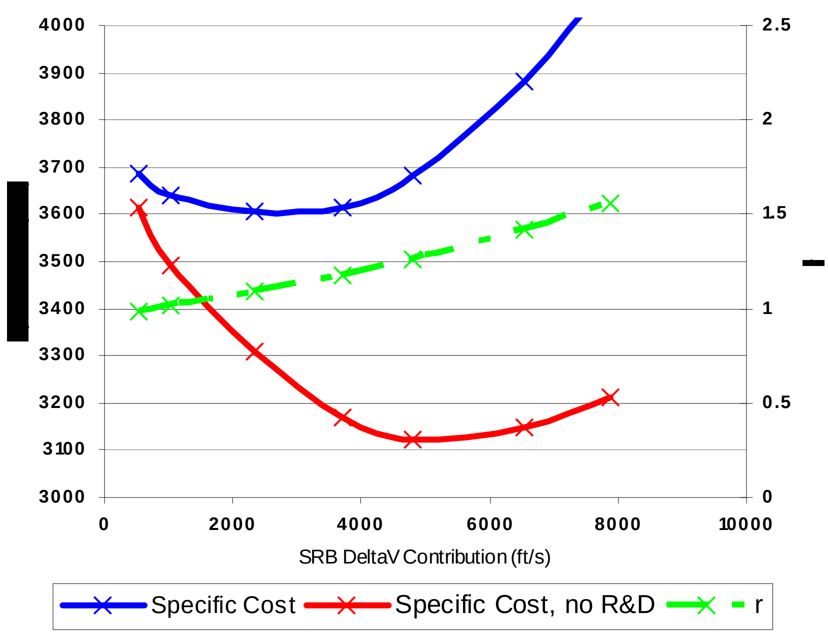


Figure 4. SSTO+SRB Specific Cost and Structure-Payload Mass Ratio Estimates vs. SRB Size

One of the major differences between the SSTO+SRB and SSTO+ET configurations are the size of the engines required on the reusable core vehicle. Because the external tanks provide no thrust on their own, the SSTO+ET configuration requires larger engines on the reusable core to achieve sufficient vehicle thrust-to-weight on liftoff. This extra engine mass must be carried all the way into orbit and retrieved. The performance of the core's engines become a key factor in determining the relative merits of SSTO+SRB and SSTO+ET configurations. With sufficiently high thrust-to-weight ratio engines, the extra engine mass needed to lift the external tank becomes insignificant and there is almost no penalty to using external tanks instead of booster stages that can provide their own thrust. To determine the level of core engine performance needed to make the SSTO+ET concept more economical than the SSTO+SRB concept, the specific cost of the both configurations using the baseline vehicle was calculated for a range of different engine thrust-to-weight ratios, *T/W*.

In comparing the SSTO+SRB configuration to the SSTO+ET configuration, the SRBs were given no development structure cost, based on the assumption that an existing solid rocket motor design would be used. For the initial comparison, the SRB and the ET were both sized to provide 5,000 ft/s of additional ΔV to the reusable core. This is around the minimum cost point of the SRB configuration, so it provides the best case for using SRBs instead of external tanks. Because the core vehicle's engines are fueled only from the external tank during the initial

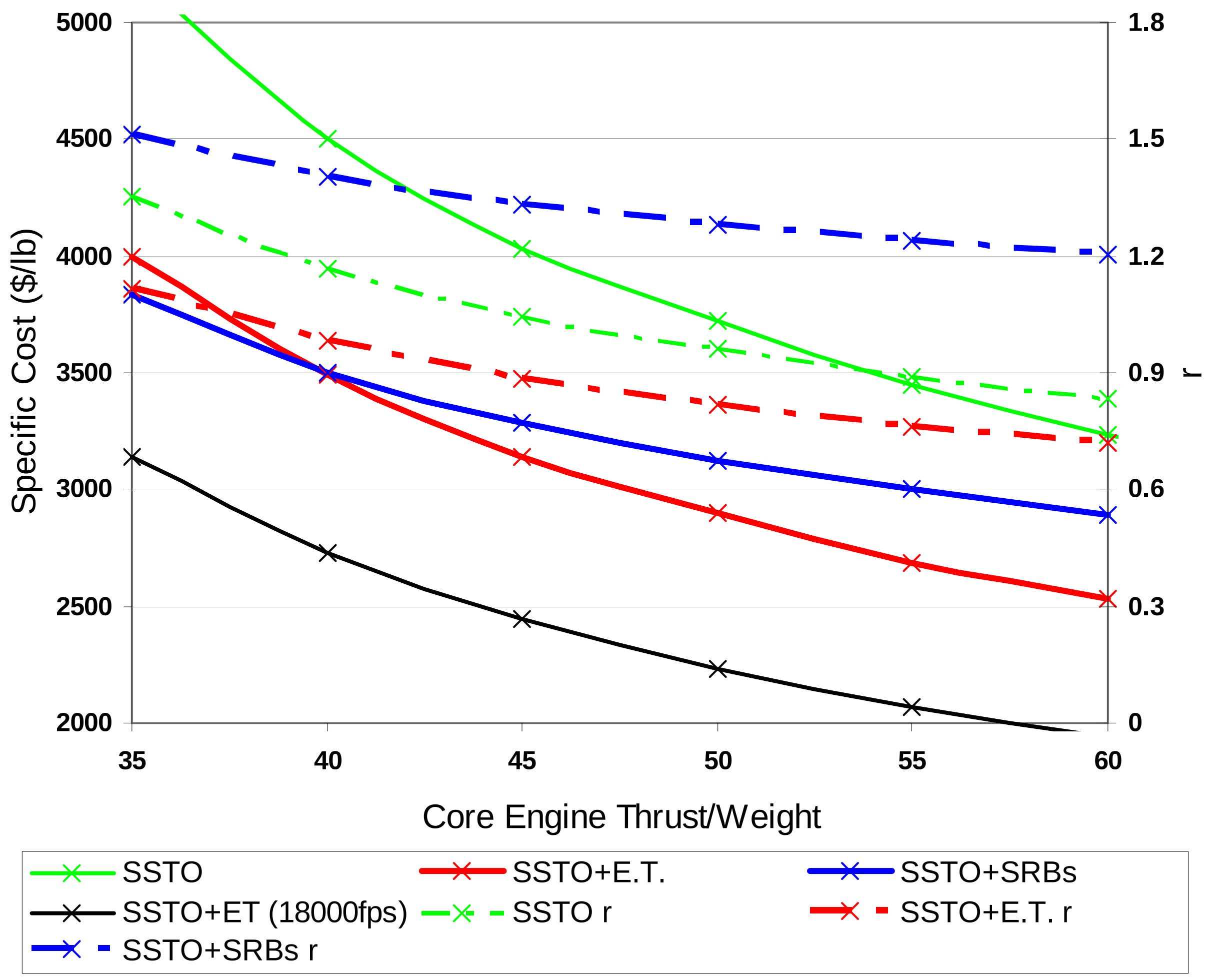


Figure 5. Specific Cost and Structure-Payload Mass Ratio Estimates vs. Core Engine Thrust-to-Weight Ratio for Baseline SSTO, SSTO+SRB, and SSTO+ET Space Launch Vehicles

portion of the SSTO+ET configuration's flight, the external tank separation velocity is at 5,000 ft/s (minus any aerodynamic and gravity losses) and the reusable core will have all of its internal fuel remaining. The SSTO+SRB configuration burns some of the internal fuel of the reusable core in parallel with the SRBs, and so the SRBs will actually separate at more than 5,000 ft/s velocity and leave a partially fueled core to continue into LEO. The results of this comparison are shown in Figure 5 where the solid blue line in represents the specific cost of the SSTO+SRB configuration and the solid red line represents the specific cost of the SSTO+ET configuration. These lines illustrate that using solid rocket boosters to augment the SSTO-like vehicle is cheaper than using external tanks only for engine thrust-to-weight ratios less than about 40. At engine thrust-to-weight ratios higher than 40, the SSTO+ET configuration is more economical. This graph assumes an engine *Isp* of 450 s, and a lower *Isp* engine would shift the crossover point between SRBs and ETs towards higher engine thrust-to-weight ratios. It seems very likely, however, that engine performance on any next-generation space launch vehicle would be good enough that SSTO+ET would be the more economical configuration.

The solid green line in Figure 5 represents the specific cost of a pure SSTO configuration using the baseline vehicle cost and performance assumptions, and shows that at over the entire range of engine thrust-to-weight ratios calculated the pure SSTO is more expensive than an SSTO-like core augmented by either solid rocket motors or external tanks. The solid black line in Figure 5 represents the specific cost of the SSTO+ET configuration if the external tank separation velocity can be 18,000 ft/s, which produces an external tank of about the optimum size for low specific cost. This line is below the SSTO+SRB specific cost line over the entire range of engine thrust-to-weight ratios calculated, and shows that the SSTO+ET is a much lower cost option when both designs are sized separately for lowest specific cost instead of limiting the external tank ΔV contribution to the optimal size chosen for the SRB configuration. The dotted blue, dotted green, and dotted red lines in Fig. 5 (all read on the secondary Y axis) represent the structure-payload mass ratios for the SSTO+SRB, pure SSTO, and SSTO+ET configurations, respectively.

The analysis in this section shows that for the estimated characteristics of next-generation LOX/LH_2 launch vehicles, a reusable true single-stage-to-orbit vehicle will have a higher specific cost for delivering payloads to LEO than a similar reusable core that is augmented by external tanks. The savings generated in amortized development cost and reduced launch vehicle mass achieved by adding external tanks to an SSTO design are more than enough to make up for the added hardware cost of expendable external tanks at any level of vehicle performance and flight rate likely to be achieved in the near future. Augmenting the reusable SSTO design with solid rocket boosters also produces a lower specific cost than a pure SSTO. With currently achievable levels of engine performance, however, the external tank augmented SSTO is cheaper than the SRB augmented SSTO, even if the SRBs have no development cost and the external tank sizing is constrained to provide no higher ΔV contribution than the SRBs. We already have good enough rocket engine technology to make the SSTO+ET configuration more economical than the SSTO+SRB configuration. We will not have the space launch flight rates needed to make the pure SSTO more economical than the SSTO+ET configuration for the foreseeable future. Anyone attempting to design a single-stage-to-orbit launch vehicle at this time or in the near future should strongly consider using external tanks reduce the launch system's specific cost.

VI. Bimese and External Tanks

External tanks can also be used to lower the specific cost of launch vehicles using identical stages by allowing staging velocities closer to optimum. As explained in Section 3, the amortized development cost can be reduced by lowering the developed structure-payload mass ratio. The developed structure-payload mass ratio can be lowered by building a launch vehicle with identical stages or stage components. The simplest case of this is a bimese rocket where the booster and orbiter stages are identical. In the case of a bimese, the developed structure-payload mass ratio would be half of the structure-payload mass ratio since only one of the stages would need to be developed and the production quantity for the stage doubled to provide the other half of the assembled vehicle. Unfortunately a bimese rocket suffers from staging inefficiencies that drive the total vehicle mass up above a conventionally sized rocket and cancels out some of the advantage of having the developed structure-payload mass ratio be based on only half of the vehicle's total mass. The hypothetical launch vehicle with a structure-payload mass ratio of 2 used to calculate current launch costs in Section 3 could be achieved as a two stage rocket and an average *Isp* of 380 s if the lower stage is allowed to be 4.6 times larger than the upper stage. If both stages are forced to be the same size in order to achieve a bimese configuration, the structure-payload mass ratio increases to 4 and the developed structure-payload mass ratio still remains 2!

Adding an external tank (or tanks) to the bimese design lowers the ΔV that must be provided by the orbiter. This results in more efficient staging and a lower overall system structure-payload mass ratio. In this case the external tank reduces the amortized development cost by lowering the developed structure-payload mass ratio of the launch system and allowing the cost savings potential of the bimese concept to be realized. To illustrate these cost savings, the specific costs and mass ratios for a baseline bimese, bimese with external tank, conventional two-stage-to-orbit (2STO), conventional three-stage-to-orbit (3STO), and trimese launch system were estimated using the cost model described in Section 2 with the Rocketcost spreadsheet. The bimese launch system crossfeeds propellant from the booster to burn both the booster and orbiter in parallel at liftoff, and then separates the booster when its propellant is expended. The bimese+ET system initially burns both the booster and orbiter engines from an external tank; the external tank separates when its propellant is expended and the vehicle continues as a normal bimese system

Table 6. Reusable Launch Vehicle Characteristics

Characteristic	2STO/3STO	Bimese/Bimese+ET/ Trimese
f	0.02	0.02
f_{ET}	N/A	1
$c_{H,}$	$2000/lb.	$2000/lb.
$c_{H, ET}$	N/A	$500/lb.
L	0.75 hr/lb. (2 stage) 1 hr/lb. (3 stage)	0.75 hr/lb. (bimese) 1 hr/lb. (others)
c_L	$100/hr	$100/hr
P_{FAIL}	0.5%	0.5%
c_{PL}	$10,000/lb.	$10,000/lb.
c_P	$0.24/lb.	$0.24/lb.
λ_D	$50,000/lb.	$60,000/lb.
$\lambda_{D, ET}$	N/A	$15,000/lb.
a	27	27
Isp	440 s	440 s
η	.918	.918
T/W	N/A	40
η_T	N/A	0.95

afterwards. The trimese launch system studied is composed of three identical major components with two components serving as boosters and one as an orbiter. The three components burn in parallel at launch while crossfeeding propellant from the boosters to the orbiter. After the trimese boosters expend their propellant, they separate and the fully fueled orbiter continues to orbit.

The Rocketcost input characteristics of these launch systems with reusable stages and expendable external tanks are given in Table 6. Both the development structure cost and the performance for these vehicles is lower than the performance assumed for the SSTO-like vehicles analyzed in Section 5. These reductions were made because the designs in this section are more forgiving of lower performance than the technically challenging SSTO systems. The bimese and bimese+ET vehicle masses were calculated by separating the structure mass into tankage-related mass and engine-related mass. Tankage-related mass was calculated from the amount of propellant and the tankage propellant mass fraction, η_T. Engine-related mass was calculated using the GLOW and engine trust-to-weight ratio, T/W, to achieve a vehicle thrust-to-weight ratio of 1.25 at liftoff. The 2STO, 3STO, and trimese vehicle masses were calculated using the same vehicle propellant mass fraction that was achieved by the bimese designs. The staging velocities for the 2STO, 3STO, and bimese+ET vehicles were chosen to minimize specific cost. The staging for the bimese and trimese designs are fixed at suboptimal velocities by the nature of the concepts. The slightly higher development structure cost for the bimese, bimese+ET, and trimese concepts reflects the increased difficulty of designing one stage that can function as either a booster or orbiter.

The results of the specific cost and structure-payload mass ratio calculations for these launch systems are given in Table 7. With the performance characteristics assumed in these calculations, the bimese system does achieve a lower developed structure-payload mass fraction than a conventionally sized 2STO vehicle. Despite having a larger recurring specific cost, the reduced amortized development specific cost allows the bimese system to have a lower overall specific cost than the conventional 2STO. By both adding another staging event and improving the staging efficiency of the bimese concept, the bimese+ET system allows a lower structure-payload mass ratio than the bimese and a substantially lower developed structure-payload mass ratio than either the 2STO or bimese system.

Table 7. Reusable Launch Vehicle Specific Costs and Estimated Structure-Payload Mass Ratios

	2STO	Bimese	Bimese+ET	3STO	Trimese
Specific Costs ($/lb)					
Amortized R&D	**$1,676**	**$1,151**	**$857**	**$1,535**	**$713**
Hardware	$36	$41	$197	$33	$39
Operations	$68	$78	$94	$83	$96
Other	$61	$63	$59	$60	$61
Recurring	**$165**	**$182**	**$350**	**$176**	**$196**
Total	**$1,841**	**$1,333**	**$1,207**	**$1,711**	**$909**
r					
Stage 1 / ET	N/A	N/A	0.346	0.503	N/A
Stage 2 / Booster	0.703	0.52	0.299	0.225	0.642
Stage 3 / Orbiter	0.202	0.52	0.299	0.101	0.321
Total	**0.905**	**1.03**	**0.944**	**0.829**	**0.963**
Developed	**0.905**	**0.52**	**0.386**	**0.829**	**0.321**
Orbiter ΔV (ft/s)	15,000	19,767	13,200	10,000	16,100

The developed structure-payload mass ratio for the bimese+ET system was calculated to be the r of one of the identical reusable stages plus ¼ of the r for the expendable external tank, because the external tank development structure cost was assumed to be ¼ of the reusable stage developed structure cost. Expending the external tank increases the hardware specific cost of the bimese+ET considerably over the plain bimese system, and the additional vehicle integration needed raises the operations specific cost as well. The savings in amortized development specific cost resulting from its lower developed structure-payload mass ratio gives the bimese+ET system a lower specific cost than the bimese despite the design's higher recurring specific costs. To show that the bimese+ET system's lower specific cost was not just a result of being essentially a three stage vehicle compared to the bimese's two stage configuration, the results for a conventionally sized 3STO launch system are also included in the table. The 3STO system is slightly cheaper than the 2STO system, but is still more expensive than either of the bimese designs.

Table 8. Expendable Launch Vehicle Characteristics

Characteristic	2STO/3STO	Bimese/Bimese+ET/ Trimese
f	1	1
f_{ET}	N/A	1
$c_{H,}$	$925/lb.	$1000/lb.
$c_{H, ET}$	N/A	$500/lb.
L	0.75 hr/lb. (2 stage) 1 hr/lb. (3 stage)	0.75 hr/lb. (bimese) 1 hr/lb. (others)
c_L	$100/hr	$100/hr
P_{FAIL}	0.5%	0.5%
c_{PL}	$10,000/lb.	$10,000/lb.
c_P	$0.24/lb.	$0.24/lb.
λ_D	$27,000/lb.	$30,000/lb.
$\lambda_{D, ET}$	N/A	$15,000/lb.
a	27	27
Isp	440s	440s
η	.928	.928
T/W	N/A	40
η_T	N/A	0.96

The trimese design is the cheapest of the five systems analyzed, with an estimated specific cost of less than $1000 per pound of payload delivered to LEO. Because the effect of adding an external tank is more pronounced for the bimese design and the fact that trimese designs received considerable analysis during the late 1960s and early 1970s, the bimese configuration was selected to receive most of the analysis effort in this section. Adding external tanks to the trimese design could improve its cost even further by allowing it to have more efficient staging as well, but the resulting savings would not be as dramatic because the staging of the basic trimese design is not as inefficient as the staging of the bimese.

Typically, external tanks are utilized with reusable launch vehicles as a sort of "staging lite." External tanks allow the designer to reduce the mass of a launch vehicle during ascent by dropping unneeded tankage while retaining the vehicle's engines. Since there is not usually a motive to "save" the engines on an expendable rocket only to discard them later in the flight, external tanks are not usually seen on a launch vehicle design unless it is at least partially reusable. In theory a launch vehicle could be designed to drop unneeded engines during ascent (as was done with the Atlas) and also drop external tanks separately to get rid of unneeded tankage mass. This technique seems to be an "answer in search of a problem" and it is simpler to just attach the engines and tankage together on an expendable rocket and drop them as one stage. The bimese+ET

Table 9. Expendable Launch Vehicle Specific Cost and Structure-Payload Mass Ratio Estimates

	2STO	Bimese	Bimese+ET	3STO	Trimese
Specific Costs ($/lb)					
Amortized R&D	**$750**	**$468**	**$444**	**$697**	**$297**
Hardware	$694	$843	$659	$645	$803
Operations	$56	$63	$80	$70	$80
Other	$52	$53	$52	$52	$52
Recurring	**$802**	**$959**	**$791**	**$767**	**$935**
Total	**$1,552**	**$1,427**	**$1,235**	**$1,464**	**$1,232**
r					
Stage 1 / ET	N/A	N/A	0.282	0.42	N/A
Stage 2 / Booster	0.579	0.421	0.259	0.191	0.535
Stage 3 / Orbiter	0.171	0.421	0.259	0.087	0.268
Total	**0.75**	**0.843**	**0.799**	**0.697**	**0.8027**
Developed	**0.75**	**0.421**	**0.400**	**0.697**	**0.268**
Orbiter ΔV (ft/s)	15,000	19,767	13,200	10,000	16,100

system is an interesting exception to this general tendency. Some of the savings for the bimese+ET design over just a bimese design comes from allowing the system to stage the orbiter at more optimal velocities. Using an external tank to accomplish this allows the booster and orbiter vehicle to remain the same and insures that the only engines used on the launch system are on those identical stages. The external tank in an expendable bimese+ET system could still reduce the overall cost of the system. It would do this, not by allowing the engines to be reused, but instead by pushing all the engines onto the identical bimese vehicles so that the engine-related development work on what is essentially a three stage vehicle only has to be done for the one stage design! Multiple copies of the one design are used for both the orbiter and booster vehicles, and the external tank itself has no engine.

To demonstrate that in the specific case of a bimese configuration adding an expendable external tank to an already fully expendable launch system can reduce cost, the analysis for the 2STO, bimese, bimese+ET, 3STO, and trimese systems were redone assuming they were expendable. The Rocketcost input characteristics of these launch systems with expendable stages and external tank are given in Table 8. The performance characteristics of these expendable launch systems are similar to the characteristics of the reusable systems with a few exceptions. The tankage propellant mass fraction, η_T (and the resulting vehicle propellant mass fraction) for the expendable vehicle is slightly higher to reflect the lighter weight of a fully expendable vehicle. The development and hardware structure costs of the main stages are also reduced because the design will not need the thermal protection and recovery systems that a reusable stage would require. As with the reusable systems, the development and hardware structure costs of the expendable 2STO and 3STO systems are lower than for the designs that require identical stages. Most significantly, the fraction of the vehicle expended, *f*, is now 1 for all system components. The results of the specific cost and structure-payload mass ratio calculations for these expendable launch systems are given in Table 7, and show the same rankings for specific cost as their reusable counterparts. The expendable bimese system has a lower specific cost than either the 2STO or 3STO systems. The expendable bimese+ET has a lower specific cost than the bimese system, thus demonstrating that under the right circumstances even the economics of a fully expendable system can be improved by incorporating an external tank. The expendable trimese system has the lowest cost of all the fully expendable designs analyzed in this section. Structure-payload mass ratios for the systems are also listed in Table 9. The developed structure-payload mass ratio for the bimese+ET system was calculated to be the *r* of one of the identical stages plus ½ of the *r* for the external tank, because the external tank development structure cost was assumed to be ½ of the stage developed structure cost.

VII. Partially Reusable Lower Stages

Previous work by Taylor[5] illustrated that the general trend for economical, multi-stage, cargo launch vehicles is to have expensive and high-performance upper stages on cheaper and lower performance lower stages. Because adding reusability to a launch vehicle stage lowers the cost at the expense of making it heavier, the “natural home” of reusability on next generation cargo launch vehicles is in the lowest stage. Not all launch vehicle stage components are valuable enough that the cost savings for reusing them justifies the cost and weight associated with making them reusable. An alternative to a fully reusable first stage is a partially reusable first stage. The partially reusable stage would put high value systems, like engines and avionics, in a recoverable module while low value systems, like tankage, are not recovered. Previous launch vehicle proposals that include partially reusable stages of this type include the National Launch System (NLS) and original Boeing EELV concepts.

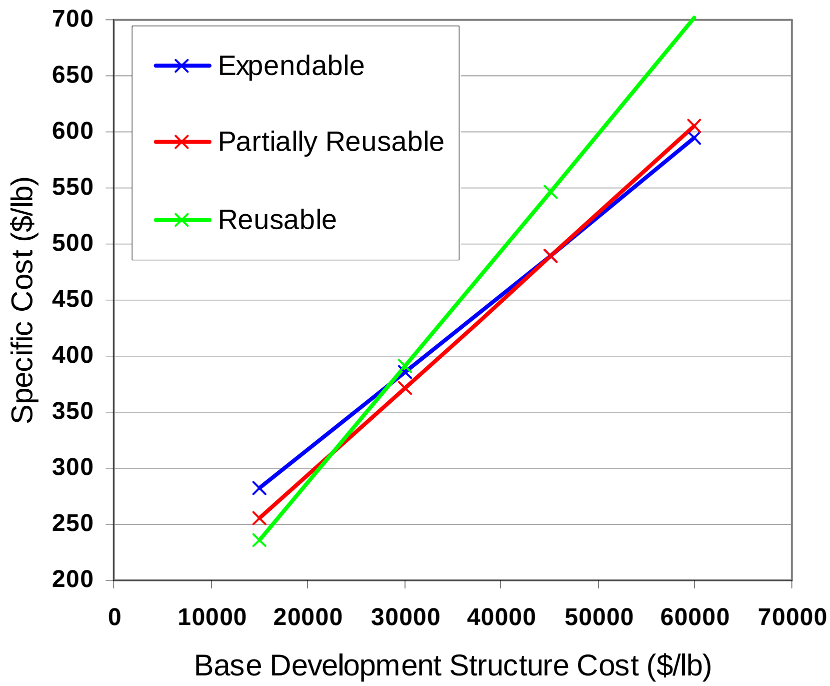
Table 10. Lower Stage Characteristics and Estimated Masses

Characteristic	Expendable	Partially Reusable	Reusable
f_{engine}	1	0.05	0.05
$f_{tankage}$	1	1	0.05
$c_{H,}$	$1000/lb	$1000/lb	$1000/lb.
$c_{H, ET}$	$500/lb	$500/lb	$500/lb.
L	0.5 hr/lb.	0.75 hr/lb.	1 hr/lb.
c_L	$100/hr	$100/hr	$100/hr
P_{FAIL}	0.5%	0.5%	0.5%
c_{PL}	$10,000/lb.	$10,000/lb.	$10,000/lb.
c_P	$0.13	$0.13	$0.13
λ_D	variable	variable	variable
a	27	27	27
Isp	315s	315s	315s
η	.939	0.937	0.930
T/W	100	87	87
η_T	0.955	0.955	0.948
M_{engine}	4,069 lb.	4,737 lb.	4,806 lb.
$M_{tankage}$	10,885 lb.	10,971 lb.	13,060 lb.
M_P	231,004 lb.	232,838 lb.	238,086 lb

To demonstrate the economics of lower stage reusability, the specific cost and mass of a new lower stage was calculated for expendable, partially reusable, and fully reusable configurations. The lower stage in this analysis will be sized to give an 80,000 lb. existing upper stage and payload an ideal ΔV of 12,500 ft/s. The specific costs of these stages were calculated using the cost model described in Section 2 with the Rocketcost spreadsheet. The performance and propellant cost for these stages were selected to approximate a current generation LOX/RP vehicle. The Rocketcost inputs for these stages and the resulting stage masses are listed in Table 10. The stage masses were calculated by separating the structure mass into tankage-related mass and engine-related mass. Tankage-related mass was calculated from the amount of propellant and the tankage propellant mass fraction, η_T. Engine-related mass was calculated using the GLOW and engine trust-to-weight ratio, *T/W*, to achieve a vehicle thrust-to-weight ratio of 1.25 at liftoff. The tankage propellant mass fraction and the engine thrust-to-weight ratio on reusable components were lowered by 15% to account for their increased sturdiness and the extra mass needed for the recovery system. The stage specific costs were calculated for various base development structure costs that were increased by an additional 25% on reusable components to account for the extra development work needed on the recovery and refurbishment aspects of the stage's life cycle. The development structure cost was handled in this two-stage manner to separate the cost due to the increased technical difficulty of a reusable system from the cost due to the competency (or incompetency) of the development organization at developing rocket vehicles.

The specific costs for the three stage concepts are graphed in Figure 6 as a function of the base development structure cost. At high development structure costs (above $45,000/lb), the lower structure mass of the expendable design gives it the lowest specific cost. At very low development structure costs (below $20,000/lb), the fully reusable design gives the lowest specific cost. Over the middle range, the partially reusable design has the lowest specific cost. Over a wide range of the likely development structure cost for this stage, the recovery and reuse of the expensive engine related hardware can be justified economically but the recovery of the propellant tanks cannot. The minimum cost configuration over this middle range of development structure costs would have a recoverable engine pod containing the expensive stage components that drops away from expendable propellant tanks after staging. It could be argued that the expendable propellant tanks in this configuration should not be referred to as "external tanks" because they probably would not be external to the main structure of the stage but from a costing standpoint they fill the same roll as an external tank regardless of their physical location in the vehicle.

Figure 6. Specific Cost Estimates vs. Base Development Structure Cost for Expendable, Partially Reusable, and Reusable Lower Stages



VIII. Conclusion

Whether it is called an expendable tank, a drop tank, a tip tank, or an ET, the external tank is still a useful concept for engineers seeking to reduce the specific cost of launch vehicles with reusable components. Under certain conditions when using identical stages to reduce amortized development cost, the external tank can even play a useful role in fully expendable launch vehicle designs. In the future, perhaps vehicle performance and vehicle flight rate will be high enough (or vehicle development costs low enough) that economical, fully reusable launch vehicle stages will eliminate the need for external tanks. Those conditions are not likely to arrive in time to displace the external tank from the next generation of reusable launch vehicle designs.

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