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ABSTRACT

Traditionally mass estimation for conceptual design of advanced launch vehicles has depended on historically – based mass estimating relationships (MERs). Furthermore different organizations have different sets of MERs based on different data sets, and even formulated in different ways. This paper attempts to compare the modern MERs used in the Space Systems Design Lab (SSDL) at Georgia Tech to the 1960's era relationships used in the NAS7-377 report on advanced propulsion design for launch vehicles. Comparisons of the weight breakdowns of a two – stage – to – orbit vehicle are made for between the Marquardt equations and the SSDL equations using two different technology assumptions. The first assumes 1970 technology for a direct comparison of the equations while the second assumes 2015 technology. Additionally technology and material advances are estimated in an attempt to justify the lower weight of the 2015 technology. The SSDL model using 1970 technology weighs in 7% heavier than the Marquardt equations for a comparable two – stage – to – orbit vehicle. When 2015 technology is applied to the same vehicle SSDL equations show a 33% savings, on the entire vehicle, could be made due to technology.

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NOMENCLATURE

AFRSI	Advanced Flexible Reusable Surface Insulation
GLOW	Gross Lift-Off Weight
Isp	Specific Impulse (sec.)
MER	Mass Estimating Relationship
NASA	National Aeronautics and Space Administration
OMS	Orbital Maneuvering System
RBCC	Rocket Based Combined Cycle
RCS	Reaction Control System
SSDL	Space Systems Design Lab
TPS	Thermal Protection System
TRF	Technology Reduction Factor
TSTO	Two – Stage – To – Oribt
TUFI	Toughened Uni-Piece Fibrous Insulation
WBS	Weight Breakdown Structure

INTRODUCTION

Mass estimation of advanced RBCC launch vehicles is a highly debated topic. Questions arise as to the method and accuracy with which the mass of a vehicle was predicted. Each conceptual design department has its own trusted method, and an inherent lack of trust in methods developed elsewhere. In order to shed more light on the topic of mass estimating relationships, an historic set of MERs were reproduced and verified. The verified equations were then compared to the modern equations used in the Space Systems Design Laboratory at the Georgia Institute of Technology (SSDL) using both 1970 technology and 2015 technology.

The chosen historic relationships were taken from "A Study of Composite Propulsion Systems for Advanced Launch Vehicle Applications" by The Marquardt Corporation, the Lockheed – California Company, and Rocketdyne [1] (the Marquardt report). The first stage weight equations used in the Marquardt report were originally developed by the Lockheed – California Company and published only internally. The second stage equations were taken from "Reusable Orbital Transport Second Summary Technical Report" by General Dynamics and Convair [2]. Many of the modern equations used in the SSDL have their heritage in those developed at NASA Langley's Vehicle Analysis Branch in the late 1980's and early 1990's.

To validate the reproduced equations a specific vehicle from the Marquardt report was modeled and compared to data provided in the report. For the first stage a full WBS was available, but for the second stage only the initial stage mass was presented. After validation, a new vehicle using a similar configuration was modeled with both the relations from the Marquardt report and the relations from the SSDL using two different technology assumptions.

REPRODUCING THE MERs

The equations for weight estimation presented in the Marquardt report were modeled in an Excel spreadsheet using iteration to converge a solution. The MERs for both stages were driven by an input mass ratio. In the report the vehicle GLOW was set at 1,000,000 lb. and the available payload mass was found. For the first stage model this meant that the upper stage was the remainder of 1,000,000 lb. less the initial mass of the first stage. Since the Marquardt report presents the payload that produces a 1,000,000 lb. vehicle, this was used as an input and the stage weight was found. This allowed the stage mass of the reproduced equations to be compared to the published stage mass.

1st Stage

The unit weights and fixed weights were presented for the first stage, including terms for manned missions. Since the first stage for both the validation and comparison vehicle were unmanned, some masses such as the cabin structure, personnel, instruments, and it was unclear how the remaining relationships would be affected for an unmanned first stage. For example, the electrical requirements of an unmanned vehicle are typically lower than for a manned vehicle, however, no adjustment for electrical system mass was presented for an unmanned system. The equipment and systems weight presents a large part of the error in the comparison of the first stage. The equation for the equipment and systems weight is:

$$W_{e\&s} = C_1 + C_2\, W_{1b/o} + C_3\, W_{1t/o} \qquad (1)$$

Where $W_{e\&s}$ is the equipment and systems weight, $W_{1b/o}$ is the first stage burnout weight, $W_{1t/o}$ is the first stage take-off weight, and C_i are the equation coefficients. This equation, among others, was modified during validation to achieve a better match between the reproduced equations and the published WBS.

Further detail is added to the first stage model by providing different mass fractions for structural and thermal components at two different staging Mach numbers. Different propellant fractions for landing and return to base propellant were provided for each engine type to account for differences in engine efficiency.

2nd Stage

The second stage equations in the Marquardt report were taken from a reusable orbital transport study [2] performed by General Dynamics and Convair. The equations were presented for only one vehicle configuration. It consisted of a manned lifting body system utilizing an aerospike engine burning hydrogen and oxygen operating at 455 seconds of Isp. Conversion to an unmanned system was straightforward since the passenger compartment and related equipment were considered part of the vehicle payload. One problem did arise due to an inconsistency in the equations for the second stage presented in ref. 1 Table XXVIII. In the individual detailed breakdown the weight of the rocket engine was shown to be:

$$W_{eng} = 0.146\ T_{stage} + 300\ \text{lb.} \quad (2)$$

Where W_{eng} is the rocket engine mass, and T_{stage} is the staging thrust of the engine. The equation for the propulsion system as a whole was:

$$W_{eng} = 0.0146\ T_{stage} + 300\ \text{lb.} \quad (3)$$

Equation 3 is a more reasonable number, producing an staging engine thrust to weight of 68.5, and is the listed thrust to weight in the original General Dynamics and Convair report [2, (vol. 1 page 56)].

VALIDATION OF REPRODUCED MERs

To provide validity to the reproduced MERs from the Marquardt report both the first and second stage vehicles were reproduced and compared to data provided. A complete WBS was available for the first stage for a variety of configurations and engines. Only one second stage vehicle was modeled in the Marquardt report, and no complete WBS was provided. The only comparison that could be made for the second stage was its initial weight.

1st Stage

The vehicle using an ejector ramjet engine with cylindrical body and wing was reproduced in order to verify the first stage relationships. A mass ratio of 1.5456 was required to reach the staging point of Mach 8. General parameters for the vehicle are shown in Table 1.

When this vehicle was modeled using the exact relationships and coefficients provided in the Marquardt report, the results were not good. For example, using the provided coefficients, Equation 1 becomes:

Table 1: 1st stage vehicle parameters.

Heat shield area	10,930 ft^2
Insulation area	10,930 ft^2
Aft structure area	3,630 ft^2
Vertical fin area	1,100 ft^2
Wing planform area	9,860 ft^2
T/W at takeoff	1.25
Inlet area	275 ft^2

$$W_{e\&s} = 2530 + 0.0135\ W_{1b/o} + 0.0192\ W_{1t/o} \quad (4)$$

Using the provided coefficients the gross weight of the vehicle was within 2.7% of the initial stage mass, but a discrepancy of up to 25% was found in the component level of the WBS. A 25% error in any part of the relationships requires a look into the sources of error, possibly caused by one or more of the following reasons. Modification over time of the MERs is not uncommon and the chosen WBS (Table XXIX from ref. 1) for comparison may not correspond in time to the coefficients published. This WBS was chosen because it contained the most detail of any presented. Further cause may be due to a difference in the method of convergence. The Marquardt report used hand iteration which tends towards less accuracy and greater potential for errors than the computer based method used for this paper. Another reason, listed above, was the lack of presented adjustment for an unmanned system. The equations presented for the first stage were for a manned system, while the vehicles designed and presented with complete WBSs were for unmanned systems. Furthermore, several sets of numbers were presented in multiple locations within the Marquardt report, and often the numbers conflicted with each other.

Due to the large component level discrepancies the coefficients were adjusted to produce a better match. For example Equation 4 now looks like:

$$W_{e\&s} = 2530 + 0.0114\ W_{1b/o} + 0.0162\ W_{1t/o} \quad (5)$$

The remainder of the modified coefficients are shown in Table 2. This method achieved a 0.05% error in the initial stage mass, and a more reasonable maximum component error of 0.6% in the landing and entry propellants. The

Table 2: Original and modified coefficients.

	Original	Modified
Inlets (lb/ft^2)	175	170
Engine T/W	31.4	33.6
LOX tank (lb/lb)	0.0255	0.0276
LH2 tank (lb/lb)	0.2	0.1547
Tail (lb/ft^2)	7.68	8.227

reproduced WBS for both the modified and unmodified coefficients is presented in Table 3.

2nd Stage

The second stage was launched from a piggy-back position on the first stage and required a mass ratio of 2.3240 to propel itself and payload due East from Mach 8 into a 262 nautical mile circular orbit. General parameters for the second stage vehicle are shown in Table 4.

The relationships provided in the Marquardt report produced a very good match to the GLOW using the coefficients presented in the detailed

Table 3: Comparison of Marquardt WBS to the reproduced relationships. All weights in lbs.

Item	Provided coeffs.	Adjusted coeffs.	Marquardt WBS
Wing	97,091	97,091	97,100
Tail	8,448	9,050	9,050
Body	50,260	41,156	41,175
TPS	27,216	27,216	27,210
Landing Gear	35,700	35,700	35,700
Main Propulsion	110,996	107,015	107,050
Systems & Equipment	28,823	23,049	23,150
Dry Weight	**358,535**	**340,276**	**340,435**
Residual Propellant	10,637	10,313	10,330
Landed Weight	**369,173**	**350,589**	**350,765**
Entry/Land Propellant	30,341	28,814	29,000
Burnout Weight	**399,514**	**379,403**	**379,765**
Ascent Propellant	353,002	353,002	353,000
1st Stage Weight	**752,516**	**732,405**	**732,765**

Table 4: 2nd stage vehicle parameters.

Plan area	550 ft^2
Wetted area	1,435 ft^2
Thrust at staging	350,000 lbf.
Contingency factor	3%

mass breakdown. The Marquardt report cites a second stage mass of 267,225 lb., corresponding to the chosen first stage WBS payload. The reproduced equations for the second stage used an engine thrust to weight of 68.5 and produced a mass of 110,858 lb., less than half that quoted in the Marquardt report. However, for an engine thrust to weight of 6.85, the reproduced equations produced a mass of 267,187 lb., within 1.0% on the stage weight. This clearly shows that the Marquardt data is based upon the wrong engine thrust to weight. With this in mind it would be nice to know what the actual payload capability of the ejector ramjet vehicle is. In order to maintain the size of the first stage, the payload of the second stage must be increased until the second stage mass reaches 267,225 lb. This point is reached at a cargo mass (not including the personnel, cabin, and related equipment) of roughly 52,500 lb. as opposed to the original 6,615 lb., a 790% increase in payload performance. No component level comparison was made since a detailed WBS was not presented for the second stage.

COMPARISON OF MARQUARDT AND SSDL MERs

Once the equations were shown to accurately represent the original vehicle the sizing parameters were modified to match *Cerberus*, shown in Figure 1, a previously designed TSTO vehicle with a similar layout to the Marquardt winged vehicle used above. Both vehicles have an RBCC ESJ first stage, rocket second stage, and burn liquid oxygen and liquid hydrogen in both stages. Resizing the vehicle for a different payload involves iterating between trajectory and mass properties, where a photographic scaling of the vehicle is used to change the mass ratio to match that required by trajectory simulation. Furthermore both have winged first stages and lifting body second stages. *Cerberus* was used as

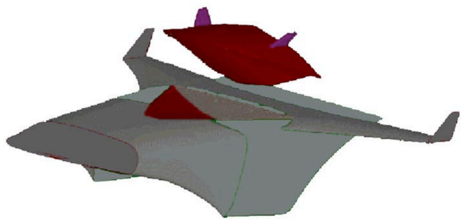

Figure 1: Cerberus *vehicle at separation.*

an example vehicle in [3], but scaled to carry a different payload. *Cerberus* was originally designed for a class at the Georgia Institute of Technology to put a 25,000 lb. payload into a 100 nautical mile circular orbit. The engine thrust to weight of 68.5 was used for the second stage comparison.

In order to compare both sets of MERs, the vehicle parameters and the mission must be identical. To meet this need the orbital maneuvering system propellant was increased such that Cerberus could reach a 262 nautical mile circular orbit. The Marquardt MERs were modified to eliminate the personnel and related equipment from the second stage equations. The payload was also increased to 25,000 lb. Cerberus uses an ejector scramjet engine, so the thrust to weight of the engines was modified according to Marquardt's extensive engine data.

Cerberus was modeled with the SSDL equations using two different sets of technologies. The first was an approximation of what would be available in 2015. The second was based on technology from the Space Shuttle, available in 1970, approximately the same time as the equations presented in the Marquardt report were intended to model. Table 5 shows the key differences between the technologies used.

Table 5: Comparison of 1970 and 2015 technologies.

	1970	2015
LH2 tank (lb/ft^3)	0.559	0.26
LOX tank (lb/ft^3)	0.808	0.33
Landing Gear (% W_{gross})	3.3%	2.6%
Body (lb/ft^2)	3.4	2.38
TPS (TRF)	0%	30%
Wing (TRF)	0%	40%
Tail (TRF)	0%	20%
Engine T/W (SLS, inst.)	13.4	22.86

1st Stage Comparison

The first stage comparison makes use of the corrected coefficients found in the validation section, and eliminates the fixed gross vehicle weight. The first stage payload is now determined by the mass of the second stage. The Cerberus first stage was modeled using the input parameters shown in Table 6. The stage initial weight using 1970 technology was 885,277 lb while the Marquardt equations placed the stage weight at 800,767 lb., 9% lighter. Using 2015 technology the stage weighed in at 532,176 lb., 39% lighter than Marquardt's equations. A WBS displaying the key items is shown in Table 7.

Of particular interest is the difference in wing mass between the models. The Marquardt method uses a multiplication factor of 9.847 lb/ft^2 on the planform area whereas the SSDL uses a more complex model, taking into account the thickness and span. The SSDL model is based on a regression of fighter wing structures from NASA Langley. For the area of the Cerberus first stage wing the Marquardt method produces a weight 54% lighter than the 1970 SSDL model. By changing the material and technology used in the wing to 2015 levels, the SSDL model is now 31% lighter than the Marquardt model. Possible contributions to the overall mass difference include the detail of the model, material changes, and differences in assumed technology. Simply switching from aluminum to the titanium aluminide used in the 2015 model nets a 40%

Table 6: Cerberus 1st stage parameters.

Heat shield area	7,600 ft^2
Insulation area	8,106 ft^2
Aft structure area	2,891 ft^2
Vertical fin area	137 ft^2
Wing area	2,200 ft^2
T/W at takeoff	0.6
Inlet area	151 ft^2
Engine T/W (SLS, installed)	29.0
Mass Ratio	1.8983

Table 7: Comparison of Marquardt and SSDL WBS for Cerberus *1st stage. All weights in lb.*

Item	SSDL 2015	SSDL 1970	Marquardt MERs
Wing	14,914	47,224	21,663
Tail	1,004	1,690	1,127
Body	45,291	109,417	77,670
TPS	12,021	20,331	19,465
Landing Gear	26,433	50,631	41,074
Main Propulsion	20,300	31,660	36,540
Systems & Equipment	11,940	14,190	24,085
Dry Weight *	**131,903**	**275,146**	**221,625**
Residual Propellant	1,986	3,084	15,231
Landed Weight	**133,913**	**278,230**	**236,857**
Entry/Landing Propellant	11,549	17,738	19,466
Post-Staging Weight	**145,463**	**295,968**	**256,323**
Ascent Reserves	1,225	1,866	0
Burnout Weight	**145,463**	**297,834**	**256,323**
Ascent Propellant	383,391	584,277	544,444
GLOW	**811,252**	**882,111**	**800,767**
Startup Losses	2,098	3,166	0
1st Stage Weight	**532,176**	**885,277**	**800,767**

* Dry weight margin is included in individual weights.

improvement in weight due to the increased modulus, and hence increased buckling load per mass. The external tank redesign for the Space Shuttle netted a 16% decrease in mass due solely to design improvement [4] between the original tank and the super light weight external tank. These two reductions easily make up the difference between the 2015 SSDL technology and Marquardt methods, however the 1970 SSDL method produces a much heavier wing than does the Marquardt method. The Marquardt method simply uses a mass per area for the wing, which works well for small perturbations around the design point, but not for large changes. Due to structural considerations the more the geometry changes the worse an assumption this is.

The differences in the body group masses are primarily due to the detail of the model. Tank weights are in good agreement with each other, at most 13% different. The primary difference is in non-integral structure that is included in the SSDL equations, but not those presented by Marquardt. The body group contains the engine inlets, whose weight is the difference between the installed and uninstalled engine thrust to weight. Marquardt calculates that this difference is 18.8 while the 2015 technology assumes the difference is 7.1. The 1970 technology finds the difference to be 15.6, producing an inlet weight within 17% of the Marquardt number. Using current thinking in RBCC engine development yields a weight 80% lighter than the Marquardt inlets for the 2015 vehicle.

The thermal protection group also shows a large difference between the two models. The Marquardt MERs produce a weight 102% greater than the 2015 SSDL weight. This is largely due to the material assumptions for the TPS. Marquardt assumes a shingled surface of either columbium or inconel with an underlying layer of felted ceramic for the shielded area at 1.42 lb/ft^2. The SSDL assumes reusable TUFI tiles for areas requiring a heat shield. For a tile with similar thermal conductivity [5] to the Marquardt combination, the TUFI tiles yield 22% weight savings. Following the same reasoning, the use of AFRSI shows a 70% weight savings over fiberfax insulation used by Marquardt. The Marquardt TPS weight is just 4% lighter than the 1970 SSDL weight. This difference may be due to the trajectory flown, or the vehicle geometry. Updating to 2015 technology yields a 38% weight savings.

The propulsion system also shows a large discrepancy between the SSDL and Marquardt MERs. Between equivalent technology levels the SSDL method weighs 13% less than the Marquardt method. The 1970 SSDL method uses an installed thrust to weight of 13.4 [1 vol. 6]. The SSDL uses regressed historical data with technology adjustments from the WATES [6] model to determine the engine thrust to weight. The 2015 technology uses an installed thrust to weight of 22.86, which is consistent with a current engine developer's thinking [7]. This difference represents the different thinking over the past 30 years in RBCC engines. The 2015 technology shows a 44% lower weight than the Marquardt equations.

Systems and equipment mass shows a 152% decrease in mass to the 2015 SSDL weight, and 112% decrease to the 1970 SSDL weight. A large difference between the Marquardt and the 2015 weight was expected since it includes avionics, environmental control, electrical, and actuation mechanisms, fields in which large weight savings have been made in recent years. Further weight differences may be due to a change in design philosophy and system redundancy.

The dry weight margin is one point on which the two sets of equations differ. Marquardt chooses to include a margin in each of their weight equations, whereas the SSDL adds a lumped mass to the dry weight. For this comparison the dry weight margin was distributed over the component masses. For the overall vehicle mass, this makes no difference, but from a management standpoint it may be nice to have the margin lumped into one number so it can be doled out to designers as the need arises.

2nd Stage Comparison

Cerberus' second stage was modeled using the sizing parameters shown in Table 8 in using the same three methods as for the first stage.

Cerberus' second stage had a separation weight of 281,174 lb. when assuming 2015 technology while the Marquardt equations showed it weighing in at 349,759 lb., nearly 24% heavier. The 1970 SSDL weight is within 2% of the Marquardt weight. Table 9 compares the two methods in detail.

Table 8: 2nd stage vehicle parameters.

Plan area	2,465 ft^2
Wetted area	5,294 ft^2
Vehicle Thrust to Weight	1.1
Contingency factor	3%
Mass Ratio	3.3852

The second stage also has several items in the WBS that differ significantly between the two different sets of MERs. The first example is the tail weight. Here the SSDL method uses:

$$W_{tail} = 5\, S_{tail}^{1.09} \qquad (6)$$

Where W_{tail} is the tail weight, and S_{tail} is the tail planform. Marquardt counts the tail as a fraction of the plan area:

$$W_{tail} = 6.8\,(0.2)\, S_{plan} \qquad (7)$$

Here S_{plan} is the total planform area of the vehicle. This method does not take into account the specific vehicle geometry or other analysis. This leaves the SSDL tail significantly lighter for both technology dates.

The body group presents a difference of nearly 38% between the 2015 SSDL weight and the Marquardt weight. This weight difference can be largely made up by structural improvements in the body and tanks. The space shuttle showed an improvement of 25% in the external tank. An additional 15% improvement can be made by switching from aluminum to carbon for structural applications based on increased material strength per mass. The 1970 SSDL weight came in just 11% lighter than did Marquardt's weight. The 1970 SSDL method uses the tankage factors from the Space Shuttle external tank, which is a load bearing structure, and cylindrical. The second stage in this study is a lifting body using non-integral tanks, which do not package well. This causes a loss of packaging efficiency and a corresponding increase in mass.

The 2015 SSDL method shows the TPS group to be 61% lighter than the Marquardt method. The TPS material substitutions from the first stage can also apply to the second stage, showing that the SSDL was aggressive with the TPS, or Marquardt was conservative. Again, the exact trajectory followed and vehicle geometry make a large difference on heat shield weight.

When the RCS and OMS propulsion systems the are combined the systems weigh nearly the same. The 1970 SSDL weight is slightly higher, likely due to Isp assumptions of the RCS and OMS thrusters. The remaining points of interest lie in the electrical conversion and distribution, avionics, and primary power. For the avionics the Marquardt equations were derived for a manned system which requires less computation than a fully autonomous system. The SSDL method produces roughly double the weight for both the electrical and primary power groups as does the Marquardt method. This is particularly odd since Marquardt equations assume dual redundancy [2, (page 45)] of those components. No good explanation was found for this discrepancy.

Table 9: Comparison of Marquardt and SSDL WBS for Cerberus *2nd stage. All weights in lb.*

Item	SSDL 2015	SSDL 1970	Marquardt MERs
Wing	0	0	0
Tail	944	1,180	7,691
Body	18,908	26,968	30,296
TPS	3,143	4,498	8,186
Landing Gear	2,235	2,991	3,798
Main Propulsion	6,435	7,831	11,613
RCS Propulsion	1,045	1,271	2,115
OMS Propulsion	1,089	1,326	0
Primary Power	946	950	831
Electrical	2,207	2,938	800
Surface Control	365	444	961
Avionics	1,000	1,000	563
Environmental Control	1,239	1,254	464
Margin	7,911	10,530	10,493
Dry Weight	**47,483**	**63,180**	**77,811**
Cargo	25,000	25,000	25,000
Residual Propellant	1,333	1,621	5,554
OMS/RCS Reserve Propellant	684	832	1,232
Landed Weight	**74,483**	**90,634**	**109,598**
Entry/Landing Propellant	138	168	0
Entry Weight	**74,621**	**90,801**	**109,598**
OMS/RCS on-orbit	6,702	8,155	0
Ascent Reserve	1,737	2,113	0
Insertion Weight	**83,060**	**90,801**	**109,598**
Ascent Propellant	198,114	241,072	240,161
2nd Stage Weight	**281,174**	**342,142**	**349,759**

CONCLUSIONS

This paper reproduces and compares Marquardt's historic mass estimating relationships for a TSTO vehicle using technology from 1970 to relationships used today with two different sets of TRFs, 1970 and 2015. Table 10 shows the gross vehicle weights of the compared vehicles. Comparison found that the SSDL equations using 1970 technology produce a vehicle just 6.7% heavier than the Marquardt equations. Using technology projected out to 2015 leaves the Marquardt vehicle 33% heavier than the futuristic SSDL vehicle.

Overall the Marquardt and SSDL equations produce very similar results if the same technology is assumed. Some of the technological and materials advances can be easily approximated in a simple manner, but only flight hardware will truly tell which prediction is correct.

Table 10: Total vehicle mass of compared vehicles. All weights in pounds.

1st Stage Weight	**532,176**	**885,277**	**800,767**
2nd Stage Weight	**281,174**	**342,142**	**349,759**
Gross Vehicle Weight	**813,350**	**1,227,419**	**1,150,526**

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