Development of a Mass Estimating Relationship Database for Launch Vehicle Conceptual Design

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Abstract

This report attempts to bring mass estimating relations (MERs) for the conceptual design of launch vehicles into the open, and establish a baseline for their comparison. Data was taken from multiple design organizations from around the country and compiled into a database that is freely available for use. To validate the equations, Space Shuttle component masses were predicted. A percentage error was reported, with the sign indicating the direction of the error. No single set of MERs is uniformly more accurate than another. To improve the utility of the equations, modifications can be made to the equations to model improved technologies, such as those used in advanced launch vehicles. Technology reduction factors are also compiled from multiple sources. No proof of their accuracy is available at this time. The greatest accuracy in predicting the mass of a future launch vehicle would be attained by using the most accurate equation for each component, and an appropriate technology reduction factor.

Acronyms & Notation

AMLS	Advanced Manned Launch System
AVID	Aerospace Vehicle Interactive Design system
EMA	Electro-Mechanical Actuator
ET	External Tank from Space Shuttle System
IHOT	Integrated Hydrogen Oxygen Technology
LaRC	Langley Research Center
LOX	Liquid Oxygen
LH2	Liquid Hydrogen
MBS	Mass Breakdown Structure
MER	Mass Estimating Relation
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NASP	National Aero Space Plane
OMS	Orbital Maneuvering System
RBCC	Rocket Based Combined Cycle
RCC	Reinforced Carbon-Carbon TPS
RCS	Reaction Control System
SRB	Solid Rocket Booster from Space Shuttle System
SSTO	Single Stage To Orbit
TRF	Technology Reduction Factor
TSTO	Two Stage To Orbit

I Introduction

Estimating the mass of future launch vehicles is typically done using parametric equations for each component of the vehicle. While effective, and fast, this method is not perfect. Many design organizations have their own equations, and do not trust equations from other sources. This paper attempts to solve this problem by making mass estimating relations freely available to the design community. Further, the Space Shuttle system is used as a reference point to validate the equations. It turns out there is no single set of mass estimating relations (MER) that is most accurate. The highest accuracy would be gained by taking the best MER for each component, from multiple sources. Additionally, technology reduction factors are supplied to enable designers to model future vehicles using equations derived from current and past technology.

II Background

Mass estimation of future air and space vehicles is typically done using parameterized equations for each component of a vehicle. These equations are then summed to find the total mass of the vehicle. For example, the mass of the anti-vortex baffles in a propellant tank, according to Brothers, is given by:

$$M_{antivortex} = \frac{\dot{m}F_{prop}}{\rho}\left(0.64 + 0.0184\rho\right)$$

Here the mass of the baffles is a function of propellant density, and mass flow rate from the tank. There is not a unique set of parameters to base the mass of the anti-vortex baffles on, and different equations use different parameters. Often a minor component, such as the anti-vortex baffles, may be included in another equation for a larger component, such as the tank mass. Due to the many ways to parameterize a vehicle component, and the available levels of detail that the vehicle can be broken into, different design organizations often have different equations to model launch vehicles. This process works, but contains flaws.

The largest problem with the currently used system is the lack of data available on space vehicles. In particular, there is only one data point for reusable launch vehicles, and none for air-breathing launch vehicles. This problem is often remedied by fitting curves to aircraft components and then shifting the intercept such that data from the Space Shuttle lies on the curve, as is done by Brothers, and MacConochie. Brothers also fits curves to a combination of expendable launch vehicles and the Space Shuttle. This ensures that all data is for space hardware, but the durability, and hence weight of components is lower for expendable vehicles. The lack of data is exacerbated by the fact that all data is not available to all design organizations. Hence each organization's in-house MERs are based on different data points. All of these methods work, but for different vehicle configurations, and over different parameter ranges. More often that not, the valid range of the parameters is unpublished and often unknown since no data points exist for comparison beyond values of current space vehicles or aircraft.

A further flaw with this approach is the consistency between the mass predictions of different organization's MERs. If one design organization uses their in-house equations for a new vehicle, and a second organization uses their in-house equations for the same vehicle, will they get the same answer? This flaw is inspired by the difficulty in comparing ideas generated at different design organizations. If two different ideas for a launch vehicle are posed and one is lighter, it is typically labeled as the better design. This could actually be the case, or one of the design organizations may be using mass estimating relationships that are heavier (or lighter) than the other organization, producing an invalid comparison of the vehicle concepts.

III Approach

This paper attempts to solve the problem of comparing vehicles through a two pronged approach. First a database of MERs was created to make a large number of equations available, and second a baseline was used to compare the predicted mass of the equations to a flight vehicle.

By providing a database of equations to the conceptual design community a common set of equations will be available to all design organizations. If the same equations are used for vehicle design at different organizations, then the results should be easy to compare.

Even if different equations are used from the database, they can be referenced, and the difference between the equations used can be found.

By comparing the compiled mass estimating equations to a baseline vehicle the validity of the equation is verified against an actual flight vehicle. The chosen reference is the Space Shuttle, specifically orbital vehicle 103 circa 1983, and external tank 7 on a due East mission [i]. Many equations in the database are not intended to model Space Shuttle technology, and are not compared.

Several organizations have provided equations for this database, and in the future users should be encouraged to submit their equations with applicable parameter ranges for inclusion in the database. The database is presented in subsequent sections of this paper. The equations were compiled from multiple sources of data, many of which are unpublished. A description of each primary source (and a sub source if cited) is provided below. On a macro level the data is organized in the order of a typical mass breakdown structure. Within each group in the MBS there are two columns, and a page for each source. The first page of each component group contains the variables used to predict the mass of components in that group, and any supporting illustrations. On each subsequent page, the reference is listed along with a brief description of the data source. The first column under each reference contains the equations and applicable parameters and known limitations. The second column is a percentage error from the Space Shuttle. In the following equation E is the percent error from the Space Shuttle, M_i is the mass predicted by the MER, and $M_{shuttle}$ is the corresponding Space Shuttle component mass.

$$E = \frac{M_i - M_{shuttle}}{M_{shuttle}}$$

A positive error percentage indicates that the equation produces a mass higher than that of the Space Shuttle, and a negative error shows an equation that predicts lighter than the Space Shuttle. All equations in the database are set up for use in the English unit system. Standard measures for this database are feet, pounds, and seconds, with pressure in psi, and power in kilowatts, unless otherwise noted.

Equations that predict this vehicle accurately are likely only good for near term technology without adjustment. This adjustment is provided in the form of a technology reduction factor. Provided the trend of the equation is correct, the mass can be reduced by a percentage to represent an improvement in material technology. The mass of a component using improved technology can be found by the following equation:

$$M_{improved} = M_{original}\,(1 - TRF)$$

Here $M_{improved}$ is the mass of the component being modeled using improved technology, $M_{original}$ is the mass of that component predicted by an MER for current technology, and *TRF* is the appropriate technology reduction factor from the last section of this paper (starting on page TRF-1). This technique allows the use of MERs created using current technology to approximate what can be done in the future. In essence, this extends the useful life of an MER.

IV Description of Sources

Each source of MERs is intended to model a different type of vehicle, or has been derived from a particular configuration. This helps decipher the applicable range of the equations, and the vehicle configuration that they will model best. This description attempts to make available to the database user some of this knowledge so that the equations provided can be used in their proper context, and with confidence.

1. I.O. MacConochie and P.J. Klich [ii]

    MacConochie worked in the Vehicle Analysis Branch at the Langley Research Center in Hampton Virginia. These equations are from NASA Technical Memorandum 78661, published in 1978. This predates the Space Shuttle first flight, but makes use of known shuttle subsystem masses. Equations are based on commercial and fighter aircraft data.

2. Dr. John R. Olds [iii]

    Dr. Olds published these equations in his PhD dissertation in 1993. They are primarily a collection of equations from sources at NASA Langley Research Center, with a few that he created himself. The equations were used for design of a vertical take off horizontal landing RBCC SSTO vehicle, shown in Figure 1. Projects and authors of the original source are listed in the database where available.

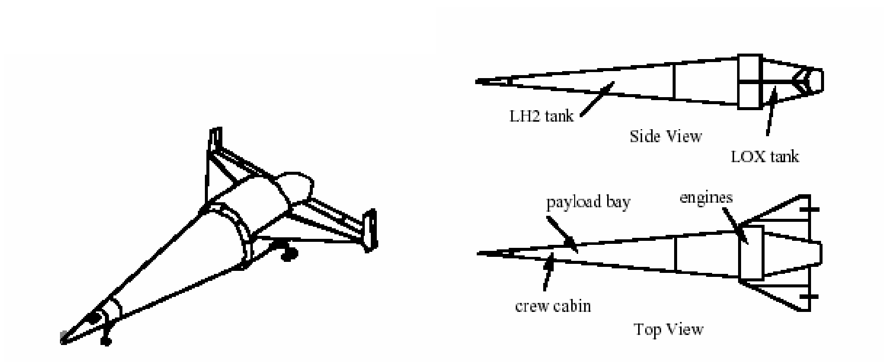


Figure 1: RBCC SSTO vehicle. [ref. 2]

3. Dr. Ted Talay

   Dr. Talay worked in the Vehicle Analysis Branch at NASA Langley Research Center. These equations were handed out as class notes for ME250, Launch Vehicle Design, at George Washington University in 1992, which he taught. Their emphasis is on rocket powered vehicles. Many of the equations provided are based on the Space Shuttle.

4. Marquardt report NAS7-377 [iv]

   a. These equations are from a report by The Marquardt Corporation in 1966**.** They are published in NAS7-377, a study of composite propulsion systems on launch vehicle mass. The study vehicle is TSTO, and takes off horizontally. The first stage uses the composite propulsion system on multiple body configurations. The lifting body version of the first stage can be seen in Figure 2 with the second stage attached. Conical and cylindrical body versions were also modeled with these equations.

b. The second stage is a rocket powered lifting body, based on a previously designed second stage by General Dynamics and Convair. These equations originate from report GD/C-DCB-65-018 [v].

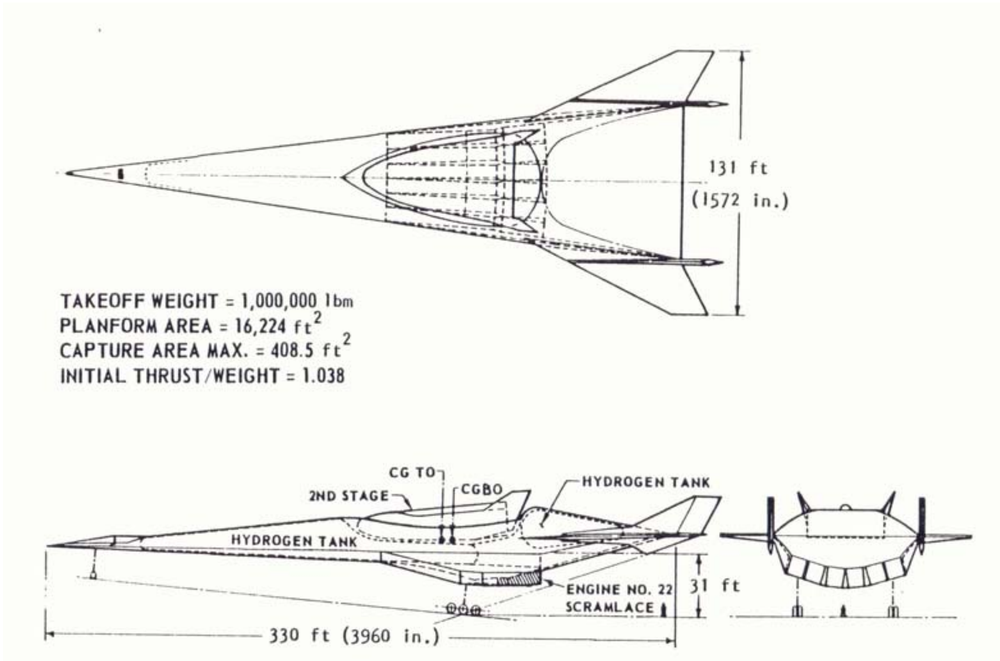


Figure 2: TSTO composite propulsion first stage with rocket powered lifting body second stage nestled on top. [ref. 4]

5. Aircraft Design: A Conceptual Approach, Daniel P. Raymer [vi]

   As the title implies, equations from Raymer are intended for use on aircraft. Only his equations for fighter/attack aircraft are provided in this database since they are subject to high speeds and similar redundancy requirements as space vehicles.

6. Bobby Brothers

   Brothers' equations are derived primarily from expendable vehicles and the Space Shuttle. He provides the most extensive set of equations, including multiple equations for many components, and careful delineation of parts based on their function and load in a vehicle. Some equations are taken from AVID, a sizing code developed by A. W. Wilhite at NASA Langley Research Center. When applicable, he also uses aircraft derived equations.

7. Airplane Design, Dr. Jan Roskam [vii]

   Roskam's focus is on aircraft, including everything from single propeller planes to fighter jets. For this database, only jet vehicles were considered, and almost exclusively fighter aircraft.

8. Forbis and Kotker, The Boeing Company [viii]

   This paper is aimed at the design of hypersonic aerospace vehicles. The only portion of this paper used is for landing gear weight.

9. Forbis and Woodhead, The Boeing Company [ix]

   This paper is also aimed at hypersonic aerospace vehicle analysis, and it appears to be an extension of the work done in the other listed paper by Forbis. Only the landing gear weight is used from this source.

10. AC-Sizer, NASA MSFC

    This data was taken from a spreadsheet sizing program written by D. R. Komar and company at NASA Marshall Spaceflight Center. Both rocket and air-breathing vehicles are provided. The primary use is for modeling future technology vehicles with wings, both air-breathing and rocket powered.

    a. Many of the equations provided are from Alpha Technologies' MER database. Alpha Technologies is run by Bobby Brothers, so many equations are derived from those in source 6, above.
    b. Wing MERs are from Boeing report AFWAL-TR-87-3056 on hypersonic aerospace vehicles.
    c. Landing gear is from report GDA-DCB-64-073.

11. Hawkins [x]

This source is focused purely on weight growth through the design cycle. The data presented is taken from a paper presented at a Society of Allied Weight Engineers Conference in Detroit Michigan, 23-25 May, 1988.

12. Dr. Ted Talay, NASA LaRC

This is from a presentation from the Space Systems Division at NASA Langley Research Center to Dave Pine, Code B at NASA Headquarters on June 10, 1993. It is titled "Effect of Concept Maturity on Weight Growth and Cost Estimation." Of primary interest is a chart showing dry weight growth on NASA space vehicle projects through the development cycle.

V Future Work

This database is only a start towards improving mass estimation for launch vehicles. In the future more equations need to be added as they become available. A second baseline point would also be very useful, especially a vehicle that uses current technologies, and has a different configuration than the Space Shuttle. This would allow verification of nearly all the equations provided in the database, and would lend some merit to the design of future vehicles. Further if an equation could predict the mass of both vehicles well, there would be improved confidence in the accuracy of the trend.

VI Acknowledgements

Due to the nature of the work, much of the data collected would not be available without the help of others. Mr. Bobby Brothers has been very helpful in providing data and information on the topic, and answering questions. D.R. Komar also has been generous in his contributions of equations to the database. Finally, Dr. John Olds has been instrumental in helping find data sources.

References

i MacConochie, I.O., “Shuttle Design Data and Mass Properties,” prepared under contract NAS1-19000, Lockheed Engineering and Sciences Company, April 1992.

ii MacConochie, I.O., Klich, P.J., “Techniques for the Determination of Mass Properties of Earth – To – Orbit Transportation Systems,” NASA Technical Memorandum 78661, June 1978.

iii Olds, J. R., "Multidisciplinary Design Techniques Applied to Conceptual Aerospace Vehicle Design," Ph.D. Dissertation, North Carolina State University, Raleigh, NC, June 1993.

iv Escher, W.J.D., Flornes, B.J., “A Study of Composite Propulsion Systems for Advanced Launch Vehicle Applications,” final report under NASA Contract NAS7-377, the Marquardt Corporation Report No. 25,294, September 1966 (7 volumes). [NASA Scientific and Technical Information Facility no. X67-19028/19034]

v Brady, J.F., Lynch, R.A., “Reusable Orbital Transport Second Summary Technical Report,” GD/C-DCB-65-018, Volume I, Contract NAS8-11463, April 1965. [NASA Scientific and Technical Information Facility no. 65X-17521]

vi Raymer, Daniel P. *Aircraft Design: A Conceptual Approach.* Washington D.C., American Institute of Aeronautics and Astronautics, 1999.

vii Roskam, Dr. Jan. *Airplane Design, Part V: Component Weight Estimation*. Ottawa, Kansas, 1989.

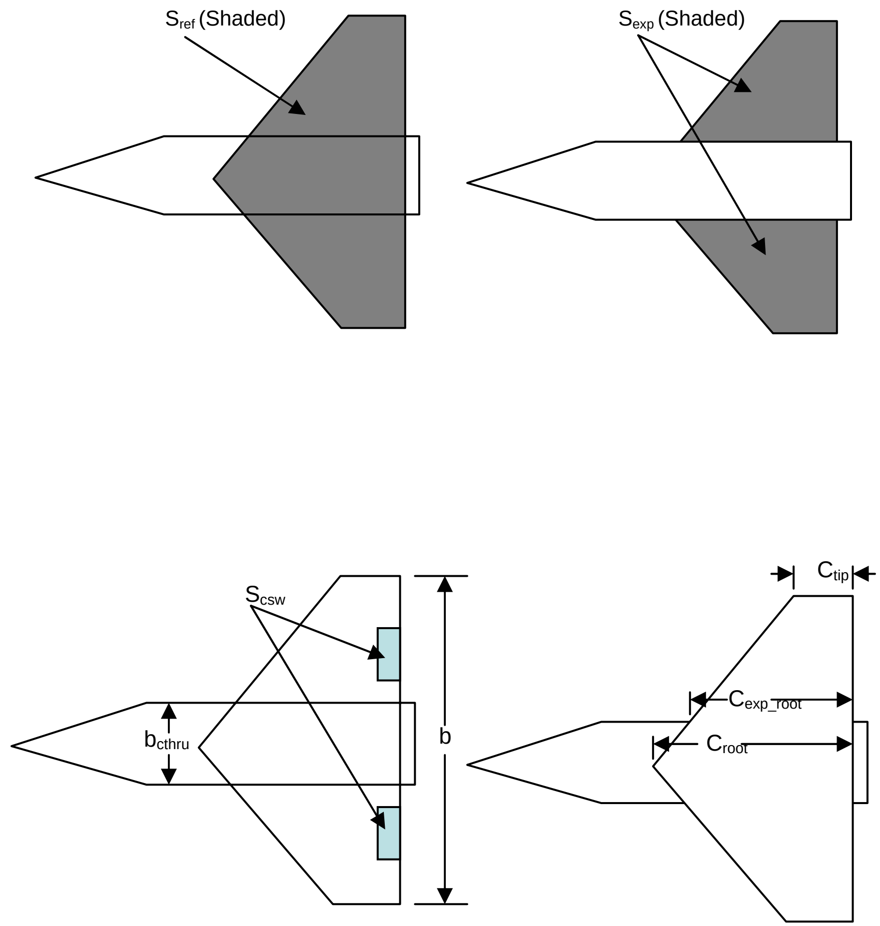
viii Forbis, J.C., Kotker, D.J., “Conceptual Design and Analysis of Hypervelocity Aerospace Vehicles, Volume 1 – Mass Properties,” final report for period June 1986 – March 1987, prepared under contract AFWAL-TR-87-3056 by the Boeing Aerospace Company of Seattle Washington, February 1988.

ix Forbis, J.C., Woodhead, G.E., “Conceptual Design and Analysis of Hypervelocity Aerospace Vehicles, Volume 1 – Mass Properties, Section 2 – Aerospace – Vehicle Mass Properties System,” final report for the period July 1988 – October 1990, prepared by Boeing Military Airplanes, Seattle Washington, October 1990.

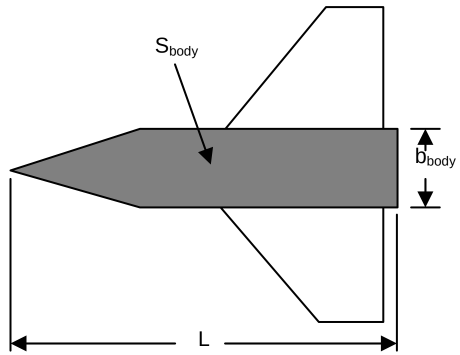
x Hawkins, K., “Space vehicle and Associated Subsystem Weight Growth,” SAWE paper 1816, May, 1988.

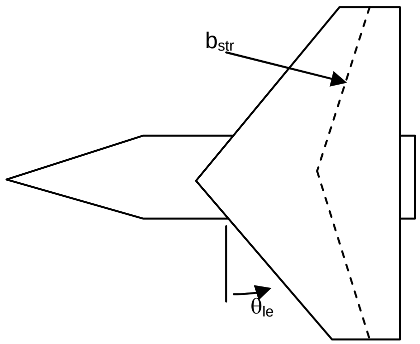
1.0 Wing

AR – Aspect ratio (b^2/S_{ref})
AR_{exp} – Exposed aspect ratio ($b_{exp}{}^2/S_{exp}$)
b – Wing span
b_{body} – Maximum width of the body
b_{exp} – Span of exposed wing (b-b_{body} at wing root)
b_{cthru} – Width of wing carry through
b_{str} – Wing structural span along the half chord line (picture)
c_{xx} – Wing chord at xx location
F_{safety} – Safety factor
M_{wing} – Mass of all components in wing group
M_{wing_exp} – Mass of exposed wing
M_{cthru} – Mass of wing carry thru structure
$M_{elevons}$ – Mass of elevons and attach structure
M_{land} – Landed mass of vehicle
M_{entry} – Entry mass of vehicle
M_{glow} – Gross liftoff mass of vehicle
M_{gross} – Gross vehicle mass on the pad or runway
N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load)
P_{exp} – Exposed wing planform loading (lb/ft^2)
q_{max} – Maximum dynamic pressure (lb/ft^2)
R_t – Taper ratio (c_{tip}/c_{root})
S_{body} – Planform area of the body
S_{csw} – Planform of wing mounted control surfaces
S_{exp} – Exposed wing planform area
$S_{fairing}$ – Surface area of wing fairing
S_{ref} – Theoretical wing planform area
$S_{strakes}$ – Planform area of wing strakes



$S_{TEextensions}$ – Planform area of trailing edge extensions
t_{xx} – Wing max thickness at xx location
TRF – Technology reduction factor
Λ – Wing sweep at 25% MAC
θ_{le} – Sweep angle of leading edge





<table>
<tr><td>Reference: 1
Derived from: Aircraft and Space Shuttle.
Options: Materials, and wing tanks.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>

$$M_{wing} = \left[N_z M_{land} \frac{1}{1+\eta\left(\frac{S_{body}}{S_{exp}}\right)} \right]^{0.386} \left(\frac{S_{exp}}{t_{root}}\right)^{0.572} \left[K_{wing} b_{str}^{\ 0.572} + K_{ct} b_{body}^{\ 0.572} \right]$$

Exposed wing material/configuration constants
K_{wing} = 0.286 – Aluminum skin/stringer, dry wing, no TPS
= 0.343 – same as above but wet wing for storable propellants
= 0.229 – metallic composite (Boron Aluminum) honeycomb dry wing, no TPS
= 0.263 – same as above but wet wing for storable propellant such as RP
= 0.214 – Organic composite honeycomb, no TPS
= 0.453 – Honeycomb dry wing super alloy hot structure, no TPS required

Wing carry-thru constants
K_{ct} = 0.0267 – dry carry-thru (integral)
= 0.0347 – wet carry-thru (integral)
= 0.100 – dry carry-thru (conventional)
= 0.120 – wet carry-thru (conventional)

Wing/body efficiency factor
η = 0.20 – for conventional vehicle to
= 0.15 – for control configured vehicle.
</td><td>2%</td></tr>
</table>

b_{body} – Maximum width of the body b_{str} – Wing structural span along the half chord line M_{land} – Landed mass of vehicle N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load) S_{body} – Planform area of the body S_{exp} – Exposed wing planform area t_{root} – Wing thickness at root	

<table>
<tr><td>Reference: 2
Derived for: Airbreathing horizontal takeoff vehicle.
Options: Wing material technology.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>

$$M_{wing-exp} = 0.82954\left[\frac{1+R_t}{t/c}\right]^{0.4}\left[\frac{N_z M_{land}}{1000}\right]^{0.48} S_{exp}^{0.67} AR_{exp}^{0.67}(1-TRF)$$

$$M_{cthru} = 0.00636\left[(1-R_t)AR_{exp}\right]^{0.5}\left[\frac{M_{land}N_z}{1000}\right]\left[\frac{b_{str}b_{body}}{t_{root}}\right](1-TRF)$$

TRF = 1.0 – for aluminum skin stringer construction
= 0.4 – for Ti3Al Beta 21S w/SiC

AR_{exp} – Exposed aspect ratio (b_{exp}^2/S_{exp})
b_{body} – Maximum width of the body
b_{str} – Wing structural span along the half chord line
M_{land} – Landed mass of vehicle
N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load)
R_t – Taper ratio (c_{tip}/c_{root})
S_{exp} – Exposed wing planform area
t_{root} – Wing thickness at root
(t/c) – Thickness to chord ratio on the wing</td><td>-13%</td></tr>
</table>

Reference: 3 Derived from: Dr. Talay, LaRC. Options: None.	Space Shuttle Comparison
$$M_{wing} = 2375\left[\frac{M_{entry} N_z b_{str} S_{ref}}{t_{root} \times 10^9}\right]^{0.584}$$ b_{str} – Wing structural span along the half chord line M_{entry} – Entry mass of vehicle N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load) S_{ref} – Theoretical wing planform area t_{root} – Wing thickness at root	43%

<table>
<tr><td>Reference: 4a
Derived from: Airbreathing booster.
Options: Maximum airbreathing Mach number.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{wing\_exp} = K_{wing} S_{\exp}$ Includes exposed wing and carry through

K_{wing} = 9.847 – for max airbreathing Mach number of 8?
S_{exp} – Exposed wing planform area

$M_{elevons} = K_{elevons} S_{csw}$ Mass of elevons using columbium, including hardware

$K_{elevons}$ = 11.51 – max airbreathing Mach number of 8
= 13.70 – max airbreathing Mach number of 12
S_{csw} – Planform of wing mounted control surfaces</td><td>N/A</td></tr>
</table>

Reference: 4b Derived from: Lifting body rocket. Options: None.	Space Shuttle Comparison
$M_{elevons} = 9.4(0.14S_{body}) + 0.07S_{body}$ Elevons and attachment for lifting body second stage. S_{body} – Planform area of the body	N/A

Reference: 5 Derived from: Aircraft. Options: Varying wing shapes.	Space Shuttle Comparison
$M_{wing} = 0.0103 K_{dw} K_{vs} \left(M_{gross} N_z\right)^{0.5} S_{ref}^{0.622} AR^{0.785} \left(t/c\right)_{root}^{-0.4} \left(1 + R_t\right)^{0.05} \left(\cos\Lambda\right)^{-1} S_{csw}^{0.04}$ Wing configuration factors K_{dw} = 0.768 – for delta wing = 1.0 – otherwise K_{vs} = 1.19 – for variable sweep = 1.0 – otherwise N_z here is the ultimate load factor = 1.5*limit load factor 1.5 is the typical factor of safety and the limit load factor is typically 2.5 AR – Aspect ratio (b^2/S_{ref}) M_{gross} – Gross vehicle mass on the pad or runway N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load) R_t – Taper ratio (c_{tip}/c_{root}) S_{csw} – Planform of wing mounted control surfaces S_{ref} – Theoretical wing planform area $(t/c)_{root}$ – Thickness to chord ratio at the wing root Λ – Wing sweep at 25% MAC	-54%

<table>
<tr><td>Reference: 6
Derived from: AVID equations from LaRC adjusted to Space Shuttle, includes aircraft for curve fit.
Options: Two equations with different parameters.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>

$$M_{wing-\exp} = 1575\left[\frac{M_{land}\,3.75bS_{\exp}}{c_{root}\left(t/c\right)\times 10^{9}}\right]^{0.67}$$

Primary wing equation

$$M_{wing-carrythru} = \left[\frac{1.06c_{root}(b_{cthru})}{S_{ref}}\right]1575\left[\frac{M_{land}\,3.75bS_{\exp}}{c_{root}\left(t/c\right)\times 10^{9}}\right]^{0.67}$$

$$M_{wing-fairing} = S_{fairing}\left[.0002499q_{\max} + 1.7008 + \left(.00003695q_{\max} - .003252\right)b_{body}\right]$$

b – Wing span
b_{body} – maximum width of the body
c_{root} – Wing chord at exposed root
M_{land} – Landed mass of vehicle
q_{max} – maximum dynamic pressure (psf)
S_{exp} – Exposed wing planform area
$S_{fairing}$ – Surface area of wing fairing
S_{ref} – theoretical wing planform area
(t/c) – Thickness to chord ratio on the wing</td><td>1%</td></tr>
<tr><td>

$$M_{wing-\exp} = 1.498S_{ref}^{1.176}$$

Includes carry through, and is considered a secondary equation.

S_{ref} – theoretical wing planform area</td><td>2%</td></tr>
</table>

<table>
<tr><td>Reference: 7
Derived from: Aircraft
Options: fixed or variable sweep wings.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>

$$M_{wing} = 3.08\left[\left\{\frac{\left(K_w N_z M_{glow}\right)}{\left(t/c\right)_{max}}\right\}\left\{\left(\tan(\theta_{le}) - 2\frac{1-R_t}{AR\left(1+R_t\right)}\right)^2 + 1.0\right\}10^{-6}\right]^{0.593}\left\{AR\left(1+R_t\right)\right\}^{0.89} S_{ref}^{0.741}$$

K_w = 1.0 – for fixed wing airplanes
= 1.175 – for variable sweep wing airplanes

AR – aspect ratio (b^2/S_{ref})
M_{glow} – Gross liftoff mass of vehicle
N_z – ultimate load factor = 1.5*2.5 (factor of safety * limit load)
R_t – taper ratio (c_{tip}/c_{root})
S_{ref} – theoretical wing planform area
$(t/c)_{max}$ – Maximum thickness to chord ratio on the wing
θ_{le} – sweep angle of leading edge</td><td>-35%</td></tr>
</table>

<table>
<tr><td>

Reference: 10b
Derived from: Hypervelocity Aircraft
Options: Continuous our discontinuous carry thru, landing gear location, strakes, trailing edge extension, and more.

</td><td>Space Shuttle Comparison</td></tr>
<tr><td>

$$M_{wing\_box} = K_{wing} K_{lcf} K_{trc} K_{arc} K_{tac} K_{swc} K_{bwc} K_{gear} K_{dwr} K_{tm} K_{ps} K_{dc} + K_{dw}$$

Elastic Axis Sweep =30deg ?
bexp=baero ?

K_{wing} $= 0.7072 S_{ref}^{1.334}$ - for continuous wing/carry-thru structures
$= 0.7072 S_{\exp}^{1.334}$ - for discontinuous wing/carry-thru structures (mid mount wings)
S_{exp} – Exposed wing planform area
S_{ref} – theoretical wing planform area

K_{lfc} – loading correction factor $= 0.00286 N_z^{0.581} P_{\exp} + 0.1624 N_z^{0.5585}$
N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load)
P_{exp} – Exposed wing planform loading (lb/ft^2)

K_{trc} – taper ratio correction factor $= 0.0141 \left(\frac{t}{c}\right)_{struct}^{-1.385} + 0.758$
$(t/c)_{struct}$ – Thickness to chord ratio of the wing structure

K_{arc} – aspect ratio correction factor $= 0.0588 AR_{\exp}^{1.148} + 0.28$
AR_{exp} – Aspect ratio of the exposed wing (b^2_{exp}/S_{exp})

K_{tac} – taper ratio correction factor $= 0.47 R_t + 0.833$

</td><td>-34%</td></tr>
</table>

Rt – Wing taper ratio = tip chord over centerline root chord

K_{swc} – sweepback correction factor $= 0.9031\cos(\theta_{elastic\_axis})^{-1.282}$

$\theta_{elastic\_axis}$ – **unknown number??**

K_{bwc} – body width correction factor $= 1.011 - 0.07\frac{b_{body}}{b_{\exp}} - 0.5\left(\frac{b_{body}}{b_{\exp}}\right)^2$

b_{body} – maximum width of vehicle body

b_{exp} – Exposed wing span = wing span less b_{body}

K_{gear} – landing gear support penalty = 1.1 – for wing mounted gear, 1.0 – otherwise

K_{dwr} – dead weight relief factor

= 1.0 – for wing without fuel or vertical tail

$= -1.2\left(\frac{M_{f\_wing}D_{f\_wing} + M_{vert\_wing}D_{vert}}{0.5M_{wing}D_{cp}}\right)$ – for wings with fuel and vertical tails attached.

D_{vert} – distance from vehicle centerline to CG of wing mounted vertical tail

D_{f_wing} – distance from vehicle centerline to CG of wing stored fuel

D_{cp} – distance from vehicle centerline to wing center of pressure

M_{f_wing} – mass of fuel in the wing

M_{vert_wing} – mass of vertical control surfaces attached to the wing

M_{wing} – mass of the wing

K_{tm} – temperature and materials factor

= 1.0 – for aluminum

= 1.15 – for titanium

= 2.8 – for nickel based superalloy

= 0.88 – for cold composite
= 0.92 – for titanium composite

K_{ps} – panel stiffness factor = 1.92 – for ceramic TPS, 1.0 – otherwise

K_{dc} – design concept factor = 0.97 – for thick truss structure design, 1.0 – otherwise

K_{dw} – discontinuous wing structural penalty = 0.0 – for continuous wing/carry-thru structures

$= \frac{1}{3}\left(M_{wing\_box_{continuous}} - M_{wing\_box_{discontinuous}}\right)$ – for discontinuous wing/carry-thru structures

$$M_{wing\_misc} = 0.1716 S_{\exp}^{1.275}\, 0.564 \left(\frac{b_{body}}{b_{\exp}}\right)^{-0.2098}$$

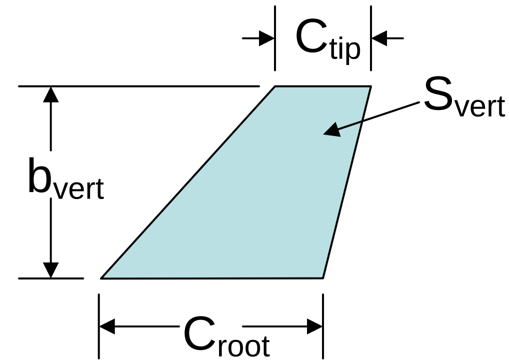
b_{body} – maximum width of vehicle body
b_{exp} – Exposed wing span = wing span less b_{body}

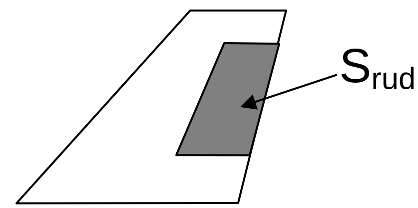
$$M_{wing\_extensions} = 6\left(S_{strakes} + S_{TEextensions}\right)$$

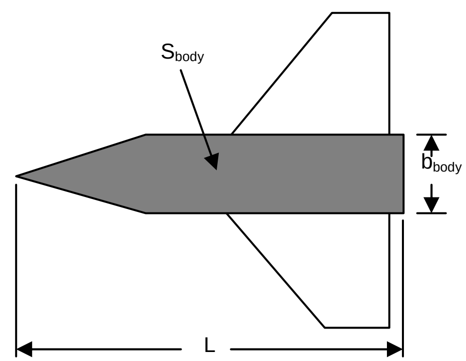
$S_{strakes}$ – planform area of wing strakes
$S_{TEextensions}$ – planform area of trailing edge extensions

2.0 Tail

AR_{vert} – Aspect ratio of vertical tail or tip fins
b_{body} – Maximum width of the body
b_{vert} – Span of tail or tip fins
c_{tip} – Tip chord of vertical tail or wingtip fins
M – Maximum flight Mach number
M_{glow} – Gross liftoff mass of vehicle
N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load)
R_{vert} – Taper ratio of vertical tail or tip fins (c_{tip}/c_{root})
S_{rud} – Planform area of rudder
S_{vert} – Total planform area of vertical tail or wingtip fins
$(t/c)_{vert}$ – Thickness to chord ratio of the vertical tail or wingtip fins
TRF – Technology reduction factor
Λ_{vert} – Sweep angle at 25% MAC







Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Materials.	Space Shuttle Comparison
$M_{tail} = K_t \left(S_{vert}\right)^{1.24}$ K_t =1.872 – aluminum skin/stringer, no TPS =1.108 – metallic composite structure, no TPS =1.000 – graphite epoxy composite structure, no TPS S_{vert} – Total planform area of vertical tail or wingtip fins	23%

Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: Wing material technology.	Space Shuttle Comparison
$M_{tail} = 5.0 S_{vert}^{1.09} (1 - TRF)$ *TRF* =1.0 – aluminum skin/stringer structure =0.2 – Ti3Al Beta 21s S_{vert} – Total planform area of vertical tail or wingtip fins	36%

Reference: 3 Derived from: Dr. Talay, LaRC. Options: None.	Space Shuttle Comparison
$M_{tail} = 1.678 S_{vert}^{1.24}$ S_{vert} – Total planform area of vertical tail or wingtip fins	13%

Reference: 4a Derived from: Airbreathing booster. Options: Maximum airbreathing Mach number.	Space Shuttle Comparison
$M_{tail} = K_{vert} S_{vert}$ K_{vert} = 7.68 – for max airbreathing Mach number of 8 = 9.20 – for max airbreathing Mach number of 12 S_{vert} – Total planform area of vertical tail or wingtip fins	22% using K_{vert}=7.68

Reference: 4b Derived from: Lifting body rocket. Options: None.	Space Shuttle Comparison
$M_{tail} = 6.8(0.2S_{body}) + 0.15S_{body}$ Vertical tail mass for a lifting body upper stage. S_{body} – Planform area of the body	89%

Reference: 5 Derived from: Aircraft. Options: Varying tail shapes.	Space Shuttle Comparison
$M_{tail} = 0.452(M_{glow} N_z)^{0.488} S_{vert}^{0.718} M^{0.341} b_{vert}^{-1} \left(1 + {S_{rud}}/{S_{vert}}\right)^{0.348} AR_{vert} (1 + R_{vert})^{0.25} (\cos(\Lambda_{vert}))^{-0.323}$ Assumes no T-tail and no rolling tail. AR_{vert} – Aspect ratio of vertical tail or tip fins b_{vert} – Span of tail or tip fins M – Maximum flight Mach number M_{glow} – Gross liftoff mass of vehicle N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load) R_{vert} – Taper ratio of vertical tail or tip fins S_{rud} – Planform area of rudder S_{vert} – Total planform area of vertical tail or wingtip fins Λ_{vert} – Sweep angle at 25% MAC	28% N_z = 3.75 0% N_z = 2.25

<table>
<tr><td>Reference: 6
Derived from: Boeing aircraft tail equations adjusted for Space Shuttle.
Options: Component inclusion.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>The following three equations must be summed to find the total tail mass.

$$M_{tail} = 26.06\left(S_{vert}\left({}^{t}\!/\!{}_{c}\right)_{vert}^{0.244} b_{vert}^{0.0364}\right)^{0.8674}$$

$$M_{vert\_spar} = \frac{c_{tip} b_{vert}}{2S_{vert}} M_{tail}$$

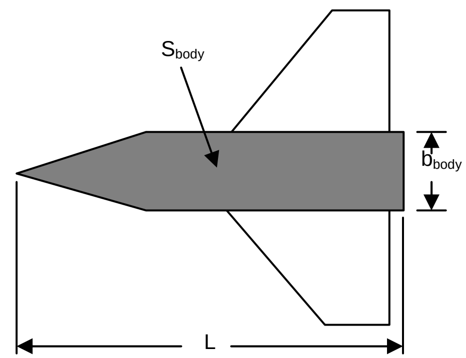
$$M_{fairing} = S_{fairing}\left(0.02499 q_{\max} + 1.7008 + \left(0.003695 q_{\max} - 0.3252\right) b_{body}\right)$$

b_{body} – Maximum width of the body
b_{vert} – Span of tail or tip fins
c_{tip} – Tip chord of vertical tail or wingtip fins
q_{max} – Maximum dynamic pressure during flight
$S_{fairing}$ – Surface area of tail fairing
S_{vert} – Total planform area of vertical tail or wingtip fins
$(t/c)_{vert}$ – Thickness to chord ratio of the vertical tail or wingtip fins</td><td>-8%</td></tr>
</table>

<table>
<tr><td>Reference: 10
Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing).
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{tail} = 28.1\left[(S_{vert})^{0.901} R_{vert}^{0.244} b_{vert}^{0.0364}\right]^{0.8674}$

b_{vert} – Span of tail or tip fins
R_{vert} – Taper ratio of vertical tail or tip fins
S_{vert} – Total planform area of vertical tail or wingtip fins</td><td>9%</td></tr>
</table>

3.0 Body



A_{as} – surface area of aft structure
A_{body} – surface area of vehicle body
$\mathrm{A}_{\mathrm{body\text{-}tank}}$ – exposed area of body minus exposed area of integral tanks
A_{exit} – Total exit area of main engines
A_{inlet} – cross sectional area of inlet
A_{tank} – Surface area of tank
b_{body} – Maximum width of the body
D_{eng} – Diameter of a main engine
D_{nose} – Diameter of the nosecone base
F_{ullage} – Ullage fraction (typically ~4 to 5%)
F_{prop} – Propellant fraction of either oxidizer or fuel
H_{body} – height of body
H_{inlet} – Height of engine inlet
Isp – Specific impulse of engines
L – Length of vehicle
L_{inlet} – Length of engine inlet
L_s – Length of single duct (for Y inlet ducts)
$\dot{m}$ – Total propellant mass flow rate (lbm/s)
M_{body} – Total mass of body group
M_{eng} – Mass of a single main engine
M_{glow} – Gross liftoff mass
M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff
M_{insert} – Insertion mass, sometimes called burnout mass
M_{land} – Landed mass of vehicle
M_{pl} – Mass of payload
$M_{strapon}$ – Mass of strap on boosters
M_{tot_fuel} – Mass of all fuel on stage
M_{tot_ox} – Mass of all oxidizer on stage

N_{crew} – Number of crew
N_{days} – Number of days spent on orbit
N_{eng} – Number of main engines on stage
N_{inlet} – Number of inlets
N_{struts} – Number of struts in engine inlet
N_t – Number of fuel tanks
N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load)
P_2 – Pressure in inlet
P_f – Pressure of fuel tank
P_{ox} – Pressure of oxidizer tank
q_{max} – Maximum dynamic pressure during flight (lb/ft^2)
S_{as} – Surface area of aft skirt
S_{base} – Surface area of base closeout
S_{body} – Planform area of vehicle body
S_{bf} – Planform area of body flap
S_{ec} – Surface area of engine compartment
S_f – Surface area of fuel tanks
S_{fwds} – Surface area of forward skirt
S_{inlet} – Surface area of inlet and cowl ring
S_{is} – Surface area of interstage structure
S_{it} – Surface area of intertank structure
S_{nose} – Surface area of nosecone
S_{ns_cowl} – Non-inlet surface area of cowl
S_{ox} – Surface area of oxidizer tanks
S_{pl} – Surface area of payload bay, not including doors
$S_{pldoors}$ – Surface area of payload bay doors
S_{tc} – Surface area of tail cone
SFC – Specific fuel consumption
T_{sls} – Total stage thrust at sea level static conditions
T_{vac} – Vacuum thrust per main engine
V_{crew} – Volume of crew cabin

V_f – Total fuel volume
V_i – Fuel volume in integral tanks
V_{ox} – Total oxidizer volume
ρ_f – Density of fuel
ρ_{ox} – Density of oxidizer
θ_{nose} – Nose cone angle

<table>
<tr><td>Reference: 1
Derived from: Aircraft and Space Shuttle.
Options: Materials, windshield, and tanks..</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$$M_{body} = K_c N_{crew}^{0.5} + K_b A_{body} N_z^{1/3} + K_f V_f + K_{ox} V_{ox}^{1.1} + K_t \left(N_{eng} T_{vac} \right) + K_{bf} S_{bf}^{1.15}$$
Crew cabin constants
K_c = 2043 – full windshield aluminum construction
= 1293 – aluminum construction with no windshield
= 1740 – full windshield composite construction
= 1140 – composite construction with no windshield

Body construction constants
K_b = 2.72 – composite structure, no TPS
= 3.20 – aluminum structure, no composites, no TPS
= 3.40 – hot metallic Ti/Rene HC, no TPS required
= 4.43 – moldline tankage; tank, body structure, cryogenic insulation integrated

Tank geometry/propellant constants
K_f, and K_{ox} – see table below.</td><td>-7%
Assumes no ascent propellant in orbiter.</td></tr>
</table>

SOURCE	PROPELLANT	K LB/FT^3	TANK DESCRIPTION					
			VOL FT^3	ULLAGE PRESSURE	INTEGRAL OR NON-INTEGRAL	MATERIAL	GEOMETRY	COMMENTS
SHUTTLE E/T	LH_2	.5918	53,515	36	INTEGRAL	AL2219		DOES NOT INCLUDE INSULATION
EN-155*	LH_2	.8430	60,387	30	INTEGRAL	INC 718		HONEYCOMB SANDWICH ADDED HONEYCOMB FOR THERMAL PROTECTION
EN-178*	LH_2	.5760	41,646	20	INTEGRAL	AL2219		ISOGRID INCLUDES 4,364 LB INSULATION
SHUTTLE E/T	LOX	.6458	19,609	38	INTEGRAL	AL2219		DOES NOT INCLUDE INSULATION
EN-155*	LOX	.7660	18,355	20	NON INTEGRAL	AL2219		POLYMIDE HONEYCOMB FOR INSULATION AND STRUCTURAL STABILIZATION
EN-178	LOX	.5160	21,841	15	INTEGRAL	AL2219		ISOGRID INCLUDES 1,704 LB INSULATION
S-1C	LOX	.804	47,250		INTEGRAL	AL2219		
EN-155	JP-5	.7000	4,819	5	NON INTEGRAL	AL2219		CONVENTIONAL SKIN STIFFENED CONSTRUCTION W/O INSULATION
	JP-5	.28					N/A	PENALTY FOR DRY-WET WING
S-1C	RP-1	.867	30,000		INTEGRAL	AL2219		

Table showing tank constants from [ref. 1].
* EN designates in-house LARC study vehicles.

K_t = 0.0030 – aluminum thrust structure
= 0.0024 – composite thrust structure

Body flap construction constants
K_{bf} = 1.59 – hot structure
= 1.38 – aluminum skin/stringer, no TPS

A_{body} – surface area of vehicle body
M_{body} – Total mass of body group
N_{crew} – Number of crew
N_{eng} – Number of main engines on stage
N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load)
S_{bf} – Planform area of body flap
T_{vac} – Vacuum thrust per main engine
V_f – Total fuel volume
V_{ox} – Total oxidizer volume

<table>
<tr><td>Reference: 2
Derived for: Airbreathing horizontal takeoff vehicle.
Options: Material technology.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{nosecone} = K_{nc} S_{nose}$

$M_{crew\_cabin} = 1455 N_{crew}^{0.5}$

$M_{pl\_bay} = K_{pl} S_{pl} + K_{pldoors} S_{pldoors} + 0.15 M_{pl}$

$M_{fuel\_tank} = K_f V_f + K_{f\_ins} S_f$ includes insulation

$M_{ox\_tank} = K_{ox} V_{ox} + K_{ox\_ins} S_{ox}$ includes insulation

$M_{aft\_body} = K_{tc} S_{tc} + K_{base} S_{base}$

$M_{cowl} = K_{ns\_cowl} S_{ns\_cowl} + 2 K_{inlet} S_{inlet} + K_{struts} L_{inlet} H_{inlet} N_{struts}$ airbreather only

K_{nc} = 2.21 – Ti3Al Beta 21S
K_{pl} = 2.21 – Ti3Al Beta 21S
$K_{pldoors}$ = 3.5 – 20% less than STS honeycomb doors (incl. fittings & mechanisms)
K_f = 0.255 – Hydrogen, wound integral Gr/PEEK
K_{f_ins} = 0.26 – Based on rohacell insulation
K_{ox} = 0.33 – LOX, aluminum lithium, non-integral
K_{ox_ins} = 0.20 – Based on rohacell insulation
K_{tc} = 2.21 – Ti3Al Beta 21S
K_{base} = 1.99 – Secondary structure (10% lower than baseline?)</td><td>-42%</td></tr>
</table>

K_{ns_cowl} =2.21 – Ti3Al Beta21S
K_{inlet} = 2.75 – Advanced materials, 150psi, top & bottom required
K_{struts} = 2.21 – Baseline structural unit weight

S_{nose} – Surface area of nosecone
S_{pl} – Surface area of payload bay, not including doors
$S_{pldoors}$ – Surface area of payload bay doors
M_{pl} – Mass of payload
S_f – Surface area of fuel tanks
S_{ox} – Surface area of oxidizer tanks
S_{tc} – Surface area of tail cone
S_{base} – Surface area of base closeout
S_{ns_cowl} – Non-inlet surface area of cowl
S_{inlet} – Surface area of inlet and cowl ring
L_{inlet} – Length of engine inlet
H_{inlet} – Height of engine inlet
N_{crew} – Number of crew
N_{struts} – Number of struts in engine inlet
V_f – Total fuel volume
V_{ox} – Total oxidizer volume

<table>
<tr><td>Reference: 3
Derived from: Dr. Talay, LaRC, rocket based.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>

$M_{fuse} = 3.4 A_{body-tanks}$ Includes fore, aft, mid fuselage, and payload bay doors

$M_{secondary} = 2.0\left(S_{base} + S_{pl}\right)$ Add any other secondary structures' areas specific to vehicle

$M_{crew\_cabin} = 2347 N_{crew}^{0.5}$

$M_{bf} = 3.135 S_{bf}$

$M_{thrust\_struct} = 0.0023 T_{vac} N_{eng}$

$M_{fuel\_tank} = \frac{K_f V_f}{(1 - F_{ullage})}$

$M_{ox\_tank} = \frac{K_{ox} V_{ox}}{(1 - F_{ullage})}$

K_f = 0.5595 – Shuttle technology
K_{ox} = 0.8086 – Shuttle technology

$A_{body-tank}$ – Exposed area of body minus exposed area of integral tanks
F_{ullage} – Ullage fraction (typically ~4 to 5%)
N_{crew} – Number of crew
N_{eng} – Number of main engines on stage

</td><td>-3%</td></tr>
</table>

S_{base} – Surface area of base closeout S_{bf} – Planform area of body flap S_{pl} – Surface area of payload bay, not including doors T_{vac} – Vacuum thrust per main engine V_f – Total fuel volume V_{ox} – Total oxidizer volume	

Reference: 4a Derived from: Airbreathing booster. Options: Maximum airbreathing Mach number, engine type, and body type.	Space Shuttle Comparison
$M_{aft\_struct} = K_{as} A_{as}$ Inconel 718 aft structure mass K_{as} = 2.86 – Max airbreathing Mach number of 8 = 3.10 – Max airbreathing Mach number of 12 A_{as} – surface area of aft structure $M_{thrust} = K_{thrust} T_{sls}$ Thrust structure mass for aibreathing booster vehicle K_{thrust} = 0.01025 – Thrust acting below body (ie. Ramjet) = 0.0070 – Thrust acting on aft expansion surface (ie. Scramjet) T_{sls} – Total stage thrust at sea level static conditions $M_{fuel\_tanks} = K_{fuel} M_{tot\_fuel}$ Mass of liquid hydrogen tanks for an airbreathing booster vehicle. This tank is integral with the forebody of the vehicle and includes structure. K_{fuel} = 0.259 – Augmented rocket = 0.409 – Ejector ramjet, or supercharged ejector ramjet = 0.416 – Ejector scramjet, or supercharged ejector scramjet = 0.341 – RL, or RRL, or SRL, or RSRL = 0.339 – SL, or RSL, or SSL, or RSSL M_{tot_fuel} – Mass of all fuel on stage	N/A Air-breathing vehicles only

$M_{ox\_tanks} = 0.0255 M_{tot\_ox}$ Mass of liquid oxygen tanks for an airbreathing booster vehicle M_{tot_oxl} – Mass of all oxidizer on stage $M_{cowl} = K_{cowl} A_{inlet}$ Mass of inlets K_{cowl} = 175 – Cylindrical body with wing configuration, 120 psia inlet pressure = 154 – Lifting body configuration, subsonic combustion, 120 psia inlet pressure = 125 – Lifting body configuration, supersonic combustion, 120 psia inlet pressure = See chart for different inlet pressures. A_{inlet} – cross sectional area of inlet $M_{crew\_cabin} = 1400 + 860$ Fixed mass for crew cabin structure, and personnel compartment. $M_{separation} = 0.0133 M_{insert}$ Separation system on booster stage, piggy-back configuration. Includes separation rockets, mounting system, and controls. M_{insert} – Orbital insertion mass of vehicle, sometimes called burnout mass.	

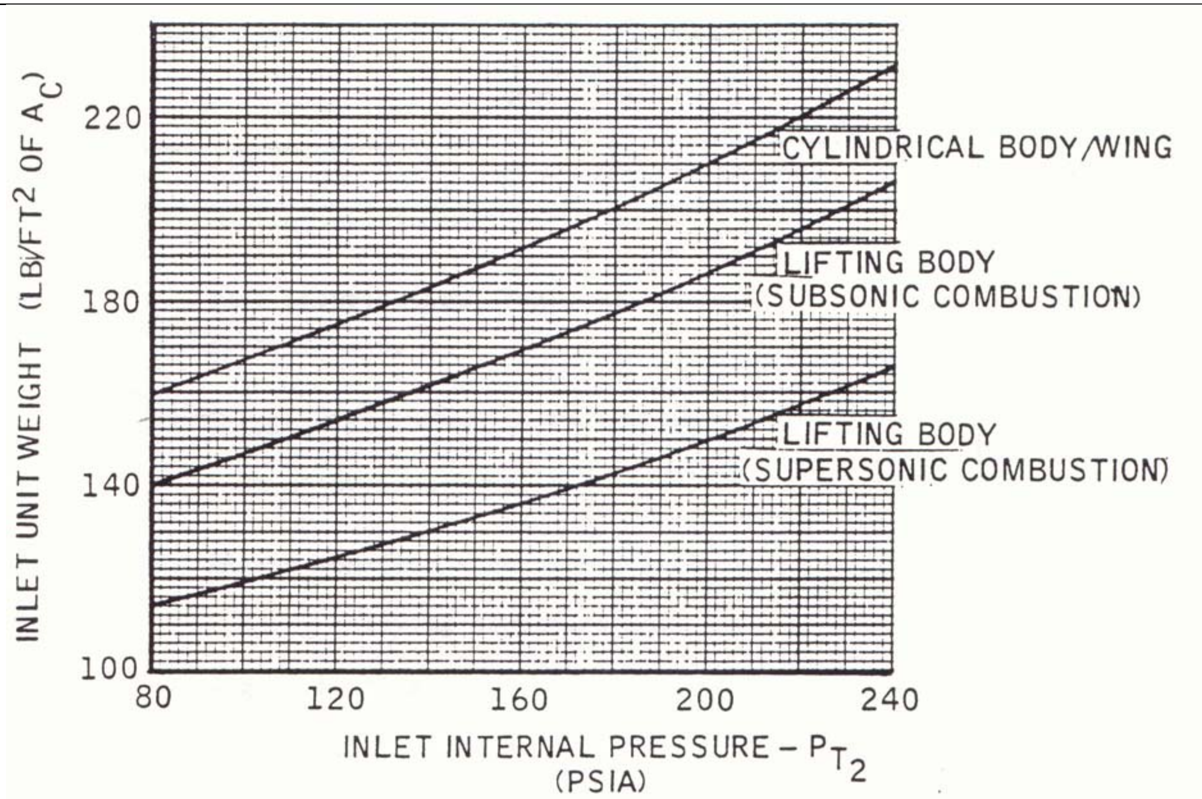


Chart showing inlet weight per square foot of inlet area as a function of inlet pressure for three different vehicle configurations. Source [ref. 4].

<table>
<tr><td>Reference: 4b
Derived from: Lifting body rocket.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{body} = 3.0724 A_{body} + 0.0008 T_{vac} N_{eng}$
Body mass including fuselage, thrust structure, and miscellaneous, for a lifting body upper stage.

$M_{crew\_cabin} = 1801 + 1.15 V_{crew} + 180$
Mass of crew cabin and windscreen/canopy. This reference recommends that the volume for the crew be calculated as: $V_{crew} = 60N_{crew}+255$

$M_{aft\_skirt} = 0.224 S_{body}$
Mass of aft skirt, aerodynamic fairing over engines.
S_{body} – Planform area of vehicle body

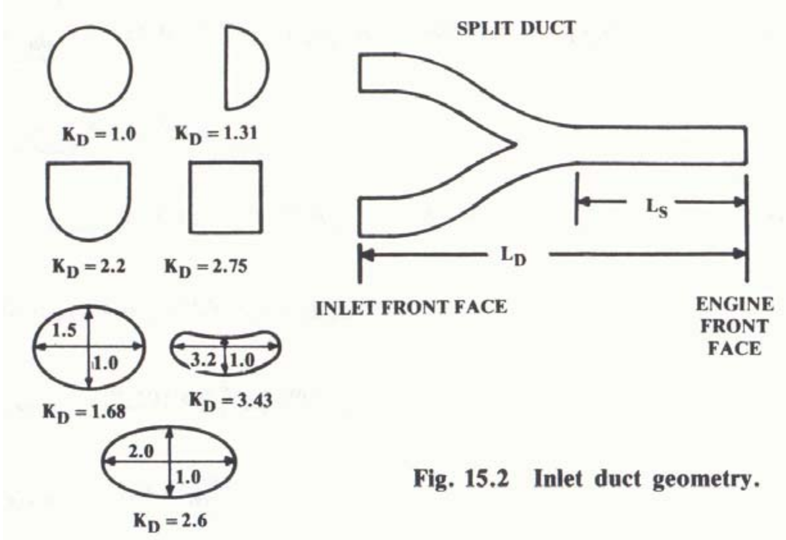
$M_{ox\_tank} = 0.0181 M_{tot\_ox}$
Mass of liquid oxygen tank for lifting body second stage
M_{tot_ox} – Mass of all oxidizer on stage

$M_{f\_tank} = 0.1188 M_{tot\_fuel}$
Mass of liquid hydrogen tank for lifting body second stage, including mounting.
M_{tot_fuel} – Mass of all fuel on stage

$M_{f\_ins} = 1.555 V_f^{0.666}$
Mass of insulation for liquid hydrogen tank on lifting body second stage.
V_f – Total fuel volume</td><td>N/A
Lifting body design</td></tr>
</table>

$M_{plbay} = 1.4 V_{plbay}$ Mass of payload bay, including doors. Recommended cargo volume is: $V_{plbay} = 0.111 M_{pl}$ V_{plbay} – Volume of payload bay $M_{sep\_syst} = 220$ Mass of separation system on second stage lifting body, piggy-back config.	

Reference: 5 Derived from: Aircraft. Options: Different inlet geometry including variable shape, and wing shape.	Space Shuttle Comparison
$M_{fuse} = 0.499 K_{dwf} M_{gross}^{0.35} N_z^{0.25} L^{0.5} H_{body}^{0.849} b_{body}^{0.685}$ $M_{thrust\_struct} = 0.013 N_{eng}^{0.795} T_{vac}^{0.579} N_z + 0.01 M_{eng}^{0.717} N_{eng} N_z$ $M_{cowl} = 13.29 K_{vg} L_{inlet}^{0.643} K_{duct}^{0.182} N_{eng}^{1.498} \left(\frac{L_s}{L_{inlet}}\right)^{-.0373} D_{eng}$ based on fighter aircraft inlet ducts $M_{fuel\_tanks} = 7.45 V_f^{0.47} \left(1 + \frac{V_i}{V_f}\right)^{-0.095} N_t^{0.066} N_{eng}^{0.052} \left(\frac{T_{vac} SFC}{1000}\right)^{0.249}$ JP fuel tanks only K_{dwf} = 0.774 – for delta wing = 1.0 – otherwise K_{vg} = 1.62 – for variable geometry inlet = 1.0 – fixed geometry inlet K_{duct} = 1.0 – circular inlet = 1.31 – half circle inlet = 2.2 – square and circle combination inlet (stretched D) = 2.75 – square inlet = 1.68 – ellipsoid, height to width ratio of 1.5:1 = 2.6 – ellipsoid, height to width ratio of 2:1 = 3.43 – smile shape (ie. F-16), height to width ratio of 1:3.2	-26%



Inlet duct geometry coefficients [ref. 5].

b_{body} – Maximum width of the body
D_{eng} – Diameter of a main engine
H_{body} – height of body
L – Vehicle length
L_s – Length of single duct (for Y inlet ducts)
M_{eng} – Mass of a single main engine
M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff
N_{eng} – Number of main engines on stage
N_t – Number of fuel tanks
N_z – Ultimate load factor = 1.5*2.5 (factor of safety * limit load)
SFC – Specific fuel consumption
T_{vac} – Vacuum thrust per main engine
V_f – Total fuel volume
V_i – Fuel volume in integral tanks

Reference: 6 Derived from: Fuselage from aircraft & Space Shuttle, others from Space Shuttle and expendable vehicles. Options: Stage number, attachment configuration.	Space Shuttle Comparison
Shuttle comparison includes fuselage, nosecap, thrust structure, payload bay and doors, crew cabin, and stage to stage attachment structure. $M_{fuse} = 2.167 A_{body}^{1.075}$ Mass of vehicle body, including base $M_{nose} = S_{nose}\left[\begin{array}{l}(14.31 - 0.003462 q_{\max})\theta_{nose}^{(0.0001034 q_{\max} - 0.5878)} + \\ \{(0.0006864 - 6.1e^{-9} q_{\max})\theta_{nose} + (4.385e^{-5} q_{\max} - 3.252e^{-3})D_{nose}\}\end{array}\right]$ right cone $M_{nose} = S_{nose}\left[2.499e^{-4} q_{\max} + 1.7008 + (3.695e^{-5} q_{\max} - 3.252e^{-3})D_{nose}\right]$ ellipsoid $M_{ox\_\tan k} = (2.44 - 0.007702\rho_{ox})V_{ox}^{(0.8548 + 0.0003189\rho_{ox})}$ P<55psi $M_{ox\_\tan k} = (1.3012 + 0.0099 P_{ox})V_{ox}^{(0.8647 P_{ox}^{0.01645})}$ 150<P<1200psi, steel tank $M_{f\_\tan k} = (2.44 - 0.007702\rho_{f})V_{f}^{(0.8548 + 0.0003189\rho_{f})}$ P<55psi $M_{f\_\tan k} = (1.3012 + 0.0099 P_{f})V_{f}^{(0.8647 P_{f}^{0.01645})}$ 150<P<1200psi, steel tank $M_{antivortex} = \frac{\dot{m} F_{prop}}{\rho}(0.64 + 0.0184\rho)$ adapt for propellant type	2%

$$M_{slosh\_baffles} = 6.77e^{-7} b_{body} V \frac{\rho^2}{1.01}$$ adapt for propellant type

For $M_{antivortex}$ and $M_{slosh\_baffles}$ the volume, density and F_{prop} need to have the correct subscript for the fluid in the tank. For example for a LOX tank F_{prop} would be the oxidizer fraction, V would be the oxidizer tank volume, and ρ would be the density of LOX.

$$M_{intertank} = S_{it} K_{it} b_{body}^{k_{it2}}$$ Structure between tanks for inline configuration

K_{it} = 26.36 – stage 1 of 1
= 27.04 – stage 1 of 2
= 21.47 – stage 2 of 2

K_{it2} = 0.5169 – stage 1 of 1 or stage 1 of 2
= 0.6025 – stage 2 of 2

$$M_{interstage} = S_{is} K_{is} b_{body}^{K_{is2}}$$ Structure connecting two stages of an inline vehicle

K_{is} = 17.92 – stage 1 of 1
= 18.57 – stage 1 of 2
= 22.94 – stage 2 of 2

K_{is2} = 0.4856 – stage 1 of 1 or stage 1 of 2
= 0.6751 – stage 2 of 2

$$M_{fwd\_skirt} = S_{fwds} K_{fwds} b_{body}^{K_{fwds2}}$$ Mass of structure between forward tank and payload or next stage

K_{fwds} = 37.35 – stage 1 of 1
= 38.70 – stage 1 of 2
= 15.46 – stage 2 of 2

K_{fwds2} = 0.6722 – stage 1 of 1 or stage 1 of 2
= 0.5210 – stage 2 of 2

$$M_{thrust} = K_{thrust} T_{vac}^{1.0687}$$ Thrust structure mass

K_{thrust} = 1.949e-3 – inline launch vehicle
= 7.995e-4 – side mount propulsion module (orbiter type)

$M_{eng\_comp} = S_{ec} K_{ec} b_{body}^{K_{ec2}}$ structure from aft tank to interstage or pad tie-down

K_{ec} = 31.66 – stage 1 of 1
= 32.48 – stage 1 of 2
= 15.97 – stage 2 of 2

K_{ec2} = 0.5498 – stage 1 of 1 or stage 1 of 2
= 0.4676 – stage 2 of 2

$M_{aft\_skirt} = S_{as}\left[2.499e^{-4} q_{\max} + 1.7008 + \left(3.695e^{-5} q_{\max} - 3.252e^{-3}\right) b_{body}\right]$ aerodynamic fairing

$M_{stg\_attach} = 0.0148\left(M_{gross} + M_{pl}\right)$

Orbiter type vehicle to ET or booster stage where attach structure stays with ET or booster stage.

$M_{stg\_attach} = 0.00148 M_{strapon}$

SRB to ET or core stage where attach structure stays with ET or booster stage

$M_{stg\_attach} = 0.000314 M_{strapon}$ SRB attach structure stays with SRB

$$M_{crew\_cabin} = 28.31\left[39.66\left(N_{crew} N_{days}\right)^{1.002}\right]^{0.6916}$$

$M_{pldoors} = 0.257 \dfrac{A_{body}}{2}$ Payload bay doors including hardware

$M_{plbay} = 0.4808 A_{body} + 0.2336 \dfrac{A_{body}}{2}$ Internal cargo bay mass, including support structure (ie.STS)

A_{body} – Surface area of the vehicle body
b_{body} – Maximum width of the body
D_{nose} – Diameter of the nosecone base
F_{prop} – Propellant fraction of either oxidizer or fuel
$\dot{m}$ – Total propellant mass flow rate (lbm/s)
M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff
M_{pl} – Mass of payload
$M_{strapon}$ – Mass of strap on boosters
N_{crew} – Number of crew
N_{days} – Number of days spent on orbit
P_f – Pressure of fuel tank
P_{ox} – Pressure of oxidizer tank
q_{max} – Maximum dynamic pressure during flight (lb/ft^2)
S_{as} – Surface area of aft skirt
S_{ec} – Surface area of engine compartment
S_{fwds} – Surface area of forward skirt
S_{is} – Surface area of interstage structure
S_{it} – Surface area of intertank structure
S_{nose} – Surface area of the nosecone
T_{vac} – Vacuum thrust per main engine
V_f – Total fuel volume
V_{ox} – Total oxidizer volume
ρ_f – Density of fuel
ρ_{ox} – Density of oxidizer
θ_{nose} – Nose cone angle

<table>
<tr><td>Reference: 7
Derived from: Aircraft.
Options: Inlet geometry and pressure.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$$M_{fuse} = 20.86 K_{inl}^{1.42} \left(\frac{q_{\max}}{100}\right)^{0.283} \left(\frac{M_{glow}}{1000}\right)^{0.95} \left(\frac{L}{H_{body}}\right)^{0.71}$$
Fuselage mass based on fighter planes using the General Dynamics method.

$$M_{cowl} = K_{inlet} N_{inlet} \left(S_{inlet}^{0.5} L_{inlet} P_2\right)^{0.731}$$
Cowl mass based on aircraft inlets
K_{inlet} = 3.0 – turbojet
= 7.435 – turbofan
q_{max} – Maximum dynamic pressure during flight (lb/ft^2)
M_{glow} – Gross liftoff mass
L – Length of vehicle
H_{body} – height of body
N_{inlet} – Number of inlets
L_{inlet} – Length of engine inlet
P_2 – Pressure in inlet
S_{inlet} – Surface area of inlet and cowl ring</td><td>3%</td></tr>
</table>

Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: Integral or non-integral tanks.	Space Shuttle Comparison
Shuttle comparison includes the rocket fuselage, body flap, payload bay and doors, crew cabin, stage to stage attachment, and separation system. $M_{fuse} = 2.8279\left(0.682 + 0.272\rho_{veh}/9.55 + 0.046\left(\rho_{veh}/9.55\right)^2\right)A_{body}$ Airbreather smeared fuse $M_{fuse} = \left(2.0833A_{body}^{1.075}\right) + \left[0.000011689\left(T_{sls}N_{eng}b_{body}\right)^{0.9846} + 5.02\left(S_{base} - A_{exit}\right)\right]$ Rocket smeared fuse Integral tanks are included in the body area for these equations. $M_{non-integral-tanks} = 1.68\left\{\left[\left(2.44 - 0.007702\rho\right)V^{(0.8548+0.0003189\rho)}\right] + \left[\frac{T_{sls}}{Isp}\frac{(1-R)}{\rho}\left(0.64 + 0.0184\rho\right)\right] + 0.000000677b_{body}\frac{V}{1.01}\rho^2\right\}$ (source 10a) Valid for all propellant types, includes slosh baffles, anti-vortex baffles, and are intended for use with pump fed engines in the horizontal mounting position. $M_{insulation\_nonintegral\_tank} = 0.2A_{tank}$ For non-integral tanks only $M_{bf} = 3.421S_{bf}$ Body flap mass $M_{plbay} = 0.5108S_{pl}$ Payload bay mass	-9%

$$M_{pldoors} = 0.5623 S_{pldoors}$$ Payload bay doors mass

$$M_{crew\_cabin} = 28.31 V_{crew}^{0.6916}$$ Crew cabin mass

$$M_{attach} = 0.00155 M_{land}$$ State to stage attachment structure, for either booster or orbiter

$$M_{sep} = 0.0404 M_{insert}^{0.7728}$$ Booster side of separation system

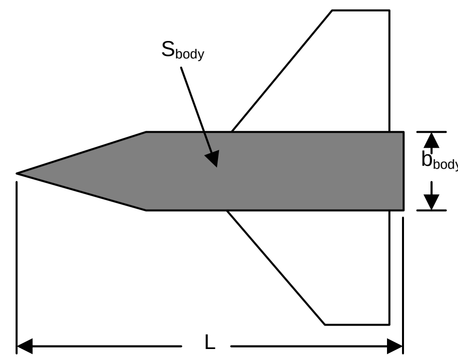
$$M_{sep} = 0.00989 M_{gross}^{0.9182}$$ Orbiter side of separation system

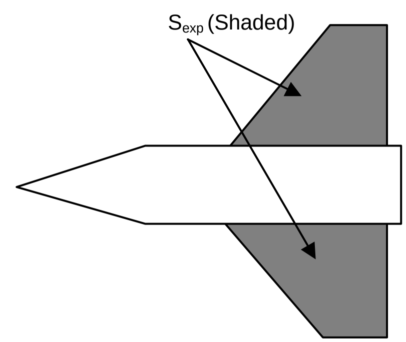
A_{body} – surface area of vehicle body
A_{exit} – Total exit area of main engines
A_{tank} – Surface area of tank
b_{body} – Maximum width of the body
Isp – Specific impulse of engines
M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff
M_{insert} – Insertion mass, sometimes called burnout mass
M_{land} – Landed mass of vehicle
N_{eng} – Number of main engines on stage
R – fraction of total ascent propellant that is the propellant used in this tank
S_{base} – Surface area of base closeout
S_{bf} – Planform area of body flap
S_{pl} – Surface area of payload bay, not including doors
$S_{pldoors}$ – Surface area of payload bay doors
T_{sls} – Total stage thrust at sea level static conditions
V – volume of propellant stored in tank
V_{crew} – Volume of crew cabin
ρ – density of propellant stored in tank

ρ_{veh} – Vehicle bulk density	

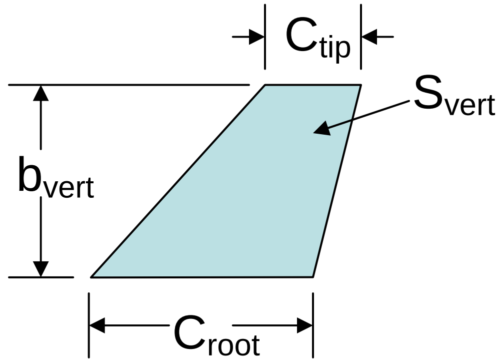
4.0 TPS

A_{acc} – Area of advanced carbon-carbon TPS
A_{body} – surface area of vehicle body
A_{body_tps} – Wetted area of TPS on vehicle body
A_{exit} – Exit area of main engines
A_{ins} – Wetted area of vehicle covered by insulation
A_{ref} – reference aerodynamic area (front projected shadow area)
$A_{sa\_standoff}$ – Area of superalloy standoff TPS
A_{sb} – exposed surface area of speed brakes
$A_{ti\_standoff}$ – Area of titanium standoff TPS
A_{tps} – Wetted area of vehicle covered by TPS
C_L – Average coefficient of lift from orbit to Mach 10
D_{nose} – Diameter of base of nosecone
H_{le} – Height of leading edge
L_{cowl_le} – Length of cowl leading edge
L_{le} – Length of leading edges (wing and nose if applicable)
L_{wing_le} – Length of wing leading edge
M_{entry} – Entry mass of vehicle
N_{crew} – Number of crew
N_{eng} – Number of main engines on stage
q_{max} – Maximum dynamic pressure during flight (lb/ft^2)
S_{bf} – Planform area of body flap
S_{body} – Planform area of vehicle body
S_{exp} – Planform area of exposed wing
S_f – Surface area of fuel tanks
S_{mono_tank} – Surface area of monopropellant tank
S_{ox} – Surface area of oxidizer tanks
S_{tps} – Planform area covered by TPS
S_{vert} – Total planform area of vertical tail or wingtip fins





T_{vac} – Vacuum thrust per main engine
V_{crew} – Volume of crew cabin
ψ_{le} – Leading edge angle **(? Sweep or angle of airfoil nose)**



Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Materials.	Space Shuttle Comparison
Shuttle comparison uses C_l = 0.65 and K_{flow} = 0.556. $M_{tps} = K_r \left(\frac{1}{K_t}\right)^{0.302} \left(\frac{M_{entry}}{(S_{body}+S_{\exp})C_L}\right)^{K_{flow}} (A_{body} + 2S_{\exp})$ Rocket vehicle, lifting re-entry. K_r = 0.140 – RSI (shuttle technology) – material/config. constant = 0.110 – RSI Advanced = 0.145 – metallic K_t = 0.100 – aluminum skin/stringer – equivalent thermal thickness of backup structure (in.) = 0.085 – titanium = 0.115 – graphite epoxy K_{flow} – Flow constant in the range below. = 0.5 – Pure laminar flow = 0.8 – Pure turbulent flow A_{body} – surface area of vehicle body C_L – Average coefficient of lift from orbit to Mach 10 M_{entry} – Entry mass of vehicle S_{body} – Planform area of vehicle body S_{exp} – Planform area of exposed wing	0%

Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: Material technology.	Space Shuttle Comparison
$M_{active\_cooling} = 150 + 2.70 L_{cowl\_le} + 2.70 L_{wing\_le} + 3.50 A_{exit}$ Based on 5deg. Cone for heat rates (Wilhite) Includes nosecap (150 lbs.), wing leading edges, cowl leading edges, and cooled engine exit are. Primarily intended for airbreather. $M_{acc} = 2.0 A_{acc}$ Advanced carbon/carbon, based on advanced NASP TPS, Shideler. For T>1800F Typically used on wing, body, and cowl windward sides. $M_{sa\_standoff} = 1.06 A_{sa\_standoff}$ Superalloy standoff, based on advanced metallic NASP, Shideler. For T>1200F $M_{ti\_standoff} = 0.508 A_{ti\_standoff}$ Titanium standoff, based on advanced metallic NASP, Shideler. For T<1200F A_{acc} – Area of advanced carbon-carbon TPS A_{exit} – Exit area of main engines $A_{sa\_standoff}$ – Area of superalloy standoff TPS $A_{ti\_standoff}$ – Area of titanium standoff TPS L_{cowl_le} – Length of cowl leading edge L_{wing_le} – Length of wing leading edge	N/A Different technology

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{tps} = 0.35\left(\frac{M_{entry}}{S_{tps}}\right)^{0.5} A_{tps}$ Shuttle technology. A_{tps} – Wetted area of vehicle covered by TPS M_{entry} – Entry mass of vehicle S_{tps} – Planform area covered by TPS	39%

Reference: 4a Derived from: Airbreathing booster. Options: Maximum airbreathing Mach number.	Space Shuttle Comparison
$M_{tps} = K_{tps} A_{tps}$ TPS mass for an airbreathing booster stage using reusable metallic (inconel and columbium shingles) TPS, including 2% contingency. K_{tps} = 1.42 – for maximum airbreathing Mach number of 8 = 1.54 – for maximum airbreathing Mach number of 12 $M_{ins} = K_{ins} A_{ins}$ Insulation mass for an airbreathing booster stage, including 2% contingency. A_{ins} – surface area requiring insulation. K_{ins} = 1.07 – for max airbreathing Mach number of 8 = 1.23 – for max airbreathing mach number of 12 A_{ins} – Wetted area of vehicle covered by insulation A_{tps} – Wetted area of vehicle covered by TPS	N/A Different technology

<table>
<tr><td>Reference: 4b
Derived from: Lifting body rocket.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{ins} = 1.51A_{body}$
Mass of external insulation on a lifting body second stage. No cover panels are used.

$M_{crew\_ins} = 5.2V_{crew}^{0.6666}$
Insulation protecting the crew cabin. This reference recommends that the volume for the crew be calculated as:
$V_{crew} = 60N_{crew}+255$

A_{body} – surface area of vehicle body
N_{crew} – Number of crew
V_{crew} – Volume of crew cabin</td><td>-57% compared to all TPS

249% compared to insulation only</td></tr>
</table>

Reference: 6 Derived from: Space Shuttle, ET, and Saturn launch vehicles. Options: None.	Space Shuttle Comparison
Shuttle uses the first 7 equations listed. $M_{tps\_fuse} = 1.366 A_{body}$ Fuselage TPS $M_{tps\_wing} = 2.845(2S_{exp})$ Wing TPS $M_{tps\_vert} = 1.572(2S_{vert})$ Vertical control surface TPS $M_{tps\_bf} = 3.468(2S_{bf})$ Body flap TPS $M_{tps\_base} = 0.82 A_{ref} T_{vac} N_{eng} / 1e^{6}$ Base TPS For boosters $T_{vac}N_{eng}$ should be replaced with T_{sls}. $M_{tps\_sb} = 1.366 A_{sb}$ Speed brake TPS $M_{insulation} = 0.508 A_{body}$ Body insulation $M_{ox\_\tan k\_ins} = 0.2574 S_{ox}$ Oxidizer tank insulation. For boosters only cryogenic oxidizer is insulated. $M_{f\_\tan k\_ins} = 0.2361 S_{f}$ Fuel tank insulation. All cryogenic fuels are insulated. Non cryogenic fuels are not.	2%

$M_{mono\_tank\_ins} = 0.2574 S_{mono\_tank}$ Mono-propellant tank insulation. All cryogenic mono-propellants are insulated except on booster stages. A_{ref} – reference aerodynamic area (front projected shadow area) A_{sb} – exposed surface area of speed brakes A_{body} – surface area of vehicle body S_{exp} – Planform area of exposed wing S_{vert} – Total planform area of vertical tail or wingtip fins S_{bf} – Planform area of body flap T_{vac} – Vacuum thrust per main engine N_{eng} – Number of main engines on stage S_{ox} – Surface area of oxidizer tanks S_f – Surface area of fuel tanks S_{mono_tank} – Surface area of monopropellant tank	

<table>
<tr><td>Reference: 10
Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing).
Options: TPS technology.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>Shuttle comparison uses tiles, and blankets, not including wing leading edge, or nose cap RCC.

$M_{tps} = K_{tps} A_{body\_tps} R_{type} + K_{wtps} R_{type} \left(2S_{\exp} + 2S_{vert} + 2S_{bf}\right) + 0.2 A_{body\_tps}$ Including insulation.
R_{type} – Percentage of TPS area covered by the type of TPS used for K_{tps}
K_{tps} – Mass per area of chosen TPS type
= 0.63 – body metallic TPS
= 1.67 – body blanket TPS
= 1.50 – body tile TPS
= 2.25 – body HEX panel TPS (active cooling)
K_{wtps} – wing, body flap, tail, and control surface TPS mass per area
= 1.59 – wing metallic TPS
= 0.49 – wing blanket TPS
= 1.50 – wing tile TPS

$M_{nose} = \pi \left(\frac{D_{nose}}{2}\right)^2 \left(0.0002499 q_{\max} + 1.7008 + \left(0.00003695 q_{\max} - 0.003252\right) D_{nose}\right)$ (source 10a) Body or TPS
For semispherical nose cap with passive TPS.

$M_{sharp} = 280 H_{le}^2 \tan(\psi_{le}) L_{le}$
For thin leading edges using the sharp TPS (density = 280 lb/ft^3)

$M_{active} = L_{le} 5.75$
For thin nose leading edge and wing and tail leading edges with active cooling.</td><td>-16%</td></tr>
</table>

A_{body_tps} – Wetted area of TPS on vehicle body D_{nose} – Diameter of base of nosecone H_{le} – Height of leading edge L_{le} – Length of leading edges (wing and nose if applicable) q_{max} – Maximum dynamic pressure during flight (lb/ft^2) S_{bf} – Planform area of body flap S_{exp} – Planform area of exposed wing S_{vert} – Total planform area of vertical tail or wingtip fins ψ_{le} – Leading edge angle **(? Sweep or angle of airfoil nose)**	

5.0 Landing Gear

L_{mg} – Length of main landing gear
L_{ng} – Length of nose landing gear
M_{glow} – Gross liftoff mass
M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff
M_{land} – Landed mass of vehicle
N_{land} = (number of gear)*1.5 – ultimate landing load factor
N_{mgw} – Total number of wheels on main gear
N_{ngw} – Total number of wheels on nose gear

<table>
<tr><td>Reference: 1
Derived from: Aircraft and Space Shuttle.
Options: Materials, and skids or wheels.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{\mathrm{lg}} = K_{\mathrm{lg}} M_{land}$
K_{lg} = 0.033 – shuttle gear
= 0.030 – advanced composite gear
= 0.0255 – composite skid system with no brakes

M_{land} – Landed mass of vehicle</td><td>15%</td></tr>
</table>

Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: None.	Space Shuttle Comparison
Same equation as above, but additionally the ratio of nose gear/main gear is 15%/85% K_{lg} = 0.026 – advanced landing gear	-9%

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
Same equation as above. K_{lg} = 0.033 – shuttle technology	15%

Reference: 4a Derived from: Airbreathing booster. Options: None.	Space Shuttle Comparison
$M_{\mathrm{lg}} = 0.0357 M_{gross}$ Gear weight for horizontal takeoff airbreathing booster vehicle M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff	N/A Horizontal takeoff only

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{\lg} = (0.036 + 0.0061 + 0.002 + 0.0008) M_{land}$ Landing gear for a second stage vehicle. Includes nose and main gear, gear bays, and attachment. M_{land} – Landed mass of vehicle	56%

<table>
<tr><td>Reference: 5
Derived from: Aircraft.
Options: Different gear styles.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>

$$M_{maingear} = K_{cb} K_{tpg} \left(M_{land} N_{land}\right)^{0.25} L_{mg}^{0.973}$$

$$M_{nosegear} = \left(M_{land} N_{land}\right)^{0.29} L_{ng}^{0.5} N_{ngw}^{0.525}$$

K_{cb} = 2.25 – for cross beam (F-111)
= 1.0 – all other gear
K_{tpg} = 0.826 – for tripod gear (A-7)
= 1.0 – all other gear

N_{land} = (number of gear)*1.5 – ultimate landing load factor
L_{ng} – Length of nose landing gear
L_{mg} – Length of main landing gear
N_{ngw} – Total number of wheels on nose gear

</td><td>-44%</td></tr>
</table>

Reference: 6 Derived from: Space Shuttle and aircraft. Options: None.	Space Shuttle Comparison
$M_{maingear} = 0.00927 M_{land}^{1.0861}$ $M_{nosegear} = 0.001514 M_{land}^{1.0861}$ For shuttle technology. M_{land} – Landed mass of vehicle	8%

<table>
<tr><td>Reference: 7
Derived from: Aircraft.
Options: Wing location.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>Used M_{land} instead of M_{glow} for comparison to Shuttle.

$M_{lg} = K_{lg}\left[K_a + K_b M_{glow}^{3/4} + K_c M_{glow} + K_d M_{glow}^{3/2}\right]$ Torenbeek method
For USAF airplanes, coefficients for other civil planes with retractable gear.

K_{lg} = 1.0 – low wing planes
= 1.08 – high wing planes
K_a = 40.0 – main, 20.0 – nose
K_b = 0.16 – main, 0.10 – nose
K_c = 0.019 – main, 0.00 – nose
K_d = 1.5e-5 – nose, 2.0e-6 – nose

M_{glow} – Gross liftoff mass
M_{land} – Landed mass of vehicle</td><td>37%</td></tr>
<tr><td>Used M_{glow} for comparison to Shuttle. Use of M_{land} produced very low gear weight.

$M_{lg} = 62.61\left(\frac{M_{glow}}{1000}\right)^{0.84}$ General Dynamics method

For USAF airplanes, fighter/attack aircraft.

M_{glow} – Gross liftoff mass
M_{land} – Landed mass of vehicle</td><td>10%</td></tr>
</table>

Used M_{glow} for comparison to Shuttle. Use of M_{land} produced very low gear weight. $M_{lg} = 129.1\left(\frac{M_{glow}}{1000}\right)^{0.66}$ Torenbeek method For USN airplanes, fighter/attack aircraft. M_{glow} – Gross liftoff mass M_{land} – Landed mass of vehicle	-17%

Reference: 8 Derived from: Aircraft. Options: Skid or wheel gear.	Space Shuttle Comparison
Method A $M_{\lg} = 0.096 M_{land}^{0.9} K_1$ K_1 = 0.6 – for skid gear = 1.0 – for wheeled gear **?** M_{land} – Landed mass of vehicle	-2%
Method B $M_{maingear} = \left[0.001 M_{land}^{0.75}\left(173 N_{mgw}^{0.14} K_1 + 35.2 L_{mg}^{0.44} K_2\right)\right]\left(1 + 0.06 K_3\right)$ Skid gear – K_1 = 0.21, K_2 = 0.52, K_3 = 0.27 ; wheeled gear – $K_1 = K_2 = K_3$ = 1.0 $M_{nosegear} = \left[0.001 M_{land}^{0.75}\left(18.9 K_1 + 9.48 L_{ng}^{0.44} K_2\right)\right]\left(1 + 0.08 K_3\right)$ Skid gear – K_1 = 1.59, K_2 = 1.77, K_3 = 0.063 ; wheeled gear – $K_1 = K_2 = K_3$ = 1.0 M_{land} – Landed mass of vehicle	-41%

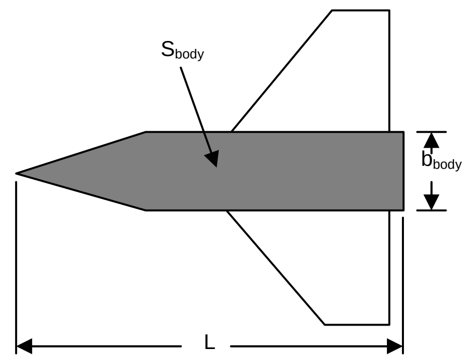
Reference: 9 Derived from: Aircraft. Options: Development risk, number of wheels.	Space Shuttle Comparison
For this reference use M_{glow} even for vertical take-off vehicles. This was used for Shuttle comparison since using M_{land} produces very low weight gear. For comparison to Shuttle, *TRF*=1.0 for both main and nose gear. $M_{main\_running\_gear} = 9.61 M_{glow} 0.001 N_{mgw}^{0.14}$ N_{mgw} – rule of thumb = $M_{glow}/50{,}000$ $M_{main\_gear\_struct} = \left(3.1 M_{land} 0.001 L_{mg}^{0.44}\right) TRF$ $M_{main\_gear\_control} = 0.18\left(M_{main\_running\_gear} + \frac{M_{main\_gear\_struct}}{TRF}\right)$ *TRF* = 0.85 – low development risk = 0.80 – moderate to high development risk = 0.70 – very high development risk $M_{nose\_running\_gear} = 1.25 M_{glow} 0.001$ $M_{nose\_gear\_struct} = \left(0.5 M_{land} 0.001 L_{ng}^{0.44}\right) TRF$ $M_{nose\_gear\_control} = 0.3\left(M_{nose\_running\_gear} + \frac{M_{nose\_gear\_struct}}{TRF}\right)$ *TRF* = 0.80 – advanced materials, all risk levels	5%

L_{mg} – Length of main landing gear L_{ng} – Length of nose landing gear M_{glow} – Gross liftoff mass M_{land} – Landed mass of vehicle N_{mgw} – Total number of wheels on main gear	

Reference: 10c Derived from: Aircraft from General Dynamics Study. Options: TPS technology.	Space Shuttle Comparison
$M_{nose\_gear} = 0.0033995M + 402$ $M_{main\_gear} = 0.02366M + 1161$ The mass, *M*, in these equations can be either the landing mass or the GLOW depending on whether the vehicle is vertical or horizontal take-off.	19%

6.0 Main Propulsion

A_{mixer} – is the cross sectional area of the mixer.
F_{ullage} – Ullage fraction (typically ~4 to 5%)
Isp – Specific impulse of engines
Isp_{sl} – Specific impulse of engine at sea level
$\dot{m}$ – Total propellant mass flow rate (lbm/s)
M_{glow} – Gross liftoff mass
N_{eng} – Number of main engines on stage
P_c – Pressure of main engine combustion chamber
P_{tank} – Pressure of propellant tanks
R_{aox} – ascent oxidizer fraction
R_{eng} – engine thrust to weight at vacuum conditions, installed
R_{v_lo} – Vehicle thrust to weight at liftoff
S_{body} – Planform area of vehicle body
T_{sls} – Total stage thrust at sea level static conditions
T_{vac} – Vacuum thrust per main engine
V_f – Total fuel volume
V_{ox} – Total oxidizer volume
V_{prop_tot} – total volume of propellant carried.
ε_i – Expansion ratio of nozzle of engine number i



Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Propellant type, and chamber pressure.	Space Shuttle Comparison
$M_{main\_prop} = \left[K_{ph} + K_n(\varepsilon_1 - 1) + K_{ne}\frac{(\varepsilon_2 - \varepsilon_1)}{P_c} + K_{na}\frac{(\varepsilon_2^{0.5} - 1)}{(\dot{m}P_c)^{0.5}} + K_{pf} + K_{ga}I_{sp} \right] N_{eng}\dot{m}$ Rocket engine prediction only. Power head constants K_{ph} = 5.34 – LOX/LH2, Pc = 3000psi = 5.18 – dual fuel engine, Pc = 3000psi = 2.48 – LOX/hydrocarbon staged combustion, Pc = 4000psi = 2.10 – LOX/hydrocarbon LH2 generator, Pc = 4000psi Nozzle constants K_n = 0.01194 – LOX/LH2 = 0.00727 – LOX/hydrocarbon = 0.015 – EN 155 (dual fuel) Nozzle extension constants K_{ne} = 9.943 – LOX/LH2 = 6.054 – LOX/hydrocarbon Nozzle extension actuator K_{na} = 60.54 – LOX/LH2 = 36.86 – LOX/hydrocarbon Pressurization and feed system constants	-15%

K_{pf} = 1.64 – current technology (1978) = 1.40 – composite/metallic feedlines Gimbal actuators K_{ga} = 0.00129 – hydraulic system (assumed due to publish date) $\dot{m}$ – Total propellant mass flow rate (lbm/s) Isp – Specific impulse of engines N_{eng} – Number of main engines on stage P_c – Pressure of main engine combustion chamber ε_i – Expansion ratio of nozzle of engine number i	

<table>
<tr><td>Reference: 2
Derived for: Airbreathing horizontal takeoff vehicle.
Options: Supercharging or not, supersonic combustion or not.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>Rocket based combined cycle engine mass.
$$M_{engines} = M_{glow}\frac{R_{v\_lo}}{R_{eng\_ui}}$$
R_{eng_ui} = Engine uninstalled thrust to weight
$= 3.99\dot{m} + 114A_{mixer}$ no inlet, no supercharging fan
$= 4.04\dot{m} + 200.5A_{mixer}$ no inlet, with supercharging fan
$$M_{press\_feed} = 1.616M_{glow}\frac{R_{v\_lo}}{Isp_{sl}}$$
$M_{purge\_syst} = (0.05V_f + 0.075V_{ox})(1 - TRF)$ for purging lines and tanks with He

A_{mixer} – is the cross sectional area of the mixer.
Isp_{sl} – Specific impulse of engine at sea level
M_{glow} – Gross liftoff mass
R_{v_lo} – Vehicle thrust to weight at liftoff
TRF = 0.6 – AMLS (from Lepsch)
V_f – Total fuel volume
V_{ox} – Total oxidizer volume</td><td>N/A
RBCC only</td></tr>
</table>

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{main\_prop} = (0.0205 T_{vac}) N_{eng}$ Equation for rocket engines N_{eng} – Number of main engines on stage T_{vac} – Vacuum thrust per main engine	-7%

Reference: 4a Derived from: Airbreathing booster. Options: Airbreathing engine type.	Space Shuttle Comparison
$M_{engines} = M_{glow} \frac{R_{v\_lo}}{R_{eng\_ui}}$ For airbreathing booster vehicle using composite propulsion R_{eng_ui} – uninstalled engine thrust to weight = 160 – Air augmented rocket = 31.40 – Ejector ramjet, max internal pressure of 150 psia = 29.00 – Ejector scramjet, max internal pressure of 100 psia = 26.45 – Supercharged ejector ramjet, max internal pressure of 150 psia = 24.07 – Supercharged ejector scramjet, max internal pressure of 100 psia = 19.36 – RL, max internal pressure of 150 psia = 16.80 – SL, max internal pressure of 100 psia = 16.21 – RRL, max internal pressure of 150 psia = 13.95 – RSL, max internal pressure of 100 psia = 17.92 – SRL, max internal pressure of 150 psia = 14.80 – SSL, max internal pressure of 100 psia = 15.29 – RSRL, max internal pressure of 150 psia = 12.53 – RSSL, max internal pressure of 100 psia $M_{eng\_controls} = 0.0012 T_{sls}$ Mass of engine control system $M_{fuel\_dist} = 0.004 T_{sls}$ Mass of liquid hydrogen distribution, purge, and vent system $M_{ox\_dist} = 0.003 T_{sls}$ Mass of liquid oxygen distribution, purge, and vent system	N/A RBCC only

M_{glow} – Gross liftoff mass R_{v_lo} – Vehicle thrust to weight at liftoff T_{sls} – Total stage thrust at sea level static conditions	

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{engines} = 0.0146 T_{vac} N_{eng} + 300$ Mass of LOX/LH2 engines for a second stage vehicle. $M_{eng\_attach} = 0.00138 T_{vac} N_{eng}$ Mass of engine attachment hardware. $M_{prop\_dist} = 0.445 S_{body}$ Mass of propellant distribution system for LOX/LH2 $M_{press\_vent} = 0.0672 V_{prop\_tot}$ Mass of pressurization and vent system for LOX/LH2 N_{eng} – Number of main engines on stage S_{body} – Planform area of vehicle body T_{vac} – Vacuum thrust per main engine V_{prop_tot} – total volume of propellant carried.	-4% Uses ET volume

Reference: 6 Derived from: Space Shuttle, ET, and Saturn launch vehicles. Options: Feed system type.	Space Shuttle Comparison
Shuttle comparison uses the LOX/LH2 engines and includes engine install, subsystems, TVC, feed (K_{feed}=2.197), purge (using volume of ET), and pressurization for pump fed engines. $M_{engines} = \frac{T_{vac} N_{eng}}{\min\left(75 : \left(5.11 \ln(T_{vac} N_{eng}) + 4.2\right)\right)}$ For rocket powered vehicles, LOX/LH2 $M_{engines} = \frac{T_{vac} N_{eng}}{\min\left(104.4 : \max\left(20.3 : 26.04 \ln(T_{vac} N_{eng}) - 207\right)\right)}$ For rocket powered vehicles using LOX/RP or N2O4/MMH propellants $M_{eng\_install} = 5.6e^{-4} T_{vac} N_{eng}$ Engine installation (bolts, connectors, etc…) $M_{eng\_subsystem} = 5.6e^{-4} T_{vac} N_{eng}$ Engine subsystems. $M_{tvc} = 0.001185 T_{vac} N_{eng}$ Thrust vector control $M_{feed} = K_{feed} \dot{m}\left(1 + 0.04 * if\,(crossfeed, 1, 0)\right)$ Propellant feed system K_{feed} – Propellant feed system constant = 2.197 – Orbiter & ET configuration = 1.482 – Orbiter without propellant tanks = 0.715 – ET type tank only	-5%

= 2.133 – Upper stage/orbiter with internal tanks
= 1.022 – Booster or monopropellant feed system (upper or lower stage)

$M_{purge} = 0.053 V_{body}$ Purge system

$M_{press} = 0.192 \dot{m}$ Booster or US type configuration, cryo propellants, autogenous system, pump-fed engines

$M_{press} = 50 + 0.192 \dot{m} + \frac{F_{ullage}}{26} 0.18 V_{prop\_tot}$ Storable stage, ambient stored He with heat exchange system.

$M_{press} = K_{press} (1.3012 + 0.99 P_{tank}) V_{prop\_tot}^{(0.8647 P_{tanks}^{0.01645})}$

K_{press} – pressure fed engine system constant
= 0.55 – pressure fed engine, cold N2/GH2
= 0.25 – pressure fed engine, hot N2/GH2
= 0.19 – pressure fed engine, gas generator system

F_{ullage} – Ullage fraction (typically ~4 to 5%)
$\dot{m}$ – Total propellant mass flow rate (lbm/s)
N_{eng} – Number of main engines on stage
P_{tank} – Pressure of propellant tanks
T_{vac} – Vacuum thrust per main engine
V_{body} – Volume of vehicle body
V_{prop_tot} – total volume of propellant carried.

<table>
<tr><td>Reference: 10
Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing).
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>

$$M_{engines} = \frac{T_{vac} N_{eng}}{R_{eng}}$$

$$M_{f\_dist} = 6.625\left(\frac{T_{sls}}{Isp_{sl}}\right)(1 - R_{aox})(1 + 0.04 * if\,(crossfeed,1,0))$$ fuel distribution

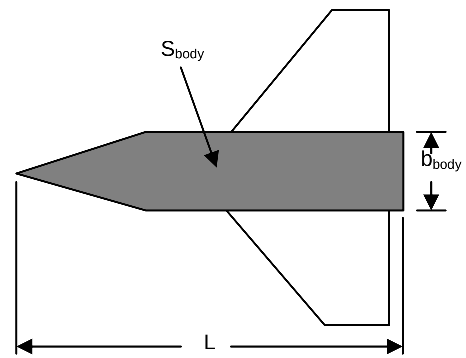
$$M_{ox\_dist} = 6.625\left(\frac{T_{sls}}{Isp_{sl}}\right)(R_{aox})(1 + 0.04 * if\,(crossfeed,1,0))$$ oxidizer distribution

$$M_{vppd} = 0.001366 T_{sls} + 0.192\left(\frac{T_{sls}}{Isp_{sl}}\right)$$ Vehicle purge, pressurization, and dump system (source 10a)

Isp_{sl} – Specific impulse of engine at sea level
N_{eng} – Number of main engines on stage
R_{aox} – ascent oxidizer fraction
R_{eng} – engine thrust to weight at vacuum conditions, installed
T_{vac} – Vacuum thrust per main engine
T_{sls} – Total stage thrust at sea level static conditions

</td><td>-34%</td></tr>
</table>

7.0 RCS



L – Length of vehicle
M_{dry} – Dry mass of vehicle
M_{entry} – Entry mass of vehicle
M_{insert} – Insertion mass, sometimes called burnout mass
M_{land} – Landed mass of vehicle
M_{pl} – Mass of payload
$M_{rcs\_propellants}$ – Total mass of all RCS propellants
M_{resid} – Mass of residual propellants
N_{vt} – Number of vernier thrusters
P_{rcs_press} – Pressure of rcs pressurization system tanks
P_{rcs_tank} – Pressure of RCS tank
R_{vt}– Vernier thruster thrust to weight
T_{req} – Required thrust from vernier thrusters for RCS system
T_{req_p} – Required thrust for primary thrusters
V_{rcs_f} – Volume of RCS fuel
V_{rcs_ox} – Volume of RCS oxidizer
V_{rcs_press} – Volume of He required as pressurant
V_{rcs_tanks} – Volume of all RCS tanks

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Storable or cryogenic propellants.	Space Shuttle Comparison
$M_{rcs} = K_{rcs} M_{entry} L$ K_{rcs} = 1.36e-4 – based on shuttle storable system = 1.51e-4 – based on advanced cryogenic system M_{entry} – Entry mass of vehicle	10%

<table>
<tr><td>Reference: 2
Derived for: Airbreathing horizontal takeoff vehicle.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>For shuttle comparison the larger thrusters (both front and rear) were considered primary, and the smaller were vernier. The actual thrust was used instead of the estimating equation provided.

Forward RCS
$M_{rcs\_vt} = N_{vt}\frac{T_{req}}{R_{vt}}$
Pressure fed LOX/LH2 from Rockwell IHOT study and AMLS
T_{req} – Required thrust from vernier thrusters = $\left[\frac{M_{entry}L50}{147141(143)}\right]$
N_{vt} = 15 – (3 in each direction plus forward) for forward RCS

$M_{rcs\_\tan k} = 0.01295 P_{rcs\_\tan k} V_{rcs\_\tan k}$ Al 2219, yield at 140% Prcs_tank, 1.75 NOF, 5% ullage

$M_{rcs\_press} = 0.0143 P_{rcs\_press} V_{rcs\_press}(1 - TRF) + 0.671(V_{rcs\_ox} + V_{rcs\_f})$ Pressurization system
Ti 6/4 tank, 3000psia, He, yield at 400% Prcs_press, 1.25 NOF, 400 R storage temp.

$M_{rcs\_install} = 0.74 M_{rcs\_vt}$ Installation hardware, lines, manifolds, etc…

Aft RCS
$M_{rcs\_vt} = N_{vt}\frac{T_{req}}{R_{vt}} + N_{primary}\frac{T_{req\_p}}{R_{primary}}$</td><td>-68%</td></tr>
</table>

LOX/LH2 from Rockwell IHOT study and AMLS

T_{req} – Required thrust for vernier thrusters = $\left[\dfrac{M_{entry} L50}{147141(143)}\right]$

T_{req_p} – Required thrust for primary thrusters = $\left[\dfrac{M_{entry} L870}{147141(143)}\right]$

N_{vt} = 12 for aft RCS

Propellant tanks, pressurization system, and lines & manifolds use the same equations as for the forward RCS list above.

M_{entry} – Entry mass of vehicle
$N_{primary}$ – number of primary thrusters, recommended = 10
N_{vt} – number of verier thrusters
P_{rcs_press} – Pressure of rcs pressurization system tanks, typically = 3000 psia for He
P_{rcs_tank} – Pressure of RCS tank = 195 – for both LOX and LH2 tanks
$R_{primary}$ – thrust to weight of primary thrusters = 39.5
R_{vt} – thrust to weight of vernier thrusters = 9.4
T_{req_p} – required thrust for primary thrusters
T_{req} – Required thrust for RCS system
TRF – Techology reduction factor = 0.0 for baseline, = 0.25 – for composite wound tanks
V_{rcs_ox} – Volume of RCS oxidizer
V_{rcs_f} – Volume of RCS fuel
V_{rcs_press} – Volume of He required as pressurant= $0.24(V_{rcs\_ox} + V_{rcs\_f})$
V_{rcs_tanks} – Volume of all RCS tanks

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{rcs} = 0.014 M_{entry}$ Assumes shuttle technology M_{entry} – Entry mass of vehicle	6%

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{rcs} = 0.0171 M_{land}$ Mass of attitude control system (includes OMS and RCS) for second stage. M_{land} – Landed mass of vehicle	20%

Reference: 6 Derived from: Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{rcs} = 0.0126 M_{insert}$ Assumes shuttle technology M_{insert} – Insertion mass, sometimes called burnout mass	8%

Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: None.	Space Shuttle Comparison
$M_{rcs} = 1184\left[\left(\left(M_{dry} + M_{pl} + M_{resid}\right)/234948\right)^{0.217}\left(\frac{L}{205}\right)^{0.434}\right]$ RCS system for airbreathing vehicle L – Length of vehicle M_{dry} – Dry mass of vehicle M_{pl} – Mass of payload M_{resid} – Mass of residual propellants (group 20.0)	N/A

Reference: 10a Derived from: Alpha Technologies, rocket based. Options: TPS technology.	Space Shuttle Comparison
$M_{rcs} = 0.008M_{insert} + 0.0046M_{insert}\left(\frac{\sum M_{rcs\_propellants}}{6600}\right)$ RCS system for a rocket vehicle M_{insert} – Insertion mass, sometimes called burnout mass $M_{rcs\_propellants}$ – Total mass of all RCS propellants	13%

8.0 OMS

M_{entry} – Entry mass of vehicle
M_{insert} – Insertion mass, sometimes called burnout mass
M_{oms_prop} – Mass of all OMS propellants
N_{oms} – Number of OMS engines
P_{oms_press} – Design pressure of OMS pressurization system tanks
P_{oms_tank} – Design pressure of OMS propellant tank
R_{oms} – OMS engine thrust to weight
T_{oms_vac} – Vacuum thrust of each OMS engine
T_{req_oms} – Required thrust from OMS engines
V_{oms_f} – Volume of OMS fuel
V_{oms_ox} – Volume of OMS oxidizer
V_{oms_press} – Volume of pressurant required
V_{oms_tank} – Volume of OMS tank

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Storable or cryogenic propellants.	Space Shuttle Comparison
$$M_{oms} = K_{oms} N_{oms} T_{oms\_vac} + K_{pps} M_{oms\_prop}$$ K_{oms} – Orbital maneuver system thruster constant = 0.0863 – based on shuttle storable propellants = 0.035 – based on advanced cryogenic propellants/engine K_{pps} – OMS propellant supply system = 0.119 – for storable propellants including pressurization = 0.152 – for cryogenic propellants including pressurization and feed M_{oms_prop} – Mass of all OMS propellants N_{oms} – Number of OMS engines T_{oms_vac} – Vacuum thrust of each OMS engine	-15%

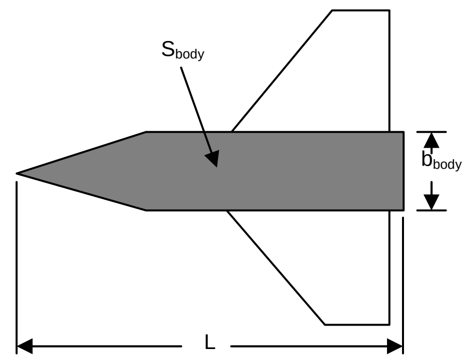
Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: None.	Space Shuttle Comparison
$M_{oms\_eng} = \frac{T_{req\_oms}}{R_{oms}}$ $M_{oms\_tank} = 0.01295 P_{oms\_tank} V_{oms\_tank}$ Al 2219, yield at 140% Prcs_tank, 1.75 NOF, 5% ullage $M_{oms\_press} = 0.0143 P_{oms\_press} V_{oms\_press} (1 - TRF) + 0.167 (V_{oms\_ox} + V_{oms\_f})$ Pressurization system Ti 6/4 tank, 3000psia, He, yield at 400% Prcs_press, 1.25 NOF, 400 R storage temp. $M_{oms\_install} = 0.74 M_{oms\_eng}$ Installation hardware, lines, manifolds, etc… R_{oms} – OMS engine thrust to weight = 22 (includes mounts, supports, igniters, etc.) P_{oms_press} – Design pressure of OMS pressurization system tanks, typically = 3000 psia for He P_{oms_tank} – Design pressure of OMS propellant tank *TRF* – Techology factor = 0.0 for baseline, = .25 – for composite wound tanks V_{oms_f} – Volume of OMS fuel V_{oms_ox} – Volume of OMS oxidizer V_{oms_press} – Volume of pressurant required (He) = 0.24(V_{oms_ox} + V_{oms_f}) V_{oms_tank} – Volume of OMS tank T_{req_oms} – Required thrust from OMS engines = M_{entry}/16 (1/16th g accel/decal)	106% Not intended for 7 ksi tank pressure used in shuttle

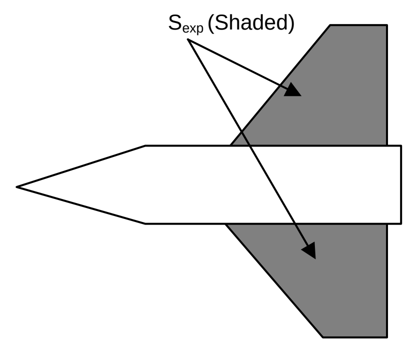
Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{oms} = 0.0146 M_{entry}$ M_{entry} – Entry mass of vehicle	14%

Reference: 6 Derived from: Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{oms} = 0.0121 M_{insert}$ M_{insert} – Insertion mass, sometimes called burnout mass	7%

Reference: 10a Derived from: Alpha Technologies, rocket based. Options: None.	Space Shuttle Comparison
$M_{oms} = 0.0045 M_{insert} + 0.0076 M_{insert} \left(\frac{\sum M_{oms\_prop}}{24175} \right)$ OMS for a rocket vehicle M_{insert} – Insertion mass, sometimes called burnout mass M_{oms_prop} – Mass of all OMS propellants	-5%

9.0 Primary Power





L – Length of vehicle
M_{apu_prop} – mass of all APU propellants on board
M_{av} – Mass of avionics (group 13.0)
M_{glow} – Gross liftoff mass
M_{land} – Landed mass of vehicle
M_{sca} – Mass of surface control & actuators (group 12.0)
N_{apu} – Number of APUs
N_{crew} – Number of crew
N_{days} – Number of days on orbit
N_{eng} – Number of main engines on stage
N_{fc} – number of fuel cells
P_{apu} – Power required per APU
P_{fc} – power required per fuel cell
S_{bf} – Planform area of body flap
S_{exp} – Exposed wing planform area
S_{tot_cont} – Total planform area of all control surfaces
S_{vert} – Total planform area of vertical tail or wingtip fins
T_{vac} – Vacuum thrust per main engine
T_{vac_gimb} – Total vacuum thrust of gimbaled engines

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Standard or accumulators.	Space Shuttle Comparison
$M_{pp} = K_{pc} S_{tot\_cont} + K_{pe} T_{vac\_gimb} + K_{pb} M_{av}$ K_{pc} = 0.712 – Standard hydraulic system = 0.610 – Hydraulic with accumulators for peak load K_{pe} = 0.97e-4 – Engine gimbal power demand K_{pb} = 0.405 – Battery power demand constant S_{tot_cont} – Total planform area of all control surfaces T_{vac_gimb} – Total vacuum thrust of gimbaled engines M_{av} – Mass of avionics (group 13.0)	-18%

<table>
<tr><td>Reference: 2
Derived for: Airbreathing horizontal takeoff vehicle.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{pp} = 396 + 176.9(N_{days} + 1) + 0.05166 M_{sca}$
Based on NASP technology (Stanley). Assumes fuel cells are 396 lb.

M_{sca} – Mass of surface control & actuators (group 12.0)
N_{days} – Number of days on orbit</td><td>-36%</td></tr>
</table>

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{pp} = 977 + 139.6 N_{days} + 39.9 N_{days} N_{crew}$ Includes fuel cells, batteries, and associated systems. N_{days} – Number of days on orbit N_{crew} – Number of crew	32%

Reference: 4a Derived from: Airbreathing booster. Options: None.	Space Shuttle Comparison
$M_{pp} = 1400 + 0.0017 M_{glow}$ Primary power for airbreathing booster vehicle. Entire power system, including conversion and distribution? M_{glow} – Gross liftoff mass	-52%

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{apu} = 10N_{crew} + 831$ Mass of auxiliary power unit for manned second stage. $M_{elec\_power} = 800$ Mass of other components, 12 people. N_{crew} – Number of crew	-57%

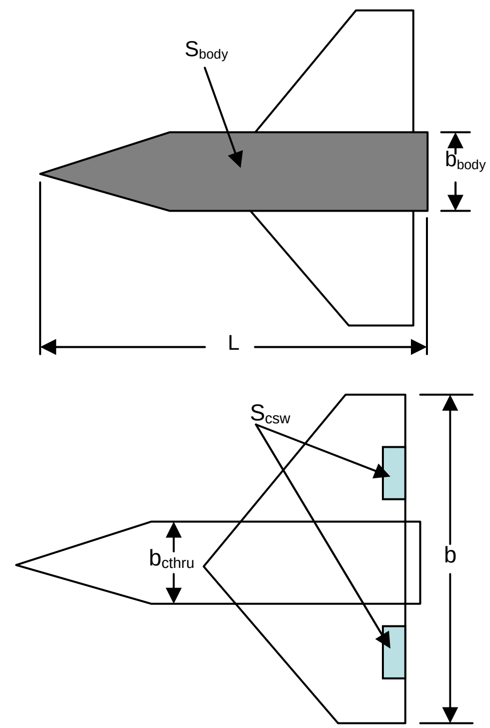
Reference: 6 Derived from: Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{batt} = 216 + 952 \frac{N_{days}}{7} * if(N_{crew} > 0,1,0)$ Battery mass, unmanned missions only **Batteries in unmanned or manned?** $M_{batt} = 216$ Battery mass for manned missions $M_{fuel\_cell} = 3030 \frac{N_{crew}}{7}$ Fuel cell mass for manned missions only N_{crew} – Number of crew N_{days} – Number of days on orbit	-17%

Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: None.	Space Shuttle Comparison
$M_{pp} = 227 \frac{L}{205} \frac{N_{apu}}{4} + 18.97 P_{apu}^{0.5} + \left(613.8 \frac{L}{205} + 0.66 P_{apu} \right) \frac{N_{apu}}{4}$ Power system for airbreather L – Length of vehicle N_{apu} – Number of APUs P_{apu} – Power required per APU	N/A

Reference: 10a Derived from: Alhpa Technologies, rocket based. Options: None.	Space Shuttle Comparison
The following equations should be summed to get M_{pp} for a rocket powered vehicle. $M_{batt} = \left(216 + 952\frac{N_{days}}{7}\right) * if(N_{crew} > 0,1,0)$ $M_{fuel\_cell} = 51.8 N_{fc} P_{fc} + 0.76(52.143 N_{crew} N_{days})$ $M_{apu} = 0.118(0.00124 T_{vac} N_{eng} + 0.55 S_{\exp} + 3.4 S_{vert} + 2.6 S_{bf} + 0.000485 M_{land}^{1.0861}) + 0.318 M_{apu\_prop}^{1.15}$ M_{apu_prop} – mass of all APU propellants on board M_{land} – Landed mass of vehicle N_{crew} – Number of crew N_{days} – Number of days on orbit N_{eng} – Number of main engines on stage N_{fc} – number of fuel cells P_{fc} – power required per fuel cell S_{bf} – Planform area of body flap S_{exp} – Exposed wing planform area S_{vert} – Total planform area of vertical tail or wingtip fins T_{vac} – Vacuum thrust per main engine	93%

10.0 Electrical Conversion & Distribution

b – Wing span
b_{body} – Maximum width of the body
H_{body} – Height of body
L – Length of vehicle
M_{dry} – Dry mass of vehicle
M_{land} – Landed mass of vehicle
M_{sca} – Mass of surface control & actuators (group 12.0)
$M_{pascent}$ – Mass of ascent propellants
N_{crew} – Number of crew
N_{days} – Number of days spent on orbit
N_{gen} – Number of power sources onboard
R_{kva} – System electrical rating = Kvolts * Amps



Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Advanced technology or Shuttle technology.	Space Shuttle Comparison
$M_{ecd} = K_{ecd} M_{land}$ K_{ecd} = 0.02 – advanced ECD system = 0.038 – shuttle technology ECD system M_{land} – Landed mass of vehicle	-20%

Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: None.	Space Shuttle Comparison
$M_{ecd} = 1875 + 0.324 M_{sca} + K_{ecd} 8.56(L + H_{body} + b_{body}) + 0.00043(L + b) M_{sca}$ Assumes use of electro mechanical actuators for control surface actuation. K_{ecd} = 0.6 – shape factor for RBCC SSTO (low due to proximity of payload bay and crew cabin) b – Wing span b_{body} – Maximum width of the body H_{body} – height of body L – Length of vehicle M_{sca} – Mass of surface control & actuators (group 12.0)	-65%

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{ecd} = 0.062 M_{dry}$ M_{dry} – Dry mass of vehicle	-1%

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
See notes under 11.0 (hydraulics) for this.	N/A

Reference: 5 Derived from: Aircraft. Options: Mission redundancy.	Space Shuttle Comparison
$M_{ecd} = 172.2 K_{ecd} R_{kva}^{0.152} N_{crew}^{0.1} L^{0.1} N_{gen}^{0.091}$ K_{ecd} = 1.45 – If mission completion required after failure = 1.0 – Otherwise L – Length of vehicle N_{crew} – Number of crew N_{gen} – Number of power sources onboard R_{kva} – System electrical rating = Kvolts * Amps	-91%

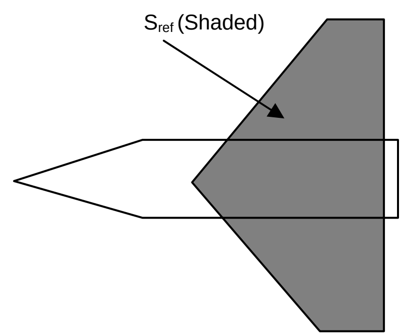
Reference: 6 Derived from: Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{ecd} = 793 + 506\frac{M_{pascent}}{1.6e^{-6}} + 2226\left(\frac{N_{days}}{7}\right) + 7633\left(\frac{N_{crew}}{7}\right)$ $M_{pascent}$ – Mass of ascent propellants N_{crew} – Number of crew N_{days} – Number of days spent on orbit	2%

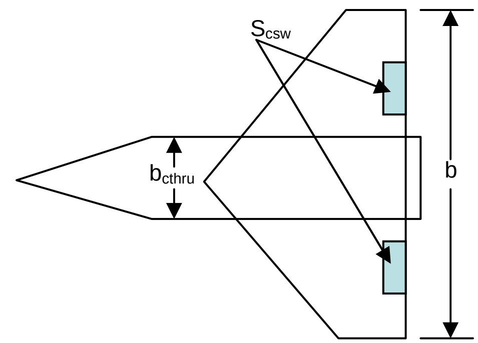
Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: None.	Space Shuttle Comparison
$M_{ecd} = 1201\left(0.90674 + 0.09326\frac{L}{205}\right)$ For an airbreathing vehicle *L* – Length of vehicle	N/A

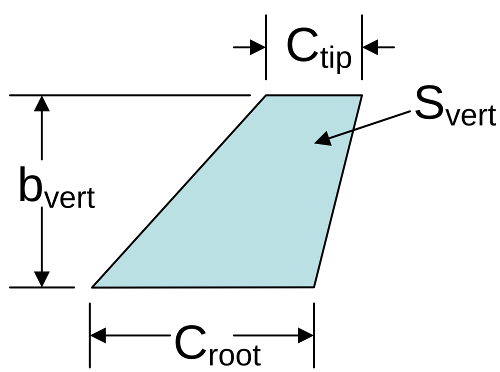
Reference: 10a Derived from: Alpha Technologies, rocket based. Options: None.	Space Shuttle Comparison
$M_{ecd} = 793 + 3.31L + 318N_{days} + 1096.4N_{crew}$ For a rocket powered vehicle L – Length of vehicle N_{crew} – Number of crew N_{days} – Number of days spent on orbit	6%

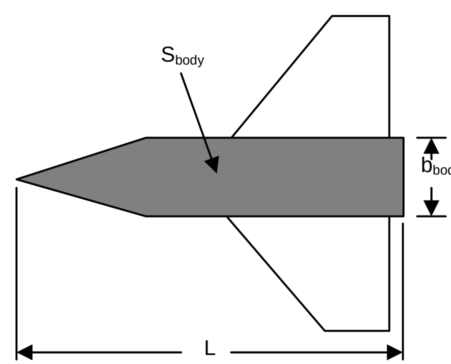
11.0 Hydraulic Systems

b_{body} – Maximum width of the body
b_{exp} – Span of exposed wing (b-b_{body} at wing root)
M_{glow} – Gross liftoff mass
M_{land} – Landed mass of vehicle
N_{eng} – Number of main engines on stage
N_{hyd} – Number of hydraulic functions on the vehicle
q_{max} – Maximum dynamic pressure during flight (lb/ft^2)
S_{bf} – Planform area of body flap
S_{body} – Planform area of vehicle body
S_{ref} – Theoretical wing planform area
S_{tot_cont} – Total planform area of all control surfaces
S_{vert} – Total planform area of vertical tail or wingtip fins
T_{vac} – Vacuum thrust per main engine
T_{vac_gimb} – Total vacuum thrust of gimbaled engines
θ_{le} – Sweep angle of leading edge









<table>
<tr><td>Reference: 1
Derived from: Aircraft and Space Shuttle.
Options: System pressure.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{hyd} = K_{hyd} S_{tot\_cont} + K_e T_{vac\_gimb}$

K_{hyd} = 2.10 – Shuttle technology base for hydraulic system
= 1.23 – For a 5000 psi system
K_e = 3.00e-4 – Shuttle gimbal technology
= 1.68e-4 – For a 5000 psi gimbal system

S_{tot_cont} – Total planform area of all control surfaces
T_{vac_gimb} – Total vacuum thrust of gimbaled engines</td><td>-2%</td></tr>
</table>

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{hyd} = 15.8 \frac{M_{land}}{S_{body}}$ For a lifting body rocket powered second stage. This was originally listed as support systems, and likely includes electrical conversion and distribution since the MBS it was included in did not have that as a separate item. M_{land} – Landed mass of vehicle S_{body} – Planform area of vehicle body	N/A

<table>
<tr><td>Reference: 5
Derived from: Aircraft.
Options: Variable or fixed sweep wings.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{hyd} = 37.23 K_{vsh} N_{hyd}^{0.664}$

K_{vsh} = 1.425 – for variable sweep wings
= 1.0 – for fixed wings

N_{hyd} – Number of hydraulic functions on the vehicle</td><td>-87%</td></tr>
</table>

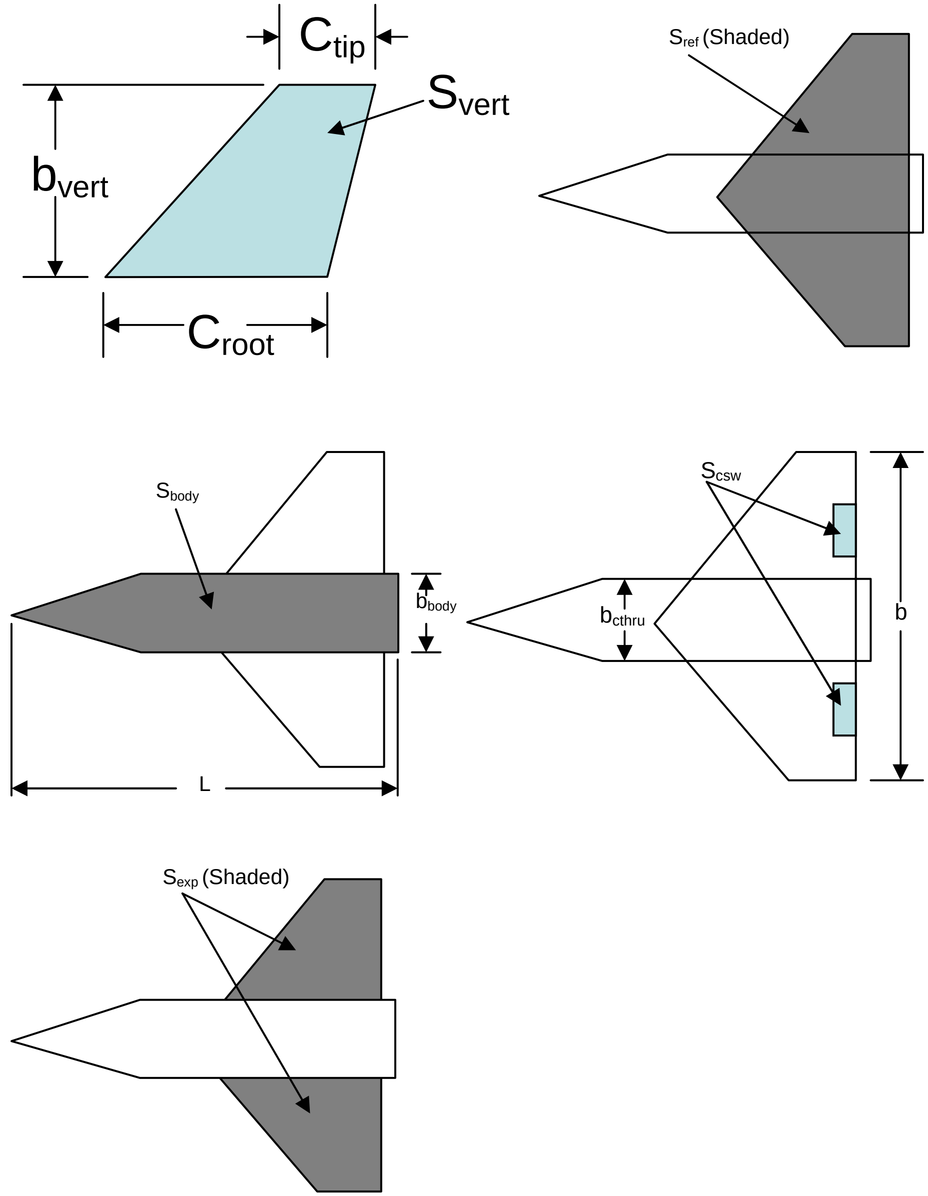
<table>
<tr><td>Reference: 6
Derived from: Power curve from Sigma (JSC study) adjusted to Space Shuttle.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$$M_{hyd} = 0.426\left[\left(S_{ref} + S_{vert} + S_{bf}\right)\frac{q_{\max}}{1000}\right]^{1.1143} + 0.001785 T_{vac} N_{eng}$$
N_{eng} – Number of main engines on stage
q_{max} – Maximum dynamic pressure during flight (lb/ft^2)
S_{bf} – Planform area of body flap
S_{vert} – Total planform area of vertical tail or wingtip fins
T_{vac} – Vacuum thrust per main engine</td><td>714%</td></tr>
</table>

Reference: 7 Derived from: Aircraft. Options: Hydraulic constant.	Space Shuttle Comparison
For the Space Shuttle comparison K_{hyd} = 0.0068 $M_{hyd} = K_{hyd} M_{glow}$ K_{hyd} = 0.005-0.0180 M_{glow} – Gross liftoff mass	0%

Reference: 10a Derived from: Alpha Technologies, rocket based. Options: None.	Space Shuttle Comparison
$$M_{hyd} = 0.326\left[\left(q_{\max} S_{ref} /1000\right)^{1.3125} + \left(L + \frac{\left(b_{\exp} + b_{body}\right)}{\cos(\theta_{le})}\right)^{1.6125}\right]^{0.849}$$ For rocket powered vehicles b_{body} – Maximum width of the body b_{exp} – Span of exposed wing (b-b_{body} at wing root) q_{max} – Maximum dynamic pressure during flight (lb/ft^2) S_{ref} – Theoretical wing planform area θ_{le} – Sweep angle of leading edge	455%

12.0 Surface Control & Actuators

M_{entry} – Entry mass of vehicle
M_{glow} – Gross liftoff mass of vehicle
M_{land} – Landed mass of vehicle
N_{crew} – Number of crew
N_{cs} – Number of flight control systems (redundancy)
R_{elevon} – Percent of wing that is elevon area
R_{vert} – Percent of vertical surfaces that are control surface
S_{bf} – Planform area of body flap
S_{body} – Planform area of the body
S_{canard} – canard planform area
S_{csw} – Planform of wing mounted control surfaces
S_{exp} – Exposed wing planform area
S_{htail} – horizontal tail planform area
S_{ref} – Theoretical wing planform area
S_{sb} – speed brake planform area
S_{tot_cont} – Total planform area of all control surfaces
S_{vert} – Total planform area of vertical tail or wingtip fins



Reference: 1 Derived from: Aircraft and Space Shuttle. Options: System pressure.	Space Shuttle Comparison
$M_{sca} = K_{sca} S_{tot\_cont} + K_{ms}$ based on hydraulic system from the Space Shuttle K_{sca} = 3.75 – for shuttle surface control and actuation technology = 3.80 – for 5000 psi system = 3.32 – for 5000 psi system using advanced materials K_{ms} = 200 – additional miscellaneous systems S_{tot_cont} – Total planform area of all control surfaces	-8%

Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: None.	Space Shuttle Comparison
$M_{sca} = 0.0163 R_{elevon} M_{entry} + 0.00428 R_{vert} M_{entry}$ Assumes EMA technology M_{entry} – Entry mass of vehicle R_{elevon} – Percent of wing that is elevon area = $\frac{S_{csw}}{S_{\exp}}$ R_{vert} – Percent of vertical surfaces that are control surface = $\frac{S_{rud}}{S_{vert}}$ S_{csw} – Planform of wing mounted control surfaces S_{exp} – Exposed wing planform area S_{vert} – Total planform area of vertical tail or wingtip fins	-58% different technology than shuttle

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{sca} = 0.0048 M_{entry}$ Assumes EMA technology M_{entry} – Entry mass of vehicle	-59% different technology than shuttle

Reference: 4a Derived from: Airbreathing booster. Options: None.	Space Shuttle Comparison
$M_{sca} = 640 + 0.013 M_{glow}$ Control and actuation mass for airbreathing booster vehicle. May include hydraulics, though source is not clear on this. M_{glow} – Gross liftoff mass of vehicle	51%

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{sca} = 28\frac{M_{land}}{S_{body}}$ Control system and actuation mass for a lifting body upper stage. M_{land} – Landed mass of vehicle S_{body} – Planform area of the body	-32%

Reference: 5 Derived from: Aircraft. Options: None.	Space Shuttle Comparison
$M_{sca} = 36.28 M^{0.003} S_{tot\_cont}^{0.489} N_{cs}^{0.484} N_{crew}^{0.127}$ N_{cs} – Number of flight control systems (redundancy) S_{tot_cont} – Total planform area of all control surfaces N_{crew} – Number of crew	-44%

Reference: 6 Derived from: Boeing equation adjusted to Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{sca} = 0.55(S_{ref} + S_{htail} + S_{canard}) + 3.4S_{vert} + 2.6(S_{bf} + S_{sb}) + 70$ S_{bf} – Planform area of body flap S_{canard} – canard planform area S_{htail} – horizontal tail planform area S_{ref} – Theoretical wing planform area S_{sb} – speed brake planform area S_{vert} – Total planform area of vertical tail or wingtip fins	29%

Reference: 7 Derived from: Aircraft. Options: Control surface configuration.	Space Shuttle Comparison
$$M_{sca} = K_{fcf}\left(\frac{M_{glow}}{1000}\right)^{0.581}$$ K_{fcf} = 106 – for airplanes with elevon control, and no horizontal tail = 138 – for airplanes with a horizontal tail = 168 – for airplanes with a variable sweep wing M_{glow} – Gross liftoff mass of vehicle	-1%

Reference: 10a Derived from: Alpha Technologies, rocket based. Options: None.	Space Shuttle Comparison
$M_{sca} = (0.55S_{\exp} + 30) + (3.4S_{vert} + 30) + (2.6S_{bf} + 10)$ For rocket powered vehicles wing, vert. tail, body flap S_{bf} – Planform area of body flap S_{exp} – Exposed wing planform area S_{vert} – Total planform area of vertical tail or wingtip fins	4%

13.0 Avionics

A_{body} – surface area of vehicle body
M_{dry} – Dry mass of vehicle
M_{glow} – Gross liftoff mass
N_{crew} – Number of crew
N_{days} – Number of days spent on orbit
N_{eng} – Number of main engines on stage
N_{pil} – number of pilots

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: 1978 or 1990 technology.	Space Shuttle Comparison
$M_{av} = K_{av} M_{dry}^{0.125}$ K_{av} = 1350 – for current technology (~1978) = 710 – for 1990 technology M_{dry} – Dry mass of vehicle	-7%

Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: None.	Space Shuttle Comparison
$M_{av} = 3300$ Constant based on NASP technology AMLS SSTO	-50% not shuttle technology

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{av} = 6564$ Constant based on Space Shuttle avionics mass	0%

Reference: 4a Derived from: Airbreathing booster. Options: None.	Space Shuttle Comparison
$M_{av} = 670 + 440 + 240$ Includes 670 lbs for instruments, 440 lbs for guidance and navigation, and 240 lbs for communication. For an airbreathing booster vehicle.	-79%

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
Compared to Space Shuttle avionics mass less instruments and displays. $M_{av} = 261 + 302$ Includes 261 lbs. for guidance and navigation, and 302 lbs. for communications. No instruments.	-85%

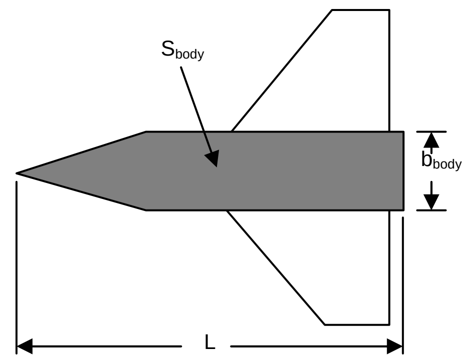
Reference: 6 Derived from: Data from EHLLV, Shuttle C studies, and Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{av} = 544 + 1067\frac{N_{days}}{7} + 3027\frac{N_{crew}}{7} + 0.27A_{body}$ Includes range safety weight. A_{body} – surface area of vehicle body N_{crew} – Number of crew N_{days} – Number of days spent on orbit	-2%

Reference: 7 Derived from: Aircraft. Options: None.	Space Shuttle Comparison
Compared to only the instruments and displays for the space shuttle. $M_{inst} = N_{pil}\left[15 + 0.032\left(\frac{M_{glow}}{1000}\right)\right] + N_{eng}\left[5 + 0.006\left(\frac{M_{glow}}{1000}\right)\right] + 0.15\left(\frac{M_{glow}}{1000}\right) + 0.012 M_{glow}$ Mass for instruments and displays only. M_{glow} – Gross liftoff mass N_{eng} – Number of main engines on stage N_{pil} – number of pilots	23% only instruments and displays

Reference: 10a Derived from: Alpha Technologies, rocket based. Options: None.	Space Shuttle Comparison
$M_{av} = 544 + 1067 * if(N_{days} > 0.1,1,0) + 3012 * if(N_{crew} > 0,1,0) + 0.27 A_{body}$ For rocket vehicle A_{body} – surface area of vehicle body N_{crew} – Number of crew N_{days} – Number of days spent on orbit	-2%

14.0 Environmental Control & Life Support System

b_{body} – Maximum width of the body
H_{body} – Height of body
L – Length of vehicle
M_{av} – Mass of avionics (group 13.0)
M_{ecd} – Mass of electronic conversion & distribution (group 10.0)
N_{crew} – Number of crew
N_{days} – Number of days spent on orbit
V_{crew} – Volume of crew cabin



Reference: 1 Derived from: Aircraft and Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{eclss} = K_{pv} V_{crew}^{0.75} + K_{os} N_{crew} N_{days} + K_{ah} M_{av}$ K_{pv} = 5.85 – pressurized volume constant K_{os} = 10.9 – oxygen supply tank constant K_{ah} = 0.44 – avionics heat load constant M_{av} – Mass of avionics (group 13.0) N_{crew} – Number of crew N_{days} – Number of days spent on orbit V_{crew} – Volume of crew cabin	5%

<table>
<tr><td>Reference: 2
Derived for: Airbreathing horizontal takeoff vehicle.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$$M_{eclss} = 141 + 729 + K_{htl} 6.79\left(L + b_{body} + H_{body}\right) + 512 + 163$$
K_{htl} = 0.6 – Shape factor for RBCC SSTO (Payload bay close to crew cabin with radiators in payload bay doors).
This equation is composed of:
141 lb – personnel systems
729 lb – equipment cooling system
512 lb – radiators
163 lb – flash evaporators
All masses are based on AMLS SSTO study by Stanley

b_{body} – Maximum width of the body
H_{body} – Height of body
L – Length of vehicle</td><td>-60%</td></tr>
</table>

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{eclss} = 2652 + 54.1 N_{crew} N_{days}$ N_{crew} – Number of crew N_{days} – Number of days spent on orbit	0%

Reference: 4a Derived from: Airbreathing booster. Options: None.	Space Shuttle Comparison
$M_{eclss} = 550$ Constant mass for environmental control. System is designed for a booster, so is for short duration and small crew to pilot the vehicle only.	N/A Short duration only

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{eclss} = 464 + 20N_{crew}$ Mass of environmental control system. **Maybe add long term facilities** N_{crew} – Number of crew	-89%

Reference: 6 Derived from: Space Shuttle. Options: None.	Space Shuttle Comparison
Shuttle comparison to the cooling system mass only. $M_{cooling} = 32.24(N_{crew}N_{days})^{1.18}$ Cooling system mass only N_{crew} – Number of crew N_{days} – Number of days spent on orbit	1%

<table>
<tr><td>Reference: 10a
Derived from: Alpha Technologies, rocket based.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>For rocket powered vehicles.
$$M_{eclss} = (0.3M_{av} + 0.15M_{ecd}) + (1410 * if(N_{crew} > 0,1,0)) + (870 * if(N_{crew} > 0,1,0)) + 15.4[N_{crew}(N_{days} + 3)]^{1.18}$$
Equipment cooling   Crew controls   Crew displays   Crew Env. Cooling

For airbreathing vehicles.
$$M_{eclss} = 1235\left\{1 + 0.27\left(\frac{L}{205} - 1\right) + 0.069\left[\left(\frac{L}{205}\right)^3 - 1\right]\right\}$$
M_{av} – Mass of avionics (group 13.0)
N_{crew} – Number of crew
N_{days} – Number of days spent on orbit
M_{ecd} – Mass of electronic conversion & distribution (group 10.0)</td><td>53%</td></tr>
</table>

15.0 Personnel Equipment

N_{crew} – Number of crew
N_{days} – Number of days spent on orbit

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Mission duration.	Space Shuttle Comparison
$M_{pe} = K_{fww} + K_{furn} N_{crew}$ K_{fww} – food, waste, and water management system: for 1 to 4 crew = 0 – for missions less than 24 hours = 353 – for missions greater than 24 hours K_{furn} – seats and other pilot/crew related items = 167 N_{crew} – Number of crew	-17%

Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: None.	Space Shuttle Comparison
$M_{pe} = 502 + 150 N_{crew}$ Based on NASP technology AMLS (Stanley) N_{crew} – Number of crew	-15%

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{pe} = 555 + 164 N_{crew}$ N_{crew} – Number of crew	-7%

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{pe} = 52 N_{crew}$ Mass of cabin furnishings. N_{crew} – Number of crew	-80%

Reference: 5 Derived from: Aircraft. Options: None.	Space Shuttle Comparison
Compared only to space shuttle personnel accommodations and furnishings and equipment. $M_{furn} = 217.6 N_{crew}$ Furnishings and seats only, no galley or water or waste management N_{crew} – Number of crew	34%

Reference: 6 Derived from: Space Shuttle. Options: None.	Space Shuttle Comparison
For space shuttle comparison the mass of the personnel group was added to the mass of the personnel equipment. $M_{pe} = 2444\frac{N_{crew}}{7} + 645N_{crew} + 86.4N_{days}$ Includes personnel in addition to personnel equipment. N_{crew} – Number of crew N_{days} – Number of days spent on orbit	38%

16.0 Dry Weight Margin

M_{dry} – Dry mass of vehicle
M_{eng} – Mass of a single main engine
M_i – Total mass of group *i* in MBS
N_{eng} – Number of main engines on stage

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{margin} = K_{margin}\left(M_{dry} - N_{eng}M_{eng}\right)$ $K_{margin} = 0.10$ M_{dry} – Dry mass of vehicle M_{eng} – Mass of a single main engine N_{eng} – Number of main engines on stage	N/A

Reference: 2 Derived for: Airbreathing horizontal takeoff vehicle. Options: None.	Space Shuttle Comparison
$$M_{margin} = K_{margin} \sum_{i=1.0}^{15.0} M_i$$ K_{margin} – Margin percentage M_i – Total mass of group i Recommends K_{margin} = 10% margin	N/A

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{margin} = K_{margin} \sum_{i=1.0}^{15.0} M_i$ K_{margin} – Margin percentage M_i – Total mass of group i Recommends K_{margin} = 15% margin	N/A

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{margin} = K_{margin} \sum_{i=1.0}^{15.0} M_i$ K_{margin} – Margin percentage M_i – Total mass of group i Recommends K_{margin} = 3% margin	N/A

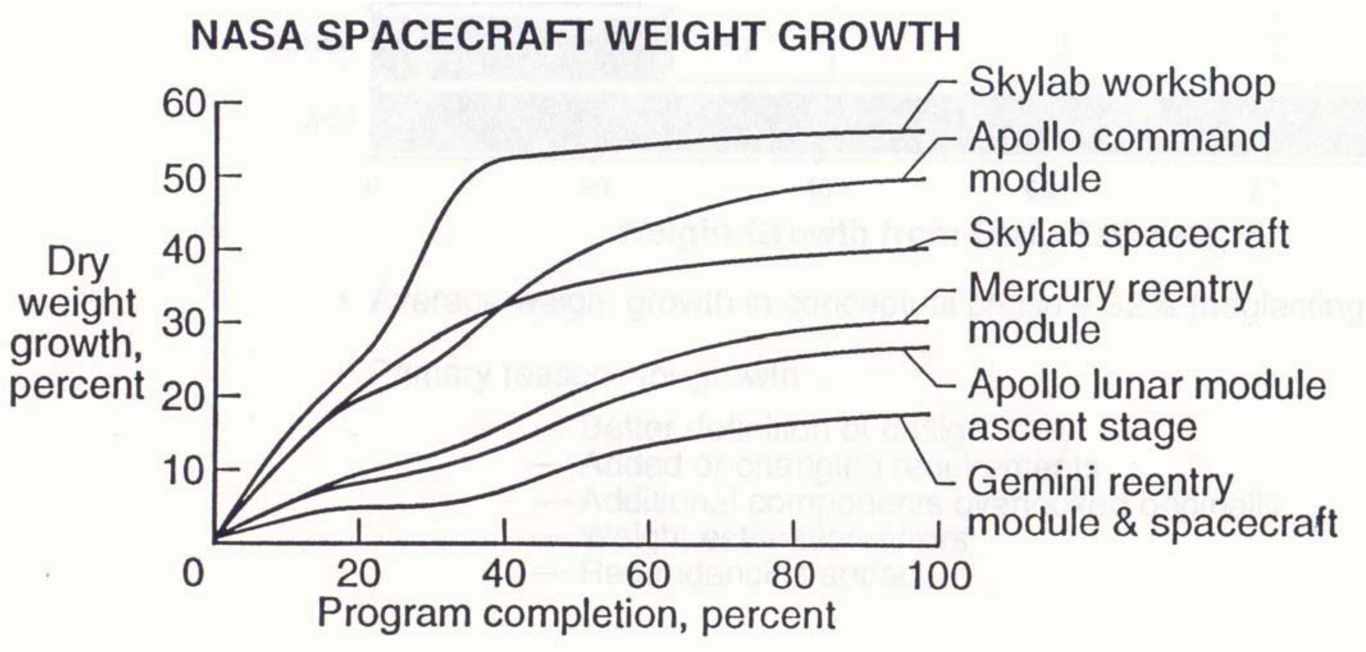
<table>
<tr><td>Reference: 6
Derived from: Data developed by Program Development PD24 (80-22).
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$$M_{margin} = K_{margin} \sum_{i=1.0}^{15.0} M_i$$
K_{margin} – Margin percentage

M_i – Total mass of group i

Recommended margin based on development status: K_{margin}=
0% - masses based on existing structures, hardware, engines, which require no modification
5% - masses based on existing structures, hardware, engines, which require some modification
10% - masses based on new designs which use existing type materials and subsystems
15% - masses based on new designs which use existing type materials and subsystems which require limited development in materials technology
20%-25% - weights based on new designs which require extensive development in materials technology</td><td>N/A</td></tr>
</table>

Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: None.	Space Shuttle Comparison
$M_{margin} = K_{margin} \sum_{i=1.0}^{15.0} M_i$ K_{margin} – Margin percentage M_i – Total mass of group i Recommends K_{margin} = 15% margin	N/A

Reference: 11 Derived from: Program data from space hardware.	Space Shuttle Comparison
$M_{margin} = K_{margin} \sum_{i=1.0}^{15.0} M_i$ K_{margin} – Margin percentage M_i – Total mass of group i Hawkins shows that historically dry weight growth from proposal to first flight is 25.5%.	N/A

Reference: 12 Derived from: NASA space programs.	Space Shuttle Comparison
Talay shows a 30% historical increase in vehicle weight from first concept documentation to time of proposal. The following chart also shows dry weight growth for NASA programs only.  Chart showing dry weight growth of NASA space vehicle programs from [ref. 12].	N/A

17.0 Crew & Gear

N_{crew} – Number of crew
N_{days} – Number of days spent on orbit

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{cg} = 400 + 560 N_{crew}$ N_{crew} is limited to between 1 and 4 people. N_{crew} – Number of crew	18%

Reference: 2 and 3 Derived for: Airbreathing horizontal takeoff vehicle, and from a rocket. Options: None.	Space Shuttle Comparison
$M_{cg} = 1176 + (311 + 23N_{days})N_{crew}$ Includes crew consumables (food), personal items, crew, and suits (Talay) N_{crew} – Number of crew N_{days} – Number of days spent on orbit	23%

Reference: 4a Derived from: Airbreathing booster. Options: None.	Space Shuttle Comparison
$M_{cg} = 650$ Constant mass, for a small crew to pilot a booster stage.	N/A

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{cg} = (220 + 35.5) N_{crew}$ N_{crew} – Number of crew	-51%

Reference: 6 Derived from: Space Shuttle. Options: None.	Space Shuttle Comparison
Equations under group 15.0 includes crew.	N/A

Reference: 10a Derived from: Alpha Technologies. Options: None.	Space Shuttle Comparison
$M_{cg} = 2550 * if\,(N_{crew} > 0,1,0) + 645 N_{crew} + 77.6 N_{days}$ N_{crew} – Number of crew N_{days} – Number of days spent on orbit	109%

18.0 Payload Provisions

M_{pl} – Mass of payload

Reference: 2 and 3 Derived for: Airbreathing horizontal takeoff vehicle, and from a rocket. Options: None.	Space Shuttle Comparison
$M_{payp} = 0.0$ Mass of provisions included in payload mass.	N/A

Reference: 6 Derived from: Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{payp} = 0.025 M_{pl}$ M_{pl} – Mass of payload	431%

19.0 Cargo (up and down)

Reference: All	Space Shuttle Comparison
Fixed value based on mission. For worst case the payload must be assumed to be returned for sizing re-entry and landing loads.	N/A

20.0 Residual Propellants

$M_{main\_usable\_prop}$ – Usable main propellant, typically $M_{pascent}$
M_{oms/rcs_usable_prop} – Usable OMS and RCS system propellants
$M_{pascent}$ – Mass of ascent propellants (group 27.0)
M_{tot_fuel} – Mass of all fuel on stage
M_{tot_ox} – Mass of all oxidizer on stage
V_f – Total fuel volume
V_{ox} – Total oxidizer volume

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: None.	Space Shuttle Comparison
$M_{resid} = 0.05 M_{pascent}^{0.79}$ Includes main propellant tank pressurization gas. $M_{pascent}$ – Mass of ascent propellants	-98% Orbiter -15% ET

Reference: 2 and 3 Derived for: Airbreathing horizontal takeoff vehicle, and from a rocket. Options: None.	Space Shuttle Comparison
Shuttle comparison only uses equation for OMS/RCS residuals. $M_{oms/rcs\_resid} = 0.05 M_{oms/rcs\_usable\_prop}$ $M_{mainprop\_resid} = 0.005 M_{main\_usable\_prop}$ $M_{main\_usable\_prop}$ – Usable main propellant, typically $M_{pascent}$ M_{oms/rcs_usable_prop} – Usable OMS and RCS system propellants $M_{pascent}$ – Mass of ascent propellants	-37% Orbiter OMS/RCS 70% ET

Reference: 4a Derived from: Airbreathing booster. Options: None.	Space Shuttle Comparison
$M_{ox\_resid} = 0.027 M_{tot\_ox}$ Residual liquid oxygen for an airbreathing booster vehicle $M_{f\_resid} = 0.027 M_{tot\_fuel}$ Residual liquid hydrogen for an airbreathing booster vehicle M_{tot_fuel} – Mass of all fuel on stage M_{tot_ox} – Mass of all oxidizer on stage	816% ET

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{ox\_resid} = 0.005 M_{tot\_ox}$ Residual liquid oxygen for a rocket second stage $M_{f\_resid} = 0.03 M_{tot\_fuel}$ Residual liquid hydrogen for a rocket second stage M_{tot_fuel} – Mass of all fuel on stage M_{tot_ox} – Mass of all oxidizer on stage	195% ET

<table>
<tr><td>Reference: 10
Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing).
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>Shuttle comparison only uses equation for OMS/RCS residuals.

$M_{oms/rcs\_resid} = 0.05 M_{oms/rcs\_usable\_prop}$

$M_{resid\_ascent\_fuel} = 4\dfrac{(0.00529 V_f)}{K_{f\_package}}$

$M_{resid\_ascent\_ox} = 2\dfrac{(0.11 V_{ox})}{K_{ox\_package}}$

$K_{f\_package}$ – Fuel tank internal packaging efficiency, takes into account baffles, spars, etc..
$K_{ox\_package}$ – Oxidizer tank internal packaging efficiency, takes into account baffles, spars, etc..

M_{oms/rcs_usable_prop} – Usable OMS and RCS system propellants
V_f – Total fuel volume
V_{ox} – Total oxidizer volume</td><td>-37%
Orbiter

18%
ET</td></tr>
</table>

21.0 OMS/RCS Reserve Propellants

g – Gravitational acceleration at the surface of the Earth
Isp_{oms} – Specific impulse of OMS engines
Isp_{rcs} – Specific impulse of RCS engines
M_{land} – Landed mass of vehicle
M_{oms/rcs_usable_prop} – Usable OMS and RCS system propellants
$M_{pascent}$ – Mass of ascent propellants
ΔV_{oms} – Total velocity change possible using OMS engines (fps)
ΔV_{rcs} – Total velocity change possible using RCS engines (fps)

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: None.	Space Shuttle Comparison
For shuttle comparison maximum ΔV was calculated from the maximum usable propellants available. For OMS ΔV = 700 fps, RCS ΔV = 200 fps. $M_{oms/rcs\_res} = M_{land}\left[e^{\left(\frac{0.005\Delta V_{oms}}{Isp_{oms}g}\right)} + e^{\left(\frac{0.005\Delta V_{rcs}}{Isp_{rcs}g}\right)} - 2\right]$ g – Gravitational acceleration at the surface of the Earth Isp_{oms} – Specific impulse of OMS engines Isp_{rcs} – Specific impulse of RCS engines M_{land} – Landed mass of vehicle ΔV_{oms} – Total velocity change possible using OMS engines (fps) ΔV_{rcs} – Total velocity change possible using RCS engines (fps)	-94% OMS/RCS residual only

<table>
<tr><td>Reference: 2 and 3
Derived for: Airbreathing horizontal takeoff vehicle, and from a rocket.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$M_{oms/rcs\_res} = 0.1 M_{oms/rcs\_usable\_prop}$

M_{oms/rcs_usable_prop} – Usable OMS and RCS system propellants</td><td>43%</td></tr>
</table>

Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: None.	Space Shuttle Comparison
$M_{oms/rcs\_res} = 0.0075 M_{pascent}$ $M_{pascent}$ – Mass of ascent propellants	-98% Using ascent propellant in shuttle only 595% Using ascent propellant in ET

22.0 RCS Entry Propellants

g – Gravitational acceleration at the surface of the Earth
Isp_{rcs} – Specific impulse of RCS engines
M_{entry} – Entry mass of vehicle
ΔV_{rcs_entry} – entry velocity change required

Reference: 1, 2, and 10 Derived from: Physics based. Options: None.	Space Shuttle Comparison
$$M_{rcs\_entry} = M_{entry}\left[e^{\left(\frac{\Delta V_{rcs\_entry}}{Isp_{rcs}g}\right)} - 1\right]$$ *g* – Gravitational acceleration at the surface of the Earth Isp_{rcs} – Specific impulse of RCS engines M_{entry} – Entry mass of vehicle ΔV_{rcs_entry} – entry velocity change required (Shuttle = 40 fps)	8%

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{rcs\_entry} = 0.00336 M_{entry}$ Assumes approximately 40 fps of ΔV. M_{entry} – Entry mass of vehicle	-16%

23.0 OMS/RCS On-Orbit Propellants

g – Gravitational acceleration at the surface of the Earth
Isp_{oms} – Specific impulse of OMS engines
Isp_{rcs} – Specific impulse of RCS engines
M_{entry} – Entry mass of vehicle
M_{land} – Landed mass of vehicle
N_{crew} – Number of crew
ΔV_{oms} – Total velocity change possible using OMS engines (fps)
ΔV_{rcs} – Total velocity change possible using RCS engines (fps)

Reference: 1 Derived from: Physics based. Options: None.	Space Shuttle Comparison
For shuttle comparison maximum ΔV was calculated from the maximum usable propellants available. For OMS ΔV = 700 fps, RCS ΔV = 200 fps. $M_{oms/rcs\_orbit} = M_{entry}\left[e^{\left(\frac{\Delta V_{oms}}{Isp_{oms}g}\right)} + e^{\left(\frac{\Delta V_{rcs}}{Isp_{rcs}g}\right)} - 2\right]$ Isp_{oms} – OMS propulsion specific impulse = 313s – storable = 440s – cryogenic Isp_{rcs} – RCS propulsion specific impulse = 289s – storable pulsing system = 398s – cryogenic pulsing system g – Gravitational acceleration at the surface of the Earth M_{entry} – Entry mass of vehicle ΔV_{oms} – Total velocity change possible using OMS engines (fps) ΔV_{rcs} – Total velocity change possible using RCS engines (fps)	-4% Using actual shuttle Isp and ΔV

Reference: 2, 3, and 10 Derived for: Physics based. Options: None.	Space Shuttle Comparison
For shuttle comparison maximum ΔV was calculated from the maximum usable propellants available. For OMS ΔV = 700 fps, RCS ΔV = 200 fps. $M_{oms\_orbit} = M_{entry}\left[e^{\left(\frac{\Delta V_{oms}}{Isp_{oms}g}\right)} - 1\right]$ $M_{rcs\_orbit} = M_{entry}\left[e^{\left(\frac{\Delta V_{rcs}}{Isp_{rcs}g}\right)} - 1\right]$ ΔV_{rcs} – Typically 15 fps for front RCS, and 35 fps for aft RCS ΔV_{oms} – Typically 500-800 fps for ascent, 50 fps for on orbit maneuvers, and 200 fps de-orbit Isp_{rcs} = 420s – LOX/LH2 pressure fed thrusters based on Rockwell IHOT work, O/F=4.0. Isp_{oms} = 462s – LOX/LH2 pump fed engines based on Rockwell IHOT work, O/F=6.0. g – Gravitational acceleration at the surface of the Earth Isp_{oms} – Specific impulse of OMS engines Isp_{rcs} – Specific impulse of RCS engines M_{entry} – Entry mass of vehicle ΔV_{oms} – Total velocity change possible using OMS engines (fps) ΔV_{rcs} – Total velocity change possible using RCS engines (fps)	-4% Using actual shuttle Isp and ΔV

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{rcs\_prop} = 0.0215 M_{land}$ All RCS propellant, including entry, for a rocket powered lifting body second stage. $M_{apu\_prop} = 543 + 30 N_{crew}$ Propellant required for APU while on orbit. M_{land} – Landed mass of vehicle N_{crew} – Number of crew	-76%

24.0 Cargo Discharged

Reference: All	Space Shuttle Comparison
Constant value dependent on the mission. Mass of payload carried to orbit, and not back to Earth.	N/A

25.0 Ascent Reserve Propellants

g – Gravitational acceleration at the surface of the Earth
Isp_{vac} – Vacuum specific impulse of main engines
$M_{pascent}$ – Mass of ascent propellants
ΔV_{ideal} – Ideal ΔV for ascent

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: None.	Space Shuttle Comparison
Space Shuttle comparison uses ascent propellant in the ET. ΔV is based on Orbiter and ET together (no SRBs). $$M_{pascent\_res} = M_{insert}\left[e^{\left(\frac{\Delta V_{ideal}\,0.005}{Isp_{vac}\,g}\right)} - 1\right] + 0.004 M_{pascent}$$ ΔV_{ideal} – Ideal ΔV for ascent = 24,994 fps calculated from maximum usable propellant load on Space Shuttle and ET combination. g – Gravitational acceleration at the surface of the Earth Isp_{vac} – Vacuum specific impulse of main engines $M_{pascent}$ – Mass of ascent propellants ΔV_{ideal} – Ideal ΔV for ascent	64%

Reference: 2, 3, and 4b Derived for: Airbreathing horizontal takeoff vehicle, and from a rocket, and lifting body upper stage. Options: None.	Space Shuttle Comparison
Space Shuttle comparison uses ascent propellant in the ET. $M_{pascent\_res} = 0.005 M_{pascent}$ Main propellant reserves are vented to orbit or transferred off-board before entry. $M_{pascent}$ – Mass of ascent propellants	50%

Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: None.	Space Shuttle Comparison
Space Shuttle comparison uses ascent propellant in the ET. $M_{pascent\_res} = 0.0075 M_{pascent}$ $M_{pascent}$ – Mass of ascent propellants	125%

26.0 Inflight Losses & Vents

$M_{pascent}$ – Mass of ascent propellants
M_{entry} – Entry mass of vehicle

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: None.	Space Shuttle Comparison
Shuttle comparison uses ascent propellant in the ET, and compares to losses in the Orbiter. $M_{plosses} = 0.0043 M_{pascent}$ $M_{pascent}$ – Mass of ascent propellants	83%

Reference: 2, 3, and 10 Derived for: Airbreathing horizontal takeoff vehicle, and from a rocket. Options: None.	Space Shuttle Comparison
$M_{plosses} = 0.01 M_{entry}$ Includes waste, purge gasses, excess fuel cell reactants, vented and lost propellants. Note: Reference 10 includes this mass after the insertion weight, meaning that it is treated as propellant lost during ascent. M_{entry} – Entry mass of vehicle	-35%

27.0 Ascent Propellants

M_{f_ascent} – Total ascent fuel
M_{glow} – Gross liftoff mass
MR – Mass ratio for ascent
R_f – Fuel ascent propellant fraction: M_{f_ascent} over M_{p_ascent}

Reference: 2 Derived for: Physics based. Options: None.	Space Shuttle Comparison
$$M_{f\_ascent} = R_f M_{glow}\left(1 - \frac{1}{MR}\right)$$ Ascent fuel mass $$M_{ox\_ascent} = M_{f\_ascent}\left(\frac{1}{R_f} - 1\right)$$ Ascent oxidizer mass M_{f_ascent} – Total ascent fuel M_{glow} – Gross liftoff mass MR – Mass ratio for ascent R_f – Fuel ascent propellant fraction: M_{f_ascent} over M_{p_ascent}	2

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
Use the same equations as reference 2, but add 60 lbs. of total propellant lost during thrust decay.	4b

Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: None.	Space Shuttle Comparison
$M_{pascent} = \frac{MR-1}{MR} M_{glow}$ M_{glow} – Gross liftoff mass MR – Mass ratio for ascent	10

28.0 Startup Losses

A_{body} – surface area of vehicle body
Isp_{sl} – Specific impulse of engine at sea level
M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff
$M_{pascent}$ – Mass of ascent propellants
M_{prop_tot} – Total propellant onboard
R_{v_lo} – Vehicle thrust to weight at liftoff
T_{start} – Main engine startup time

Reference: 1 Derived from: Aircraft and Space Shuttle. Options: Startup loss factor.	Space Shuttle Comparison
$M_{start\_loss} = M_{prop\_tot} K_{su}$ K_{su} – startup losses = 0.001 to 0.002 M_{prop_tot} – Total propellant onboard	-22% using K_{su}=0.002

<table>
<tr><td>Reference: 2
Derived for: Airbreathing horizontal takeoff vehicle.
Options: None.</td><td>Space Shuttle Comparison</td></tr>
<tr><td>$$M_{start\_loss} = 2M_{gross}\frac{R_{v\_lo}}{Isp_{sl}}$$
Assumes a 4 second ramp up of engines before hold down is released.

Isp_{sl} – Specific impulse of engine at sea level
M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff
R_{v_lo} – Vehicle thrust to weight at liftoff</td><td>44%</td></tr>
</table>

Reference: 3 Derived from: Dr. Talay, LaRC, rocket based. Options: None.	Space Shuttle Comparison
$M_{start\_loss} = 0.01 M_{pascent}$ $M_{pascent}$ – Mass of ascent propellants	291%

Reference: 4b Derived from: Lifting body rocket upper stage. Options: None.	Space Shuttle Comparison
$M_{start\_loss} = 0.00128 M_{pascent}$ Startup losses. $M_{buildup\_loss} = 210 + 30$ Propellant lost during thrust buildup (210 lbs. LOX and 30 lbs. LH2) $M_{pascent}$ – Mass of ascent propellants	-44% Using ET ascent propellant

Reference: 10 Derived from: NASA MSFC 3rd generation launch vehicle office (mostly airbreathing). Options: None.	Space Shuttle Comparison
$M_{start\_loss} = T_{start} M_{gross} \dfrac{R_{v\_lo}}{Isp_{sl}}$ Losses while starting the engines $M_{boiloff} = K_{boil} \dfrac{A_{body}}{21357}$ Boiloff while waiting on the pad or runway K_{boil} = 4359 – for LH2 = 104 – for LOX A_{body} – surface area of vehicle body Isp_{sl} – Specific impulse of engine at sea level M_{gross} – Gross vehicle mass on the pad or runway prior to liftoff R_{v_lo} – Vehicle thrust to weight at liftoff T_{start} – Main engine startup time	351% Based on 6s from ignition to release 1% at 1.4s

Technology Reduction Factors

Reference: 3
These *TRFs* represent near term improvements. For example AMLS or NASP.

Near term mass reduction by system. The technology reduction factor (*TRF*) is listed on the right. The new mass (improved technology) is found using the following equation:

$$M_{new} = M_{original}(1\text{-}TRF)$$

1.0 Wing	44%
2.0 Tail	44%
3.0 Body & secondary struct.	38%
Crew cabin	38%
Body flap	44%
Thrust structure	38%
LOX & LH2 tank	0%
4.0 TPS	35%
5.0 Landing gear	9%
6.0 Main Propulsion	15%
7.0 RCS	0%
8.0 OMS	0%
9.0 Primary Power	0%
10.0 ECD	18%
11.0 Hydraulics	0%
12.0 Surface Control (EMA)	0%
13.0 Avionics	50%
14.0 ECLSS	10%
15.0 Personnel Equipment	0%

Reference: 6
Derived from data provided by Airframe Team, September 1999.

Technology mass reduction factors by material. The *TRF* is listed on the left. The new mass (improved technology) is found using the following equation:

$$M_{new} = M_{original}(1-TRF)$$

0% - Structural designs based on current aluminum alloy, ie. *Saturn V*, original ET
10% - Structural designs based on aluminum lithium alloy, ie new lightweight ET
20% - Wing structural designs based on advanced composites and materials
25% - Propellant tanks structural designs based on advanced composites and materials
30% - Interstages and body structural designs based on advanced composites and materials