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Economics of Separated Ascent Stage Launch Vehicles

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A separated ascent stage launch system is one where multiple vehicles launch and ascend separately, but cooperate through either aerial refueling or momentum transfer to help one of the vehicles attain orbit. The use of separated ascent stage launch vehicles first requires overcoming the difficult technical hurdle of in-flight refueling or in-flight docking of launch vehicles during ascent. If this technical hurdle could be overcome, however, separated ascent stage launch vehicles would have many characteristics beneficial for economical space access. The physical separation of component units would allow a large number of identical stages to be used without needing to account for complex interactions between the stages or designing the stages to function in a large number of configuration locations. The use of a large number of identical stages would reduce the development cost for the launch system and could also reduce hardware and operations costs through economies of scale. By changing the number of cooperating vehicles in the launch system a wide variety of missions might be carried out with the same basic design, allowing a high flight rate and further improving the launch system's economics. It is unknown whether these benefits would justify the development effort needed to perfect in-flight mating of launch vehicles.

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
MTOW	=	maximum take-off weight
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
η	=	propellant mass fraction = $M_P/(M_P+M_S)$
λ	=	structure cost = C_{TOT}/M_S
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
λ_D = vehicle development structure cost

I. Introduction

The largest component of the cost of delivering payloads to orbit on new space launch vehicles with current technology is the amortized development cost. This cost can be reduced by reducing the amount of development work required for a new launch vehicle. One way to accomplish this is to use identical stages or stage components so that a new launch vehicle can be developed for the cost of only one stage. Using a launch system comprised of identical launch components could also reduce hardware and operations costs through economy of scale. If a small number of identical components are used then staging inefficiencies occur. If a large number of components are clustered then complex interactions and the varying constraints of different configuration locations in the cluster makes the vehicle development program difficult. Physically separating some of the identical launch system components during ascent would allow a large number of components to be used without incurring the design difficulties that come with large stage clusters. This could be accomplished using a flock of separated ascent stage launch vehicles that launch simultaneously and cooperate through in-flight refueling and/or momentum transfer to help one of them achieve orbit. Separated ascent stage launch vehicles might be able to carry out a very wide range of missions by varying the number of vehicles launched and their trajectory and therefore reduce amortized development cost even further by providing more missions to amortize the development cost over. Separated ascent stage launch vehicles would require the demonstration of in-flight refueling or docking for space launch vehicles. This could be an expensive technology to develop, and this paper attempts to produce a "back of the napkin" estimate of the cost savings that separated ascent stage launch vehicles could generate in order to encourage further debate on the concept of in-flight refueling or docking of 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 greater detail 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 in Eq. (2) is the recurring launch costs. The second term in Eq. (2) 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 napkin" 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, but that additional level of detail is not required for the analysis 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.

$$\lambda_L = Lc_L \qquad (5)$$

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

$$q = \eta / (1 - \eta) \quad (8)$$

A computer spreadsheet 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

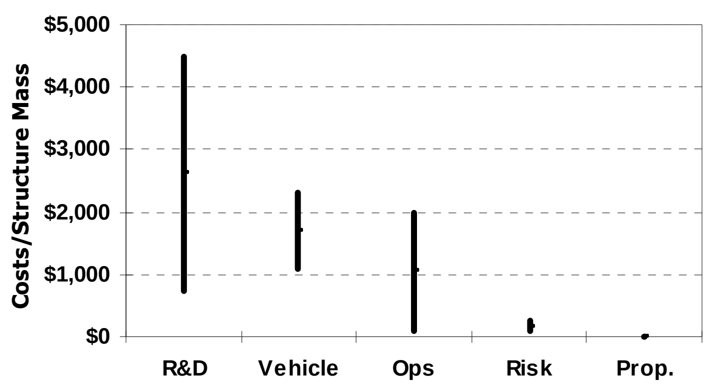
Characteristic	Values
f	1
c_H	\$1100/lb.to \$2300/lb.
L	1 to 20
c_L	\$100/hr
P_{FAIL}	2% to 5%
c_{PL}	\$10,000/lb.
c_P	\$0.1/lb. to \$0.25/lb.
λ_D	\$20,000/lb.to \$120,000/lb.

Table 3 gives the estimated current state of launch vehicle 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 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. This conclusion contradicts the "conventional wisdom" that the two biggest components of space launch cost are vehicle hardware and flight operations. 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 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. Reduction of Amortized Development Cost

Even if all other cost components could be eliminated, typical amortized development costs would still keep space launch costs above $1000/lb of payload to LEO. If truly cheap space access is desired, then some way must be found to reduce the amortized non-recurring costs of the next generation of launch vehicles. The second term in Eq. (2) represents this amortized vehicle development cost as r_D (λ_D/a). To build a future economical launch vehicle will require increasing the amortization factor, *a*, reducing the vehicle development structure cost, λ_D, reducing the developed structure-payload mass ratio, r_D, or some combination of these strategies.

A. Increasing the Amortization Factor

Amortization factor, *a*, determines the fraction of the launch vehicle development cost that is charged to each launch. In a simple analysis, ignoring interest and inflation, amortization factor is simply equal to the number of launches the non-recurring development costs will be divided between. Perhaps the most common approach to reducing the burden that amortized development cost puts on space launch access is to increase the amortization factor, *a*, by spreading the development cost over more launches. This usually requires increasing the vehicle flight rate. Considerable work has been done in trying to find new markets, such as space tourism or suborbital package delivery, for space launches that will allow high vehicle flight rates. Another way to decrease the amount of development cost that must be amortized by each space launch is to share the development cost of technologies with other users. For example, if a new rocket engine is used on several launch vehicles then the development cost for that engine can be spread over the launches of all those vehicles. Much of the early space launch vehicles shared components with similar military missiles, so the cost of those developments did not have to be borne entirely by the civilian space launch applications. Savvy manufacturers can sometimes still get government programs to pay for developments that are subsequently used on private space launch vehicles, but whether this is a case of multi-use technology spreading the amortization of development costs or just a way to shift development cost to the taxpayers so that an uneconomical vehicle can still be profitable is not always clear.

B. Reducing Vehicle Development Structure Cost

Vehicle development structure cost, λ_D, is the cost per unit mass of vehicle structure to design a new space launch vehicle. One way to reduce the vehicle development structure cost is to effectively manage the organization developing the vehicle to accomplish the development program cheaply. Effective program and organizational management can have a huge impact on cost savings, but being non-technical in nature the methods to accomplish this are beyond the scope of this paper. Traditional management methods for accomplishing aerospace vehicle development at low cost include clearly defining the vehicle's mission objective at the start of the development program and not changing it, using a small, highly talented development team, and isolating the development team in a remote location where there is little else to do and uninvolved higher level managers are not tempted to make up excuses for an on-site visit.

Another way to reduce vehicle development structure cost is to reduce the technical difficulty of the development program. This can be accomplished by making careful design decisions early in the program to minimize R&D costs. Components, such as engines, can be sized and designed for a low development cost at the expense of vehicle performance and often produce a more economical vehicle than if the components were designed to maximize performance. The launch vehicle configuration can be selected to avoid uneconomically ambitious development requirements. Some vehicle configurations, such as a single-stage-to-orbit (SSTO) vehicle, require very low structural mass fractions and/or are very sensitive to spiraling growth in vehicle mass. A large amount of development effort is required to find every weight savings possible to make these technically demanding configurations work. Design problems that arise in them 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 operations looks great when examined from the standpoint of recurring costs, but might be unaffordable when amortized development costs are included. Savings in vehicle hardware and operational costs that technically demanding launch vehicle configurations are usually selected to achieve can be more than offset by higher vehicle development structure costs the configurations require.

C. Reducing Developed Structure-Payload Mass Ratio

Developed structure-payload mass ratio, r_D, is the mass of new vehicle structure that must be developed divided by the payload mass. One way to reduce the developed structure-payload mass ratio is to reduce the overall vehicle structure-payload mass ratio, r. If a smaller launch vehicle is built, then it seems natural that it would cost less. If the developed vehicle mass for is reduced by reducing the total vehicle mass for a fixed payload mass, then certain recurring launch vehicle costs would be reduced as well. Unfortunately, reducing vehicle structure-payload mass ratio usually requires saving vehicle weight or improving vehicle performance which increases the vehicle development structure cost. For a launch vehicle design, there will be a vehicle structure-payload mass ratio that provides a minimum development cost. Cost savings resulting from a vehicle smaller than this optimum structure-payload mass ratio would be more than offset by increased costs incurred in the more difficult development program.

Fortunately there are other ways to reduce the developed structure-payload mass ratio without having to reduce vehicle's structure-payload mass ratio. Developed structure-payload mass ratio is only equal to structure-payload mass ratio if the development program must develop the entire vehicle. If not all of the launch vehicle has to be developed, then not all of the vehicle mass has to be counted as part of the developed structure-payload mass ratio. For example, if a good existing upper stage exists then that stage might be combined with a newly developed lower stage. Only the mass of the new lower stage would need to be counted when calculating the developed structure-payload mass ratio. While this "new launch vehicle" would really only be a partially new vehicle, there is no advantage to having a new upper stage for the sake of newness if the replacement of the old one cannot be justified economically. Similarly, existing solid rocket motors might be used as solid rocket boosters for an otherwise new launch system in order to reduce the size of the development program.

In addition to reusing old stages to reduce the developed structure-payload mass ratio, smaller vehicle components can be reused as well. Engines, structure, electronics, or other parts that have been proven to have good performance in past launch vehicles might be reused in future ones with the development program focusing only on portions of the vehicle where the cost of a new component can be justified by its increased performance. A launch vehicle that is designed in this evolutionary way could have a noticeably lower amortized development cost. There would still be some new development effort associated with the older parts to integrate them into the new vehicle and so the exact value of the developed structure-payload mass ratio might not be clear, but it would obviously be lower than the full value of the vehicle's structure-payload mass ratio.

Even if the entire vehicle and its components are the result of a new development effort, the developed structure-payload mass ratio might still be lower than the vehicle's structure-payload mass ratio if it uses identical stages. 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 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 III could be achieved as a two stage rocket and an average Isp of 380s 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!

Trimese launch vehicles, composed of three identical major components with two components serving as boosters and one as an orbiter, were studied in the late 1960's and early 1970's and would have a developed structure-payload mass ratio that was only one third of the structure-payload mass ratio. The staging inefficiencies are not as bad with this design and using the same assumptions described above a trimese launch vehicle would have a structure-payload mass ratio of about 2.3 and a developed structure-payload mass ratio of about 0.8. The greater complexity of having one stage that could function as both a booster and orbiter would drive up the vehicle development structure cost, but this "back of the napkin" analysis suggests that a trimese configuration might still be a viable candidate for a low-cost next generation launch vehicle based on its low developed structure-payload mass ratio.

V. Separated Assent Stage Vehicles

If amortized development cost can be reduced by using a trimese design, then it is logical to ask if using even larger numbers of identical stages might result in even greater savings. Ed Keith proposed[12] an Asparagus-Stalk Booster composed of seven identical units that crossfeed propellant and are dropped in pairs as they are emptied. Using pressure fed rockets with an Isp of only 290s, this concept was project to have a structure-payload mass ratio of about 3.8 but a developed structure-payload mass ratio of 0.54. With the development of modular micro-electro mechanical rockets[13], perhaps a launch vehicle could be constructed of thousands of identical subcomponents that detach when no longer needed. As the number of identical clustered stages in the vehicle increases, however, additional development problems occur that increase the developed structure cost and may wind up actually increasing the amortized development cost while the developed structure-payload mass ratio declines. This increased developed structure cost comes from the increasing technical challenge of accounting for the interactions between a large number of identical components and designing one stage that can function in any location in the vehicle despite the differences in the structural, aerodynamic, thermodynamic, and acoustic environments. These interaction problems would be reduced or eliminated, however, if the identical components that comprised the launch system could be physically separated from each other instead of clustered together. This is not impossible if a separated ascent stage space launch system is used.

A separated ascent stage launch system is one where multiple vehicles launch and ascend separately, but cooperate through either aerial refueling or momentum transfer to help one of the vehicles attain orbit. Perhaps the earliest proposed example of a separated ascent stage launch vehicle is a variation of the Black Horse rocketplane. In addition to the normal mode of operation, a "speculative operational mode" was also proposed[14] where the payload of the rocketplane could by increased by launching two rocketplanes simultaneously on a suborbital, exo-atmospheric trajectory. Once out of the atmosphere, the rocketplanes would rendezvous and one would extend a refueling boom to the other and transfer its remaining propellant. The empty rockeplane would return to earth while the refueled rocketplane reignited its engines and continued to orbit.

Another example of a separated ascent stage launch system is the family of FLOC[15] (or Flock[16]) concepts developed by Novatia Labs. The original FLOC space launch concept involved the simultaneous launching of a number of identical, reusable launch vehicles configured in a bimese arrangement. The bimese stages would crossfeed propellant and the booster stage would return to Earth when empty. The remaining single upper stages would pair up, dock with each other, and continue as before until the next set of boosters was empty. The remaining stages would continue this process of pairing up and crossfeeding propellant until only one upper stage was left with enough velocity and fuel to reach orbit. Novatia described different flock sizes and configurations for different payload masses and vehicle performance targets, with the largest "flock" being comprised of 64 identical rocketplanes that are launched as 32 sets of bimese vehicles which then undergo six sets of stage separations and five sets of redockings until the 63 boost vehicles have returned to Earth and the one orbiter vehicle has arrived in LEO with a very large payload. The developed structure-payload mass ratio for this 64 unit flock would be 1/64th of the total system structure-payload mass ratio, but because the 64 units are only ever flown either singly or in pairs the vehicles would not need to be designed to operate in any larger of a cluster than a simple bimese rocket would. In this case, the complexities of clustering 64 identical units have been eliminated but at the cost of requiring a new technical hurdle of in-flight refueling to be overcome.

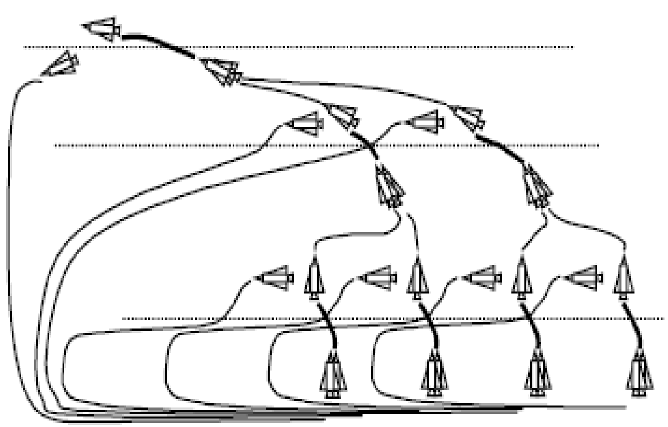


Figure 2. FLOC Architecture: A Flock of 2^n rocket planes[16]

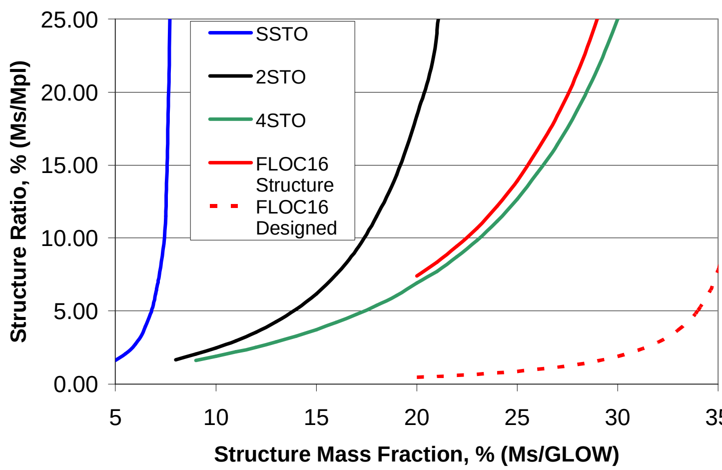
From a cursory examination of its characteristics it would seem that the FLOC concept was conceived to minimize developed structure-payload mass ratio at the expense of a much higher development structure cost, but the reality is almost exactly the opposite. Novatia proposed the FLOC concept as a way to reduce vehicle development and hardware costs by producing a system that is much less sensitive to weight growth than previous launch vehicle concepts and could deliver a useful payload to orbit with a relatively large dry mass fraction. Goff[15] describes the analysis that lead to the FLOC concept as follows:

> The consequence of the limit law is that launch systems must be designed to maximize performance. Almost every engineering tradeoff is made in favor of performance, at the expense of cost, reliability, safety, reusability, maintenance, turn-around time, etc. Not to do so yields systems with zero capability. The result is extremely costly access to space with frequent failures… The primary benefit of Flock is a fuel scaling law that is not a limit law and system performance that is relatively insensitive to the dry mass fraction of the individual boosters. This means that a Flock rocket plane can be designed to life cycle costs instead of performance. High dry mass fractions offer many new engineering tradeoffs, which can benefit cost, safety, reliability, maintainability, turnaround times, small ground crews, etc.

The original baseline FLOC design used to illustrate the concept was based on a reusable 400 ton GLOW rocketplane with an Isp of 372 and a structure mass fraction of 29%. This structure mass fraction was a WAG[†] made by Novatia based on the Boeing 747's dry mass fraction. With a maximum payload in the 32 unit flock configuration (31 boosters and 1 orbiter) of 178 tons this system would have an overall structure-payload mass ratio of 20.9 but a developed structure-payload mass ratio of only 0.65. This developed structure-payload mass ratio is only slightly better than the trimese configuration described above, but it is accomplished with 7s less of Isp and a structure mass fraction approximately three times as large. If the technical hurdles of mid-flight refueling can be overcome, then the FLOC vehicle would present a much easier design challenge than a reusable trimese vehicle.

The insensitivity of the FLOC concept to weight growth is primarily a product of two factors. The first that the repeated separation and docking of FLOC rockeplanes as some of them run out of fuel, which is called "binary staging" by Novatia, has a benefit to system performance similar to that of regular staging. This effect is demonstrated in Figure 3, where the weight growth of different launch vehicle concepts is illustrated. In this figure the structure-payload mass ratio and developed structure-payload mass ratio of a 16 unit FLOC is graphed against structure mass as a fraction of GLOW alongside those for single stage, two stage, and four stage to orbit vehicles with the same Isp. Figure 3 demonstrates that the 16 unit flock, which undergoes four binary staging events, shows almost the same insensitivity to weight growth that a conventional four stage rocket does.

Figure 3. Calculated Launch Vehicle Structure-Payload Mass Ratio vs. Structure Mass Fraction (Isp = 372)



The ability to increase the payload capacity of a FLOC launch system by doubling the flock size and adding another binary staging event to the trajectory gives it even greater insensitivity to weight growth than a conventional multi-stage rocket. If unexpected weight growth during vehicle development reduced the payload of a conventional launch vehicle below mission requirements then an expensive redesign effort would be required. With a FLOC launch system, the payload could be carried by the next highest flock size. Increasing the number of rockeplanes in the flock would increase hardware and operations cost, and require a redesign of the trajectory but it would not require the development of a new vehicle. It would not be nearly as easy to add a new lower stage to a conventional launch vehicle.

The ability of the FLOC concept to change the number of binary stagings and the size of the flock to accommodate a wide variety of payload masses potentially allows the amortized development cost to be reduced even further by increasing the number of launches conducted. A twenty ton payload could be launched with an eight unit flock one day, and the next mission might use a thirty-two unit flock to launch a two-hundred ton payload. Launched singly or in pairs, the FLOC rocketplanes could also perform suborbital missions such as space tourism rides or micro-gravity experiments. By allowing one vehicle design to economically perform a wide variety of space launch missions the flight rate would be increased, with a corresponding improvement in its amortization factor. This capability also provides the opportunity to start with easier missions and build the FLOC fleet into a

[†] Wild Guess

progressively more capable system as additional rocketplanes are acquired and the binary staging maneuver is perfected.

VI. FLOC Rocketplane Economic Analysis

Separated ascent stage launch vehicle concepts like FLOC offer the potential to greatly reduce vehicle development cost, and allow higher weight design choices that improve hardware and operations costs as well, at the expense of requiring the development of in-flight rocket refueling. The cost of a development program to demonstrate in-flight rocket refueling is not known, but examining the savings that separated ascent stage launch vehicles could generate in other areas will provide useful information on how expensive an in-flight rocket refueling demonstration program could be and still be justified economically. Novatia provided sufficient technical information on its original baseline FLOC configuration to permit a "back of the napkin" cost analysis of it. Two different types of cost analysis were carried out. Both cost analyses were done using the cost model described in Section II. The first assumed that the costs of the FLOC rocketplane behaved like the costs of other rocket-based space launch vehicles and used historical cost data for rockets. The second cost analysis was done assuming that the costs of the FLOC rocketplane were more like airplane costs and used historical airplane data.

A. Rocket Economic Analysis

The FLOC vehicle analyzed in this section is a 400 ton rocketplane composed of 116 tons of structure, 71 tons of propellant, and 213 tons available for either payload or additional propellant. The configuration analyzed in this section will be the 32 unit flock launching its maximum payload of 178 tons to low Earth orbit. This configuration has a structure-payload mass ratio of 20.9 and a developed structure-payload mass ratio of 0.65. Despite the potential for the FLOC to increase the amortization factor with a wide range of payload capability, the amortization factor, *a*, used for this analysis will remain 27. Assuming the high structure mass fraction of the vehicle allows an inexpensive development program, as Novatia hopes, but that the vehicle development program cost is still within the historic range for rocket launch vehicles a developed structure cost of $20,000 per pound was initially chosen for the cost analysis. This gives an amortized development cost per pound of payload delivered to orbit of (0.65 lb./lb. * $20,000/lb)/27, or $481 per pound of payload. This is a considerable reduction from the $1,500 to $9,000 per pound specific amortized development cost estimated for current launch vehicles in Section II.

Assuming that the vehicle hardware costs are also at the low end of historic values for rocket launch vehicles at $1000 per pound of structure and that the reusable rocketplanes are worn out at a rate of 1% per flight, vehicle hardware costs were estimated to be $209 per pound of payload. The specific cost of risk for FLOC is more difficult to estimate than for conventional launch vehicles because a launch vehicle failure would not destroy the payload if it occurs on a vehicle that is not carrying the payload and because additional pairs of vehicles might be launched so that even if one had to drop out of the flock because of a malfunction the launch might still be carried out using a spare vehicle. Assuming a cargo value of $10,000 per pound and a chance of a vehicle failure during the launch of 1%, the cost of risk was estimated to be about $20 per pound of payload. This low probability of failure assumes that bringing two partially-fueled rocket vehicles together and docking them during ascent can be done routinely. If the binary staging maneuvers cannot be made safe enough to be carried out routinely then the FLOC concept will fail for reasons other than just economics. Propellant costs were estimated at about $4 per pound of payload, which is so small as to be within the error of other cost estimates in this section. Using a labor cost of $100 per hour, the specific labor cost would be $(2090**L*) per pound of payload where *L* is labor intensity in units of hours per pound of structure. The lowest current launch vehicle labor intensities are around 1 hr/lb, which would give a huge specific labor cost for FLOC of $2090 per pound of payload because of the overall high structure-payload mass ratio of the 32 unit flock. One of the primary goals Novatia had for FLOC was to improve vehicle operations cost at the expense of a higher structure mass fraction. Assuming that in using the relatively high (for a rocket) structure mass fraction of 29%, Novatia succeeded in reducing the required labor intensity by an order of magnitude the resulting specific operations cost would be $209 per pound of payload.

Cost	*Value*
Amortized Vehicle Development	*$481/lb.*
Vehicle Hardware	*$209/lb.*
Operations	*$209/lb.*
Risk	*$20/lb.*
Propellant	*$4/lb*
TOTAL	*$923/lb.*

Table 4. Estimate of 32 unit FLOC specific costs (rocket cost based)

The result of this rocket-based cost estimate, summarized in Table 4, shows that a cost to LEO of less than $1000 per pound of payload is probably achievable with FLOC if you neglect the cost of developing and demonstrating in-flight rocket refueling capability. This analysis also shows that the goal of radically reducing operations cost with a more robustly designed vehicle must be at least partially achieved to reach this. If operations labor intensity cannot be reduced below current best practices for space launch vehicles then the high structure-payload mass ratio of the concept will cause unacceptably high operation costs to overshadow any amortized development cost savings the system achieves.

B. Plane Economic Analysis

There is a shortage of historical data for estimating the developed structure cost of rocketplanes. The X-15 program cost[17] $676 per pound of vehicle structure in then-year dollars, which converts to approximately $4,300/lb. in current year dollars. Based on data from http://www.astronautix.com, Scaled Composite's SpaceShip One cost approximately $9,500 per pound of the rocketplane's structure mass. The X-15 data is without the engine development cost, however the baseline FLOC design was sized for use with an existing engine design. In the future, information from the X-37 program may provide additional useful cost data. Using the higher example of $9,500/lb. and the amortization factor of 27, amortized development cost was estimated at $229 per pound of payload delivered to LEO.

Cost	*Value*
Amortized Vehicle Development	*$229/lb.*
Vehicle Hardware	*$104/lb.*
Operations	*$21/lb.*
Risk	*$20/lb.*
Propellant	*$4/lb*
TOTAL	*$378/lb.*

Table 5. Estimate of 32 unit FLOC specific costs (airplane cost based)

Using a typical aircraft hardware cost estimate[18] of $500 per pound of structure, the specific cost of hardware is estimated at $104/lb. Using high-end estimates[18] of supersonic aircraft labor intensity of 0.01 hrs/lb. provides an operations cost of only $21/lb. Propellant and risk cost estimates were left the same in both rocket and plane cost estimate methods. The result of this airplane-based cost estimate, summarized in Table 5, shows a cost of less than $400 per pound of payload is probably achievable if you neglect the cost of developing and demonstrating in-flight rocket refueling capability. The key unknown factors that might alter this cost estimate are the amortization factor, which might be much higher than 27 considering the versatility of the design, and the fraction of the vehicle expended per flight, which might be much lower considering the airplane-like structural mass fraction of the vehicle. If both of these unknowns were more favorable than the conservative estimates used in this analysis then specific costs around $100/lb. to LEO might be achievable with the baseline FLOC launch system in a 32 unit configuration.

The disparity between these two cost estimates brings up the question of whether a FLOC rocketplane would be more like a rocket or more like a plane in cost. FLOC would need to be capable of space operations and re-entry, which are not airplane-like features. On the other hand, Table 6 shows that in terms of structural mass fraction the FLOC baseline vehicle is closer to an airplane than a rocket, and might allow airplane-like design and manufacturing techniques to be used as Novatia hopes.

Table 6. Values of Strucure Mass/GLOW or Structure Mass/MTOW

Vehicle	M_s/M_{tot}
AtlasV + CentaurIII	7%
Ariane 5	14%
FLOC Baseline	29%
KC-135	31%
B-58 Hustler	31%

VII. In-Flight Rocket Refueling and Alternatives

Whether or not the savings generated by separated ascent stage launch systems like FLOC can justify the cost of a development program to demonstrate in-flight rocket refueling depends on the technical difficulty of such a program and the economics of potential competing launch vehicle technology. In-flight rocket refueling may sound incredible at first, but both the Black Horse and FLOC teams took steps in their proposals to address the issue. Clapp et. al. also remind us that

> The idea of refueling an airplane in flight must have seemed bizarre to anyone witnessing the Wright Brother's first flights. By the 1920s it had been demonstrated and today it is done routinely. Doing it on the way to space, and in space itself, are just the next steps up the ladder[14].

The proposed Black Horse rocketplane-to-rocketplane propellant transfer was to be done with a refueling boom extended between the rocketplanes during about a six-minute exo-atmospheric portion of their suborbital flight. With the propellant transfer expected to take about two minutes, that left approximately four minutes to conduct an orbital rendezvous between two cooperative spacecraft traveling on nearly parallel trajectories in the same direction. This is a much easier task than the rendezvous with a non-cooperative target headed in the opposite direction that anti-ballistic missile missions require!

Novatia's second paper[15] on the FLOC concept proposes ways to reduce the technical difficulty of the binary staging maneuvers. One of these proposals is to adjust the FLOC trajectory so that, like the Black Horse proposal, all rocketplane rendezvous take place outside the atmosphere in order to eliminate aerodynamic interactions between the vehicles. Some of the Novatia concepts even eliminated fuel transfer between the vehicles all together. These concepts required the rocketplanes to launch separately and then pair up after leaving the atmosphere with one of the rocketplanes providing thrust. After it is empty, the booster rocketplane returns to Earth while the remaining rocketplanes pair up again. This process would continue until one of the vehicles had been put into orbit. Having only one of the pair firing its rockets reduces the number of engine restarts required to only two per rocketplane and eliminates the need for propellant transfer. Unlike the Black Horse concept, which relied on a refueling boom, these FLOC proposals require the rocketplanes to dock securely with each other.

VIII. Conclusion

The use of separated ascent stage launch vehicles, like the FLOC concept, first requires overcoming the difficult technical hurdle of in-flight refueling or in-flight docking of launch vehicles during ascent. If this technical hurdle could be overcome, however, separated ascent stage launch vehicles would have many characteristics beneficial for economical space access. The physical separation of component units would allow a large number of identical stages to be used without needing to account for complex interactions between the stages or designing the stages to function in a large number of configuration locations. Using a large number of identical stages would greatly reduce the developed structure-payload mass ratio and therefore the amortized development cost. It would also generate economies of scale that could reduce the recurring hardware and operations costs as well. The ability to add additional stages to a separated ascent stage launch system without developing additional hardware by increasing the size of the vehicle flock and redesigning the trajectory allows it to accommodate a wide variety of payload masses and missions with the same basic vehicle design. This versatility would further reduce amortized development costs by providing more missions to spread the development cost across, thus increasing the amortization factor. It could also reduce developed structure cost substantially by allowing higher structure mass fraction designs, that could be developed with more commercial-off-the-shelf parts and airplane-like development programs, to still function as practical space launch systems. Developed structure cost could be reduced even further by allowing a gradual flight test program of increasingly ambitious in-flight mating events instead of requiring the 'all at once' flight tests normally required for space launch rocket vehicles[17]. The reduction in developed structure cost would contribute further to low amortized development costs. Considering the potential savings that could be generated by separated ascent stage launch vehicles, serious analysis should be conducted on the possibility of in-flight refueling or docking of launch vehicles. It should not be dismissed as 'obviously' impractical. Even if the short time needed to complete the in-flight rendezvous and the trajectory limitations imposed by the need to minimize aerodynamic interference make co-operative multi-vehicle flocks impractical as launch vehicles, the concept might still be useful as a way to reduce the cost of future interplanetary (or even interstellar) missions where high delta-V's are also needed but the time and atmosphere problems would be absent.

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