TECHNIQUES FOR THE DETERMINATION OF MASS PROPERTIES OF EARTH-TO-ORBIT TRANSPORTATION SYSTEMS

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THE UNITS FOR THE PHYSICAL QUANTITIES DEFINED IN THIS PAPER ARE GIVEN BOTH IN THE INTERNATIONAL SYSTEMS OF UNITS (SI) AND IN THE U.S. CUSTOMARY UNITS. TABLES II AND III ARE IN U.S. CUSTOMARY UNITS. COMPUTER RESULTS HAVE BEEN PROGRAMMED FOR THE PRINTOUT IN BOTH UNITS. THE MEASUREMENTS AND CALCULATIONS WERE MADE IN U.S. CUSTOMARY UNITS.
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SUMMARY

As a possible follow-on Earth-to-orbit transportation system, single- and two-stage winged vehicles are being studied. All propellants are carried internally and the vehicle returns to base for an aircraft-type landing. Such studies require a reasonably accurate means of rapidly determining the mass properties of the overall system when various vehicle design parameters are varied. Two techniques have been developed; one involves the trending of the overall vehicle to a new size when mass properties are already known from a prior detailed analysis; the other technique involves trending each subsystem from known space shuttle, aircraft, and applied research hardware to determine overall mass properties by summation of the trended subsystems of vehicles for which little is known initially.

Several fairly extensive documents for mass estimating have been published for two-stage fully reusable and single-stage-to-orbit systems (refs. 1 through 6). The intent of the present work is to extend present capabilities to emerging new classes of vehicles. This has been achieved by modifying these equations which were originally intended for commercial or fighter aircraft for the Earth-to-orbit vehicles--vehicles which are markedly different in structural concept and material usage.
I. Vehicle Geometry

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II. Subsystem Masses (all units in this section are in Kg or Lbm)

\[ m \] = total propellant mass in vehicle
\[ m_a \] = avionics mass
\[ m_{ARES} \] = ascent residual and reserve mass
\[ m_b \] = body mass including tanks
\[ m_d \] = vehicle dry mass less margin
\[ m_{DES} \] = mass of vehicle at descent (i.e., mass of vehicle after execution of dec. bit maneuver)
\[ m_e \] = vehicle mass at "entry" i.e., at 122 Km (400,000 ft) altitude after depletion of entry attitude control propellants
\[ m_{ENG} \] = mass per main engine
\[ m_f \] = estimated fixed masses (total) in the vehicle including payload, cargo bay doors and structure, manipulator, avionics, crew compartment and crew
\[ m_c \] = on-orbit and deorbit attitude control and maneuver propellant
\[ m_g \] = landing gear, manipulator, and docking system mass
\[ m_{gr} \] = gross mass
\[ m_{INJ} \] = vehicle mass at injection
\[ m_I \] = inert mass of vehicle, or gross less fixed masses and ascent propellant (subscripts 1 and 2, point design and off-point design vehicles)
\[ m_L \] = landed mass
\[ m_{ma} \] = manipulator mass
\[ m_s \] = docking and separation system mass
\[ m_{pt} \] = total propellant mass
\[ m_p \] = ascent propellant mass
\[ m_{p1} \] = ascent propellant mass in point design vehicle
\[ m_{p2} \] = ascent propellant mass in off-point design vehicle
\( m_w \) = wing mass
\( m_{sc} \) = surface control mass
\( m_{env} \) = environmental control system mass
\( m_{mar} \) = growth mass
\( m_{PS} \) = thermal protection system mass
\( m_{Wu} \) = unit wing structural mass
\( m_c \) = vertical tail mass
\( m_d \) = OMS maneuver system mass
\( m_r \) = all up reaction control system dry mass, including engines and tanks
\( m_{el} \) = electrical subsystem mass
\( m_s \) = separation and docking system mass
\( m_{pf} \) = maneuver engine pressurization and feed
\( m_R \) = reusable surface insulation average unit mass over the entire vehicle
\( m_{0P} \) = OMS propellant mass
\( m_{1P} \) = unit masses of point design and off-point design wing respectively based on exposed planform
\( m_{av} \) = avionics mass
\( m_{uf} \) = residual and unusable fluids
\( m_m \) = maneuver engine mass
\( m_{DOW} \) = prime power mass
\( m_{ROMS} \) = mass of GMS reserves
\( m_{RRC} \) = mass RCS reserves
\( m_{orr} \) = OMS + RCS reserves
\( m_{RCSe} \) = entry RCS propellant
\( m_{INF} \) = inflight losses
\( m_h \) = hydraulics system mass
II. (CONT'D)

GLOW = gross liftoff mass

III. Mission

$\Delta V_{\text{IDEAL}}$ = delta "Vee" equivalent for total mission, m/sec (ft/sec)

$\Delta V_{\text{ro}}$ = delta "Vee" equivalent for attitude control on-orbit, m/sec (ft/sec)

$\Delta V_{\text{o}}$ = delta "Vee" equivalent for maneuver system, m/sec (ft/sec)

$\Delta V_{\text{re}}$ = delta "Vee" equivalent for attitude control entry, m/sec (ft/sec)

IV. Propulsion Performance

$m$ = mass flow per engine, Kg/sec (lb/sec)

$T_{\text{VAC}}$ = vacuum thrust per main engine, Newtons (lbf)

$I_{\text{s-re}}$ = average reaction control system specific impulse during reentry, sec

$I_{\text{s-ro}}$ = specific impulse of reaction control engines on-orbit, sec

$I_{\text{s-m}}$ = specific impulse of the OMS maneuver engine degraded for the estimated number of restarts, sec

$I_{\text{s-e}}$ = main engine vacuum specific impulse, sec

$\lambda_1$ = performance mass fraction or ascent propellant divided by gross mass for the point design vehicle (dimensionless)

$\lambda_2$ = the performance mass fraction of the off-point design (dimensionless)

$\lambda_1'$ = point design trending propellant mass fraction (or ascent propellant divided by gross mass less fixed mass) dimensionless

M.R. = mass ratio equals gross lift-off mass divided by burnout mass (dimensionless)
V. Subsystem Constants: Paragraph numbers below correspond to items in mass properties tabulations, Tables II and III, and the suggested format of the Mil Spec (ref. 12)

1.0/2.0 Wing and Tail

\[ W_m \] = Wing material/configuration constant
\[ W_c \] = Wing carry-through material/configuration constant
\[ V_t \] = Material/configuration tail constant

3.0 Body

\[ B_c \] = Crew cabin constant
\[ B_b \] = Body structure constant
\[ B_{bf} \] = Body flap
\[ B_f \] = Fuel tank constant including insulation and non opts
\[ B_o \] = Oxidizer tank constant including insulation and tank non opts

4.0 Thermal Protection System

\[ K_T \] = A constant for average TPS mass, based on wetted area

5.0 Landing Gear

\[ K_L \] = Function of landed mass

6.0 Main Rocket Engine

\[ R_{ph} \] = Engine power head
\[ R_n \] = Nozzle
\[ R_{ne} \] = Nozzle extension
\[ R_{na} \] = Nozzle extension actuator
\[ R_{pf} \] = Pressurization and feed system
Subsystems Constants (Cont'd)

\[ R_{ga} \] = Engine gimbals actuator
\[ c \] = Expansion ratio

7.0 Reaction Control System Constants

\[ R_{RCS} \] = Overall system constant

8.0 Maneuver Engine Constants

\[ M_{me} \] = A point-design constant for the maneuver engine (includes nozzles, actuators, etc.)
\[ M_t \] = Tank system constant for maneuver propellants
\[ M_{pf} \] = Pressurization and feed system constant for maneuver system

9.0 Auxiliary Power Constants

\[ A_{ap} \] = Auxiliary power unit constant
\[ A_{ac} \] = Actuator system constant

Power Subsystem Constant

\[ PW_D \] = Battery power demand constant
\[ PW_e \] = Engine power demand constant
\[ PW_C \] = Surface control power demand constant

10.0 Electrical Conversion and Distribution

\[ E \] = A constant for generators and wiring (does not include generator drive)

11.0 Hydraulic Conversion

\[ H_{cs} \] = Control surface power constant
\[ H_e \] = Engine gimbals system power constant

12.0 Control Surface Constants

\[ S_{sc} \] = Aero surface control constant
\[ S_{ps} \] = Pilot related controls
13.0 Avionics

\[ M_{av} = \text{avionics mass constant} \]

14.0 Environmental Control

\[ E_c = \text{Pressurized volume constant} \]
\[ E_o = \text{Oxygen supply constant} \]
\[ E_a = \text{Avionics heat load constant} \]

15.0 Personnel Provisions

\[ P_{pf} = \text{Food, waste, and water management systems} \]
\[ P_{ps} = \text{Seats and other pilot and crew related items} \]

16.0 Margin

\[ \text{MAR} = \text{a percentage of dry mass less engine mass to allow for growth uncertainty} \]

17.0 Personnel (i.e., crew and mission specialists)

\[ P_p = \text{mass of individual personnel including personal gear, life support, and crew accessories} \]

20.0/27.0 Fluids

\[ R_{rf} = \text{Residual fluids} \]
\[ R_o = \text{OMS reserves constant} \]
\[ R_{pL} = \text{pre-launch losses and engine thrust build-up} \]
\[ R_r = \text{RCS reserves constant} \]
\[ R_{ar} = \text{ascent reserves constant} \]
\[ R_{ap} = \text{ascent propellant residuals constant} \]
\[ R_{inf} = \text{inflight losses constant} \]
VI. Miscellaneous

\[ K_{tk} = \text{tank area constant} \]

\[ r = \text{tank radius (m, FT)} \]

\[ V_2 = \text{volume of LH}_2\text{ tanks, or fuel tank (m}^3, \text{FT}^3) \]

\[ V_1 = \text{volume of LOX tanks, or oxidizer tank (m}^3, \text{FT}^3) \]

\[ \lambda_p = \text{packaging efficiency or body propellant volume/total body volume (dimensionless)} \]

\[ F = \text{ratio of dimensions of photographically enlarged vehicle to point design vehicle (dimensionless)} \]

\[ g = \text{gravity constant, m/sec}^2 (\text{FT/sec}^2) \]

\[ C_L = \text{average lift coefficient during entry, dimensionless} \]

\[ D = \text{days in orbit} \]

\[ e = 2.718 \text{ (constant)} \]

\[ e_g = \text{radius of gyration proportionality constant} \]

\[ f = \text{body wing efficiency factor and is the ratio of lift intensity on body to lift intensity on wing (f = 0.2 for conventional vehicle to 0.15 for control configured vehicle)} \]

\[ P_c = \text{main engine chamber pressure, N/cm}^2 (\text{Lb/in}^2) \]

\[ P_u = \text{tank ullage pressure, N/m}^2 (\text{Lb/in}^2) \]

\[ N_z = \text{load factor equals safety factor X ultimate load factor}. \]

\[ N_c = \text{number of crew, mission specialists, and passengers} \]

\[ N_e = \text{number of engines} \]

\[ \rho = \text{tank wall density (Kg/m}^3, \text{Lb/in}^3) \]

\[ \sigma = \text{tank wall limit stress (N/cm}^2, \text{Lb/in}^2) \]

\[ R_{PL} = \text{pre-launch losses} \]
I. INTRODUCTION

Two techniques for determining the mass properties of space transportation systems are presented in this report; namely, overall vehicle trending which requires little detailed information about a given vehicle and subsystem trending which requires more detailed subsystem analyses. A single-stage-to-orbit rocket powered vehicle (designated EII 155) is used extensively as an example in discussing both techniques. The vehicle is an in-house design. It is launched vertically but is provided with wings for a horizontal landing (fig. 1).

For each of the two techniques, sample computer results have been included (Appendices A and B). For the overall vehicle trending case, a sample problem and the computer tabulated results have been included (Appendix A). The method is useful for rapidly obtaining projections of vehicle performance as a rocket stage when vehicle size is altered. The savings in computer input time is substantial since only about six inputs are required for this method which is based on knowledge of the way in which subsystem masses vary as a function of vehicle size (Table I).

When a more detailed analysis is required, each subsystem is trended and the summation of all the subsystem masses gives the overall vehicle mass. In this regard, the shuttle orbiter subsystem characteristics have been utilized extensively to establish constants for "current technology" subsystems. For this reason, a sample test case of this vehicle has been included which shows the actual current masses of the various orbiter subsystems configured with the Systems Engineering
Mass Properties Program (SIMP) (table II). For various reasons, it is not intended that the program results check "exactly" with the shuttle orbiter since these latter masses vary with subsystem maturity, and in some cases added technology leverage, applied as the shuttle program progresses.

For both the shuttle orbiter and single-stage-to-orbit vehicle (EL-155), a test case is shown in Appendix D while sample computer inputs for this program have been included in Appendix C. Utilizing the shuttle subsystems as a starting point, other program input constants have been established (table III). The constants (other than shuttle) usually represent utilization of some type of advanced material, method of construction, or other developmental subsystem.
II. OVERALL VEHICLE TRENDING - DETAILED DESCRIPTION OF METHODOLOGY

When a point design vehicle mass is established, a trending technique is useful for determining masses for other vehicles without analyzing the new vehicle subsystem-by-subsystem. Only six computer inputs are required to determine overall mass properties of the vehicle. This technique is useful when two-stage systems are involved and the relative sizes of the orbiter and booster are being varied in order to optimize the system. Likewise the technique is useful when resizing single-stage systems to meet varying mission and design requirements.

The trending technique is based on the knowledge that for moderate changes in vehicle size most of the subsystem masses vary in accordance with some exponential related to vehicle dimension. For instance, the mass of the thermal protection system is a linear function of vehicle wetted area assuming the entry profile and entry planform loading are not too different. In terms of vehicle length, this subsystem varies as $L_r^2$ (where $L_r$ is vehicle reference length).

Main propulsion system tank mass, if nonintegral, is much more sensitive to vehicle physical size and varies as $L_r^3$ or directly as propellant mass since tank mass is approximately equal to a constant times propellant volume for any given shape. (This can be proven from the basic relationships for tank volume, wetted areas, and hoop stresses.) The assumption is inaccurate to the extent that insulation weight is proportional to tank wetted area or dimension squared (not cubed) and tank nonoptimums decrease slightly as size increases. Further, if the tank is integral and carrying body loads, most of the tank wall will be designed by compressive limit crippling loads and not pressure.
When the vehicle's rocket engine system is similarly assessed, these masses vary as $L_r^3$. For example, doubling the vehicle's length yields eight times the propellant tank capacity or very nearly eight times the lift-off mass. If the same lift-off thrust-to-mass ratio is to be maintained to maintain similar performance, then eight times the engine thrust (hence mass) is required; engine mass being approximated by a constant times thrust. This assumption is inaccurate to the extent that dry mass is a slightly decreasing percentage of gross mass.

Similar logic is applied to other subsystems (table I) identifying the exponential most applicable to the given subsystem. Assuming all the dry masses vary as $L_r^2$ (reference length squared) and propellant mass as $L_r^3$ (volume) then the relationship between vehicle mass and propellant loading for point designs 1 and 2 can be shown to be:

For the propellant masses:

$$\frac{m_{p2}}{m_{p1}} = \left[\frac{L_{r2}}{L_{r1}}\right]^3$$  \hspace{1cm} (1)

and for the inert (or "dry" mass):

$$\frac{m_{I2}}{m_{I1}} = \left[\frac{L_{r2}}{L_{r1}}\right]^2$$  \hspace{1cm} (2)

or

$$m_{I2} = \left[\frac{m_{p2}}{m_{p1}}\right]^{2/3} m_{I1}$$  \hspace{1cm} (3)
For simplicity of analysis inert and dry mass are used here interchangeably; $m_I$ includes all subsystem dry masses and a small percentage of residual fluids, both of which are proportional to vehicle size.

As stated earlier, external thermal protection system mass and other major subsystems vary as $L_r^2$; however, the main rocket engines and tanks vary as $L_r^3$. If all subsystems varied as $L^2$ an exponential of two-thirds in equation (3) would be valid; but if all subsystems varied as $L^3$ then the exponential of unity would be exact. It has been found from actual tests of the equations that an intermediate value of five-sixths gives general agreement with actual detailed vehicle designs (see subsequent discussions in this section for sensitivities and vehicle comparisons). Equation (3) then becomes:

$$m_{I2} = \left[ \frac{m_{P2}}{m_{P1}} \right]^{5/6} m_I$$

Another aspect of trending involves provisions in the resulting equation for subsystem masses which do not change appreciably with vehicle size. For instance, for a given mission, crew complement would remain constant; typically, avionics and environmental control remain fairly constant. For a constant volume cargo bay, doors and structure remain fairly constant. This leads to the necessity for provisions in the trending equation for the elements hereinafter referred to as "fixed" mass.
When separating the fixed mass the equation for the performance mass fraction can be written as propellant loading divided by gross mass which consists of main propellant, fixed masses, and inerts, or:

\[
\lambda_2 = \frac{m_p}{m_p + m_f + m_{i2}}
\]  

(5)

Now the concept of trendin mass fraction is introduced, which equals the main ascent propellant loading divided by the inerts plus propellant. In this definition fixed masses are again excluded, or:

\[
\lambda_1' = \frac{m_p}{m_{i1} + m_p}
\]  

(6)

or solving for \(m_{i1}\):

\[
m_{i1} = \left[1 - \frac{\lambda_1'}{\lambda_1'}\right] m_p
\]  

(7)

Substituting the value for \(m_{i1}\) in equation (4) it becomes:

\[
m_{i2} = \left[\frac{m_p}{m_{i1}}\right]^{5/6} \times \left[\frac{1 - \lambda_1'}{\lambda_1'}\right] m_p
\]  

(8)

15
substituting this value in turn for \( m_{p2} \) in equation (5) and dividing numerator and denominator by \( m_{p2} \) it becomes:

\[
\lambda_2 = \frac{1}{1 + \frac{n_f}{m_{p2}} + \left[ \frac{m_{p1}}{m_{p2}} \right]^{1/6} \times \frac{1 - \lambda_1}{\lambda_1}}
\]

or:

\[
K_{V1} = \left( \frac{m_{p1}}{m_{p2}} \right)^{1/6} \times \frac{1 - \lambda_1}{\lambda_1}
\]

which is henceforth defined as the point design technology/configuration constant.

Equation (9) for the off-point design then becomes:

\[
\lambda_2 = \frac{1}{1 + \frac{n_f}{m_{p2}} + \left[ \frac{1}{m_{p2}} \right]^{1/6} K_{V1}}
\]

In equation (10) above, as propellant mass, \( m_{p2} \), of the new point design increases so does the performance mass fraction, \( \lambda_2 \). In effect,
the "fixed" masses are becoming a smaller and smaller percentage of propellant loading, \( m_p_2 \). "Growing" the vehicle then is one means of improving design performance. The one-sixth exponential in equation (10) is the result of the assumption of five-sixths in equation (4). The sensitivity to two-thirds power rule and cubed rule can be seen in upper and lower curves respectively in figure 2 wherein the final equation yields exponentials of one-third and zero. The difference in estimated propellant loading between one-third and one-sixth exponential is approximately 1 percent for the due-east mission (see dotted horizontal line in figure 2).

To obtain the values for mass ratio in figure 2 the definitions for performance mass fraction and gross mass for the off-point design are applied, namely:

\[
M_{gr} = \frac{m_p_2}{\lambda_2} \tag{11}
\]

and

\[
M.R.2 = \frac{M_{gr}}{M_{gr} - \lambda_2 M_{gr}} = \frac{1}{1 - \lambda_2} \tag{12}
\]

For a booster, fixed masses are small since no payload is carried within the vehicle, and due to its physical size, crew and avionics would normally be small compared to vehicle mass. With little error, when applied to a booster, equation (9) can be reduced to:
In the equation above, $\lambda_1$ has become $\lambda_1$, performance mass fraction, since no fixed masses are assumed.

Reducing the point design technology/configuration constant, $K\nu_1$, in equation (10) or (13) is another alternative to improving performance as opposed to "growing" the vehicle which was mentioned previously. This constant is strongly influenced by the extent to which emerging technologies are applied to body and wing structure. It is also influenced by configuration, particularly cargo bay shape and size, and the ingenuity of the designer in packaging the vehicle for the least amount of required structure.

In the design process, a required propellant loading is estimated. When the structural analysis has been completed, or the subsystem elements weighed by simpler means, such as subsystem trending, it is found that the vehicle either has inadequate or excessive performance. Equation (10) is then useful to simply trend the vehicle to the required propellant loading from the point design with much reduced input time.

Two plots are shown for equation (10) in figure 3. The point design A is shown having a performance mass fraction $\lambda_1$ of .876 or a mass ratio (M.R.) of 8.08 for a payload requirement of 29,500 Kg. If detailed analysis should show that the inert mass of the vehicle has increased, then a new point design "B" of the same propellant loading...
results. This new generic design trends along a line of lower mass ratios for any given propellant loading. In order to restore the vehicle to the same performance line, the vehicle must be trended (i.e., "GROW") to point C. Any other changes in basic configuration, materials technology, or other would be considered as a change in point design and would trend along a new line. In figure 4, point design B is trended for various payloads ranging from 0 to 45 Mg, all with a 15 ft X 60 ft cargo bay. These plots are obtained by merely changing the value of the "fixed" masses in equation (10) by the amount of change in cargo mass. The constant performance line is shown for this dual fuel SSTO for a 50 X 100 n.m. orbit. This vehicle, if trended from a vehicle carrying 22.5 Mg to one carrying a payload of 45.4 Mg (100 Klb) would have a GLO of approximately 2.44 Gg (5.4 Mlb) (figure 4, Pt. D).

In figure 5, these values are replotted for a constant orbit to show ascent propellant mass and vehicle gross mass as a function of payload carried. The growth factor from the slope of the curve is fairly constant at 24.5 to 1, for the mass range shown but closer inspection shows that payload is growing at a slightly faster rate than vehicle gross mass (and propellant loading).

In figure 6, sensitivities to changes in estimated fixed masses are shown. The most identifiable fixed masses, as stated earlier for these vehicles, are payload, payload bay and doors, crew compartment and avionics, totaling an estimated 45 Mg (100,000 lb).

However, from a plot in figure 6, it can be seen from the envelope of point designs as indicated by the ellipse for phase A/B orbiters that a fixed mass of 91 Mg (200 Klb) and an exponential of 1/C gives
the best agreement for this type of vehicle. Winged orbiters from these two-stage systems are simply smaller single-stage-to-orbit (SSTO) vehicles, in that they have very similar propulsion, a cargo bay, a crew compartment, and internal LOX/LH$_2$.

Phase A/B boosters are shown on the trending curve, figure 6. They are essentially lower technology SSTO's with no cargo space and lighter thermal protection but with a turbojet cruise system not required by SSTO's, the SSTO vehicles having sufficient crossrange to "glide" back to base after return from orbit. It would not be unusual for a cruise-back system on a booster of a two-stage system to weigh well over 50 Mg (ref. 9 and 10) when the jet fuel is included.

Many factors dictate the sizing of a two-stage vehicle such as the design philosophy for the booster, whether heat sink and low staging velocity, or thermally protected and high staging velocity. But, it can be seen from figure 6 that a number of the phase A/B designs optimized at a point on the trending curves where increasing vehicle size further would yield diminishing rates of increase in performance (an expected solution).

From further inspection of figure 6, it can be seen that single-stage vehicles are located on the low slope portion of the curve where relatively large changes in vehicle size are required for small performance gains.

III. VEHICLE MASS ESTIMATING BY SUBSYSTEM TRENDING

Trending subsystem-by-subsystem is the technique used when a point design mass is to be established or when parametric-type analyses are to
be made to study the effects of changes in one subsystem on the overall system. A sample computer case for this method has been included for the in-house, single-stage-to-orbit vehicle. For computer printout, see Appendix B. The method is described in detail in the following paragraphs. These paragraphs are numbered to correspond to the subsystem designations used in the widely accepted mass properties reporting system of reference 11. Groupings such as (a), (b), (c), etc., have been added for clarity.

(a) STRUCTURE AND THERMAL PROTECTION GROUP

1.0 WING

In reference 2, wing mass equations take into account landed mass, load factor, span, theoretical wing area, and root thickness. A revised equation is used which incorporates exposed wing and exposed span as opposed to total span and theoretical wing. Also, the wing carry-through is treated as a separate term. In addition, a wing efficiency factor is used to better reflect redistribution of total lift between body and wing as the relative size of wing and body change. The resulting wing equation is:

\[
m_W = \left[ N_Z \cdot m_L \left( \frac{1}{1 + f \frac{S_b}{S_W}} \right) \right]^{n_1} \left[ \frac{S_W}{L} \right]^{n_2} \left[ m_n \left( \frac{L_W}{L_b} \right)^{n_2} + m_c \left( \frac{L_b}{L_b} \right)^{n_2} \right]
\]
In the equation above, \(m_L\) is landed mass, \(W_m\) and \(W_c\) are material configuration constants for wing and wing carry-through, respectively. Other symbols are defined in the symbols list. Earlier equations were structured from historical data for aircraft with relatively narrow bodies and distinct wing carry-throughs and typically were of the following form:

\[
m_w = \left( N_z \cdot m_L \right)^{n_1} \left[ \frac{S_{\text{THEO}} \times \text{TOTAL SPAN}}{T_{\text{Root}}} \right]^{n_2}
\]

(15)

where: \(S_{\text{THEO}}\) = theoretical wing area

\(T_{\text{Root}}\) = theoretical wing root thickness

\(n_1\) and \(n_2\) = exponentials (see Table III)

By separating carry-through and exposed wing (last bracketed term) in equation (14), the equation is more flexible and more accurate for a wider range of Earth-to-orbit vehicles and does not give large errors when, for instance, the area ratio of body-to-wing is large. In the extreme, a vehicle with a wide body and small fins (for wings) would have a large theoretical wing but very small exposed wing. In this latter case, an equation (which is based solely on theoretical wing) gives large errors in wing mass.

Similar problems were found with the first (or loads) term of the older wing equation since the equation did not recognize the lift distribution between body and wing for various body-to-wing area ratios. To improve this aspect of the equation, a body/wing efficiency factor, \(f\),
has been added along with a body-to-exposed wing area ratio. The body wing efficiency factor applied to the ratio recognizes these changes in wing/body lift distribution due to body size.

In the proposed equation, when the exposed wing area, $S_w$, approaches zero or $S_b/S_w \rightarrow \infty$ wing mass approaches zero, a desired result. Conversely, as wing area becomes larger (i.e., body area small), the term $S_b/S_w$ approaches zero and the first (bracketed) term in the proposed equation approaches a maximum value of $N_{zmL}$, which is the normal load factor times the landed mass of the entire vehicle. The vehicle is, in essence, a flying wing and all the vehicle load is carried by the wing. Both of the above changes in the wing equation, i.e., the modification to the loads or first bracketed term, and the wing and carry-through geometry, or last two terms, gives better agreement with known advanced SSTO point design vehicle wing masses. It does not give erroneous results when, as cited above, large changes are made in the relationship between wing and body planform areas.

Also included in equation (14) are wing and wing carry-through material/configuration constants which can be varied for a specific point design. All the wing-related constants are very sensitive to material selections, whether metallic, metallic composite, or organic composite; or to configuration, whether skin stringer or honeycomb. Other factors adding to the complexity of the selection is whether the wing is dry, wet (cryogenic or storables); whether the thermal protection system is integral with the wing or an add-on, therefore, not appearing in the wing weight statement but under thermal protection, or whether the wet wing is for a vertical takeoff or horizontal takeoff vehicle.
Equally complex is the wing carry-through which may be a separate structure (as in most aircraft) or integrated with body structure as in the shuttle and most of the SSTO's being studied.

Of equal importance to wing-mass trending are the methods used for comparison of unit structural masses. In the past, wing unit mass has been variously defined because of the variety of mass-to-area ratios used, namely for mass:

a) Exposed wing mass
b) Exposed wing plus wing carry-through
c) a) and b) with or without thermal protection

and for areas:

a) Exposed wing planform
b) Structural planform (exposed wing+structural wing carry-through)
c) Theoretical wing

To add to the complexity of the definitions, the effects of wing size on unit mass must be considered even when the mass-to-unit area ratio definition is agreed upon. From equation (14) and the definition of unit mass, as:

\[ m_{wu} = \frac{m_w}{S_w} \] (16)

It can be shown using equation (14) that, for equal wing loadings, the relationship between unit masses for dimensionally different but geometrically similar wings is related by some exponential, usually not zero, or:

\[ \frac{m_{wu2}}{m_{wu1}} = (F)^n \] (17)
where: $F$ is the ratio of dimensions of wing point designs, 2 and 1, and $n_3$ is some exponential. For the wing equation of reference 1, this value is 0.312 or the unit mass of the wing is growing roughly as the cube root of dimensional ratio even though wing loading is constant.

2.0 TAIL

Tail unit mass is also size dependent and the equations reflect this fact. One such equation for tail mass in reference 1 is:

$$m_t = V_t \left( S_t \right)^{1.24}$$  \hspace{1cm} (18)

Like the wing, "$V_t$" the material configuration constant, is dependent on the type of materials and construction used. Defining tail unit mass as gross structural tail mass divided by area, or dividing both sides of equation (18) by $S_t$, and assuming area is a function of dimension squared for a photographically enlarged tail:

$$\frac{m_{t,u_2}}{m_{t,u_1}} = (F)^{0.48}$$  \hspace{1cm} (19)

This is a similar result to that found in wing equations based on historical data. The in-house SSTO vehicle mentioned earlier has a tail profile area of 125.4 m$^2$ (1350 ft$^2$) while the shuttle has an area of only 38.4 m$^2$ (413 ft$^2$). The area ratio is 3.27 or dimensional ratio (assuming exactly similar geometry) is approximated by $(3.27)^{1/2}$ or size factor $F = 1.81$. From equation (19), the ratio in unit masses for the same technology level due to size effects alone is 1.33 or a 33 percent increase can be expected.
3.0 BODY GROUP

The body group is more difficult to assign to a mass equation because of the unique features of each generic design. For Level I masses, the major subelements included in the body masses are crew compartment, body shell, thrust structure and body flap. In earlier phase A/B shuttle studies (references 8 through 10), the main propellant tanks were carried in the main propulsion system mass unless they were load carrying. In this latter case, they were carried under body structure. For consistency in mass property reporting in this paper, they are always carried in the body group, whether load carrying or not.

The body mass equation recommended for Level I mass properties determination (when constants are selected for the generic design) is:

\[ m_b = B_c (N_c)^5 + B_b \cdot S_{WET_b} (N_z)^{1/3} + B_f (V_2) + B_o (V_1)^{1.1} + B_t \cdot T_{VAC_{Ne}} + B_{bf} \cdot S_f^{1.15} \]  

(20)

The above subelements are easily identifiable in most preliminary designs and enhance the accuracy of body mass and c.g. estimates as opposed to using an overall mass average for all body elements. In the above equation \( B_c, B_b, B_f, B_o, B_t \), and \( B_{bf} \) are material configuration constants for cabin, body proper, main fuel tank, main oxidizer tank, thrust structure, and body flap. The number of crew, mission specialists or passengers, \( N_c \) appears in the body equation since the size of the
pressurized cabin space is related to the number of persons to be accommodated. "N" is found from the product of safety factor times an ultimate load factor or is 1.4 X 3.75 for the shuttle.

In the body equation, non-optimums should be accounted for in the "B" constant. These should include door cutouts, doors, gear wells, intertank adapters (between main propellant tanks), thermal protection system support fairings, ring frames and stringers (where required), cargo bay, and other miscellaneous structural subelements. The term $S_{wet}$, for the body should not include those areas of the body for which the integral tankage forms the moldline.

For a tank, assuming equal ullage pressure, $P_u$, and material allowances, $\sigma$, tank wall thickness from the hoop stress formula is:

$$ t = \frac{P_u \cdot r}{\sigma} \quad (21) $$

Now tank surface area, assuming the same tank proportions, is given by:

$$ S_T = (\text{Constant}) \cdot r^2 \quad (22) $$

Tank mass is found from the product of area, thickness, and density of the material of which the tank is made, or:

$$ m_T = A_T \cdot t \cdot \rho \quad (23) $$

In the above equation for tank mass, the area of the tank can be defined as a constant times tank radius and the value of tank wall thickness by equation (21). Equation (23) then becomes:
\[ m_T = (\text{constant}) \, r^2 \times \frac{P_u}{\sigma} \quad r = (\text{constant}) \, r^3 \quad (24) \]

Now tank volume

\[ \text{Vol.} = \text{constant} \, (r^3) \quad (25) \]

Therefore, masses and volumes are both functions of \( r^3 \) or,

\[ \frac{\text{Tank mass}}{\text{Tank volume}} = (\text{constant}) \quad (26) \]

The above statement is valid assuming uniform pressure throughout the tank with no insulation, no changes in non-optimums or tank loads with size. In a gravity field, or under the influence of engine thrust, however, tank hydraulic pressures increase as a function of tank dimension along the line of action of thrust or gravity axis. At the bottom of a tank containing a dense propellant, such as LOX, the hydraulic pressure alone at the bottom of a 60 foot tank under 1.3 g's acceleration is 297 kPa (43 psi) compared to the shuttle external LOX tank design ullage pressure of 262 kPa (38 psi). It can be seen that tank mass is, therefore, sensitive not only to tank wetted area and wall thickness due to ullage pressure, but is also affected by hydraulic head linearly increasing from the free surface of the fluid. Because of this, the mass center of tanks carrying relatively dense propellants falls aft of the tank centroid and further, the tank structural mass grows faster than a linear factor with volume. Because of hydraulic head, if
convenient from other packaging considerations, it is desirable to utilize tanks for dense propellants such as RP and LOX, which are as short as possible along the resultant of the thrust and gravity axes. The hydraulic head effects of LH₂ are not considered significantly large and are ignored for purposes of trending. LH₂ insulation masses, however, are significant for a hydrogen tank and should be accounted for. Generally, it is found there is no payload advantage in insulating LOX tanks (ref. 12) but other factors may make this necessary.

Tank configuration effects are significant and some tanks are long and slender giving rise to higher hydraulic heads for tanks carrying relatively dense propellants such as LOX and RP. Tapered tanks are also heavier.

4.0 INDUCED ENVIRONMENTAL PROTECTION

In the past, induced environmental protection (when distinct and separate such as the shuttle system of silica tile) has often been based on a constant times wetted area. In this paper, this equation is being updated to reflect sensitivities to ballistic coefficient and thermal capacity of the backup structure, namely:

\[ m_{TPS_u} = K_R \left( \frac{1}{\bar{t}} \right)^a \left( \frac{m_e}{S C_L} \right)^b (S_{WET_v}) \]  

(27)

where \( K_R \) is the material/configuration constant for the thermal protection and \( \bar{t} \) is a constant reflecting heat capacity of the backup structure; \( m_e \) is vehicle entry mass.
The equation is based on an assumption of equilibrium glide conditions and equal entry times for the point design and off-point design vehicle. If flow over the entire vehicle is laminar during the entire entry, the exponential "b" would be 0.5 for the assumptions made. On the other hand, if the flow over the entire vehicle is turbulent during the entire entry, the exponential would be 0.8. Because of the lack of current knowledge as to when transition occurs and over what percentage of the vehicle body, it is not possible to determine exact vehicle point design thermal protection masses. The equation is useful, however, in vehicle design when differing assumptions are made for entry planform loading, $C_L$, backup structure, and the amount of turbulent flow merely to establish trends in vehicle mass. The estimator studying such trends between one point design and another should be consistent in the exponential used; without more detailed information, an exponential of 0.65 would be a reasonable choice.

A value for the constant $K_R$ was obtained for the reusable surface insulation concept by substituting the known quantities in equation (27) and solving for $K_R$ for the shuttle design. The shuttle-derived constant is based on the assumption that flow is laminar over the vehicle. The constant does not include allowance for the mass of carrier panels. Many of the concepts being considered require carrier panels on the isogrid main body propulsion tanks but not, for instance, on a smooth wing stress skin.

5.0 LANDING, DOCKING, AND RECOVERY

For this category in the Level I weight statements, the landing gear mass is taken simply as a percentage of landed mass. For the shuttle, the manipulator and tank separation system are included in this category.
When these masses are deleted in the shuttle Mass Properties Report (ref. 7), the landing gear mass, as a percentage of landed mass, equals 3.3 percent.

If composites are extensively used, it is estimated that this figure can be reduced to 3.0 percent. If skids were operationally practical and adaptable to a large SSTO, the 3.0 percent figure could conceivably be reduced further to an estimated value of 2.55 percent. It is estimated that conventional gear without brakes would yield a similar figure to that shown for skids, i.e., 2.55 percent. (A ground arrester would be employed for stopping such a vehicle.) Separating the manipulator and docking functions is desirable since these items are mission oriented and not necessarily related to vehicle size. The equation for trending purposes then becomes:

\[ m_g = K_L \times m_L + m_{ma} + m_S \]

Where: \( K_L \) is the landing gear constant and \( m_{ma} \) and \( m_S \) are manipulator and separation systems masses, respectively; and \( m_L \) is the landed mass.

(b) PROPULSION GROUP

6.0 PROPULSION ASCE:IT

For sensitivity studies, once a "point design" main propulsion system is established, the total main propulsion system mass (not including main tankage) can be expressed as:
\[ n_{\text{eng}} = \left[ n_{\text{ph}} + n_{\text{n}} (n - 1) + n_{\text{ne}} \left( \frac{E_2 - E_1}{P_c} \right) + n_{\text{na}} \left( \frac{E_3 - 1}{(nP_c)^2} \right) \right] + \frac{R_{\text{pf}} + R_{\text{ga}}}{\lambda_{\text{se}}} \left( \hat{\Gamma} \right) \cdot N_e \]

This equation is based on unpublished information obtained from LeRC. The terms in order of appearance in equation (29) refer to power head mass, fixed position nozzle mass, nozzle extension, extension actuator, pressurization and feed systems, and engine girthal actuator mass. The latter is based on vacuum specific impulses. In the above equation, \( N_e \) is the number of engines in any generic grouping. This not only applies to type of fuel used, but also as to whether the engine is fixed or gimbaled, extendable or fixed nozzle. If the vehicle trending involves a fixed propulsion system design, then the bracketed term can be replaced with a constant and \( N_e \) or \( m \) varied with vehicle size for a constant or nearly constant thrust-to-weight.

7.0 PROPULSION - REACTION (OR ATTITUDE) CONTROL, RCS

The mass equation for the control (or attitude control) system used and devised here is based on the following assumptions:

1. That angular acceleration rate in pitch, yaw, and roll is directly proportional to applied torque and inversely as vehicle moment of inertia.

2. That applied torque, with only small variations, is proportional to vehicle size (e.g., double the physical size, or \( L_r \), and moment arms from vehicle e.g. to RCS pods are roughly doubled).
3. That vehicle moment of inertia is proportional to the product of vehicle mass and radius of gyration or equals \( m_e \times (e \times L_r)^2 \) where "e" is a constant identifying radius of gyration location when multiplied by vehicle reference length, \( L_r \).

When utilizing the above assumption and further assuming no change in the total RCS mission impulse required or angular rates:

\[
m_R = R_{RCS} \times m_e \times L_r
\]  

(30)

A characteristic point design constant, \( R_{RCS} \), for equation (30) for a storable system can be determined from the shuttle orbiter. Any significant change in an off-point design vehicle from the shuttle, such as the transfer of aft RCS modules from body to wing tips, substitution of cryogenic system for storable, change in engine thrust-to-vehicle mass ratio or total impulse requirement would require a re-determination of the values of the constant to avoid gross errors in mass estimation of the system. A cryogenic RCS system has a somewhat higher dry mass for the typical Earth-to-orbit transportation system, but the lower corrosivity and toxicity of these propellants may be the determining factors. For very high total impulse requirements, the cryogenic system, due to its higher specific impulse, has a clear-cut advantage in mass over the storable RCS system.

8.0 PROPULSION - ORBITAL MANEUVERING SYSTEM, OMS

Because of the mass advantage in propellant savings for the higher OMS impulse requirements over RCS, the cryogenic system in this case has a more clear-cut advantage. In the equations, engines and
tanks are treated separately since, unlike the RCS system, CMS total ΔV requirements typically vary widely from mission to mission. The OMS engine mass including tanks and feed system mass is:

\[ m_o = M_{me} X N_e X T_{vac} + M_t X m_{op} \]  

(31)

Where: \( M_{me} \) and \( M_t \) are constants for the maneuver engine, tankage and pressurization and feed, respectively.

(c) POWER GROUP

9.0 PRIME POWER

Both the auxiliary power unit and fuel cells are included in this category. The Auxiliary Power Unit is assumed to be designed by peak aerocontrol system requirements with engine gimbal requirements having a secondary effect. Constants shown in Table III are based on shuttle and the following surface control rates:

- Elevon = 20°/sec
- Rudder = 14°/sec
- Body flap = -3°/sec + 1°/sec
- Speedbrake (Priority rating logic)

For these rates, it is assumed that control surface power is directly proportional to the total aerocontrol movable surfaces exclusive of speedbrake. It is further assumed that, with reasonable accuracy, the control surface power constant, \( PW_c \), in the equation below
could be altered in direct proportion to the increase or decrease in control surface rates since power is proportional to rate. The Prime Power mass is then:

\[ M_{\text{Pow}} = P_{W_c} \times S_c + P_{W_e} \times \Sigma T_{VAC} + P_{W_b} \times m_a \]  

(32)

Where: \( P_{W_c} \), \( P_{W_e} \), and \( P_{W_b} \) are constants reflecting demands from surface controls, main engines, and avionics subsystems, respectively.

Hydraulic power unit mass remains essentially constant with a pressure increase to \( 3.45 \times 10^6 \text{ N/m}^2 \) (5,000 psi); however, the mass of this subsystem will decrease with the utilization of accumulators to handle peak loads due to the reduction in total horsepower required and the more efficient operation at normal power loads of the unit (i.e., reduced peak to normal power ratios).

Prime power also includes battery power. For this subsystem, it is assumed that its mass is directly proportional to the mass of the avionics, \( m_a \), for a given technology level, the last term in equation (32). A mass for an advanced technology avionic system is given in reference 6.

10.0 ELECTRICAL CONVERSION AND DISTRIBUTION

Electrical conversion and distribution is assumed to be proportional to vehicle landed mass or:

\[ m_{eL} = E \times m_L \]  

(33)
The above constant, $E$, is assumed to be somewhat sensitive to mission but is relatively insensitive to configuration.

11.0 HYDRAULIC CONVERSION AND DISTRIBUTION

The mass trend equation for hydraulic conversion and distribution is similar to the prime power equation for hydraulics but with altered constants, namely:

$$m_h = H_{cs} \times S_{sc} + H_e \times N_e \times T_{vac}$$  \hspace{1cm} (34)

Where: $H_{cs}$ and $H_e$ are constants related to surface controls and engine gimbal actuation, respectively.

The baseline constants in Table III are derived from the shuttle. For advanced technology hydraulics (such as high pressure system), the two constants above are reduced due to the utilization of smaller diameter hydraulic lines.

12.0 SURFACE CONTROLS

Surface control mass (actuators, etc.), like the control power source, is assumed to be directly proportional to movable surface control area for a given rate, or:

$$m_{sc} = S_{sc} \times S_c + S_{pc}$$  \hspace{1cm} (35)
In the above equation, $S_c$ is surface control area and $S_{sc}$ and $S_{pc}$ are surface control and pilot control related constants. It is assumed that pilot related controls are independent of vehicle size.

(d) MISCELLANEOUS

13.0 AVIONICS

Avionics mass is assumed to be relatively insensitive to vehicle size and equals:

$$M_{av} \left( m_d \right)^{1/3}$$

Where: $M_{av}$ is a constant and $m_d$ is dry mass.

14.0 ENVIRONMENTAL CONTROL

Environmental control system is assumed to be sensitive to the wetted area of pressurized compartments or in terms of volume, an exponential of two-thirds. Pressurized compartments include wheel wells and cabin. (On the shuttle, the cargo bay is not pressurized.) For the electronics, it is assumed that heat input to the cabin, for a given technology, is directly proportional to electronics mass.

The total system mass for the cabin is relatively insensitive to mission duration. (The D factor shown below reflects principally oxygen container mass for the number of man-days on orbit.) or:

$$m_{ENV} = E_c \left( V_p \right)^{.66} + E_o \times N_c \left( D \right) + E_a \, m_a$$  \hspace{1cm} (36)
15.0 PERSONNEL PROVISIONS

This category includes the fixed life support system, food, waste, and water management systems, fire detection, pilot and crew stations. This category is relatively insensitive to mission duration beyond an estimated one-day limit. Below a one-day limit, it is assumed that the bulk of the food, waste, and water management systems could be removed, or that "PPf" in the equation below would be reduced to zero. However, individual personnel provisions, such as seats, are directly related to the number of pilots, mission specialists, and passengers, and must be included. Therefore:

\[ m_{PP} = PP_f + PP_s (N_c) \] (37)

16.0 MARCIN (OR GROWTH ALLOWANCE)

Margin is equal to a constant times the dry mass of all the sub-systems less engine mass. (NOTE: On the shuttle, a 10 percent growth margin is already included in the engine mass and the equation below is structured to be consistent with this practice.)

\[ m_{\text{margin}} = \text{MAR} (M_d - N_e m_{\text{eng}}) \] (38)

17.0 PERSONNEL

This category includes mass of crew, mission specialists, etc., and personnel-related GFE equipment and accessories, or:

\[ m_{\text{per}} = P_m + P_p X N_c \] (39)
The above equation is applicable to one or more crew and is zero for an unmanned vehicle.

18.0 PAYLOAD PROVISIONS

Payload provisions are a fixed input to allow for any special installation and mounting equipment.

(e) PAYLOAD

19.0 PAYLOAD RETURNED

Payload returned is a fixed input.

(f) FLUIDS INVENTORY (ON ORBIT AND ENTRY)

20.0 RESIDUAL AND UNUSABLE FLUIDS

Residual and unusable fluids include gaseous propellants and pressurization gases in addition to trapped propellants. The amount of unusable fluids depends on manifold, line and sump design. For a given vehicle the residual or unusable fluids is taken as a constant times the ascent propellant mass to an exponential or:

\[ m_{uf} = R_{rf} \times m_p^{.79} \]  

(40)

21.0 RESERVES OMS AND RCS

Orbit maneuvering system and attitude control system reserves are determined by:

\[ m_{ROMS} = m_L \left[ e \left( \frac{R_o \Delta V_o}{T_s \Delta m \Delta g} \right) - 1 \right] \]  

(41)
\[ m_{RRCS} = m_L \left[ e \left( \frac{R_r \Delta V_{RO}}{I_{s\,ro} \cdot g} \right) - 1 \right] \] (42)

Where: \( R_o \) and \( R_r \) are constant percentages of \( \Delta V \) requirements.

An overall constant may be substituted for the bracketed terms in equations (41) and (42) providing there are no changes in mission or engine design. The same is true for equations (43) and (44) in the next two sections.

22.0 RCS PROPELLANT ENTRY

Based on estimated reentry attitude control requirements of the shuttle and interpreted as an equivalent \( \Delta V \):

\[ m_{RCS_e} = m_e \left[ e \left( \frac{\Delta V_{Re}}{I_{s\,Re} \cdot g} \right) - 1 \right] \] (43)

The specific impulse in the above equation is a degraded value reflecting a lower performance due to increased ambient pressure during entry. Entry mass is the vehicle mass after depletion of all usable RCS propellants.

23.0 RCS AND OMS PROPELLANT CONSUMABLES

On-orbit and deorbit attitude control and maneuver propellants are:

\[ m_c = m_0 + m_t \]
or:

\[ m_c = m_{DESC} \left[ e \left( \frac{\Delta V_0}{I_{s_m} g} \right) -1 \right] + m_{DESC} \left[ e \left( \frac{\Delta V_{Ro}}{I_{s_ro} g} \right) -1 \right] \]  

Both the RCS and OMS requirements, it should be noted, are based on the vehicle mass at descent, which is defined herein as the vehicle mass with return cargo, entry RCS, and all residuals and reserves. In so doing, on-orbit maneuvers are assumed to take place after discharge of cargo. This is done for consistency in vehicle sizing but, of course, is sensitive to mission details.

(g) PAYLOAD DELIVERED

24.0 CARGO DISCHARGED

On-orbit net cargo or cargo delivered less cargo returned.

(h) FLUIDS INVENTORY (ASCENT PHASE)

25.0 ASCENT RESERVES AND ASCENT RESIDUALS

\[ m_{ARES} = m_{INJ} \left[ e \left( \frac{R_{ar} \Delta V_{IDEAL}}{I_{s_e} g} \right) -1 \right] + R_{ap} \cdot m_p \]  

The reserve propellant requirements are taken as a fraction of the ascent ideal \(\Delta V\) to allow for launch dispersions. In addition, a percentage of the ascent propellant must be allowed for residuals.
Injected mass is gross less start-up and ascent propellants, inflight losses, ascent residuals and reserves; the ascent residuals and reserves being dumped prior to on-orbit maneuvers. A constant may also be substituted here for the bracketed term with reasonable accuracy providing trajectory and vehicle drag characteristics remain reasonably constant.

26.0 **INFLIGHT LOSSES**

Inflight losses include boiloff, prime power, environmental control, and hydraulic effluents or for mass estimating purposes:

\[ m_{\text{INF}} = R_{\text{INF}} \times m_p \]  \hspace{1cm} (46)

For very accurate trajectory analysis, the effluents can be reflected in fractionally lower specific impulse but it is considered sufficiently accurate to subtract one half the losses from the vehicle mass after depletion of usable ascent propellant and prelaunch losses.

27.0 **ASCENT PROPELLANT**

Ascent propellant is taken as usable propellant, or as the propellant load less prelaunch boiloff, preignition and thrust buildup losses or:

\[ m_p = m (1 - R_p) \]  \hspace{1cm} (47)
Where: $R_{pl}$ is a small fractional percentage (.001 to .002) and for system sizing purposes is inconsequential and, if not accounted for, merely results in a fractional percent error in the interpretation of ullage volume.

IV. GENERAL DISCUSSION

In the previous paragraphs, mass estimating relationships have been discussed in the order in which they appear in the mil standard 38310 (ref. 11). The trending equations have been structured to give better results for Earth-to-orbit transports. In addition, equation format has been improved over earlier equations which had previously been utilized primarily for commercial and military aircraft. Suggested constants have been provided in Table III.

Of all the subsystems, body structure is one of the heavier elements and on the shuttle, constitutes approximately 27 percent of the total dry mass. Similarly, this structural group is probably one of the most variable in total mass being heavily dependent upon configuration factors. One configuration factor is the cargo bay shape and location. For a single-stage-to-orbit winged vehicle, the cargo bay could be located in the nose of the vehicle and the main propellant tanks designed to form a simple body of revolution, one of the lightest possible structural masses could result. For aerodynamic and other mission oriented reasons, this may not be practical and various other shapes evolve. If cargo return is required from orbit, it may be necessary to locate the cargo bay in the vicinity of
the vehicle normal c.g. to minimize "cargo-in" and "cargo-out" c.g. excursions during horizontal flight. This latter location generally will effect a decrease in propellant tankage packaging efficiency and an increase in body structural non-optimums.

For the above reasons, no one set of constants can be provided which will apply to all body structural designs. The same is true, but to a lesser extent, of all subsystems. It is, therefore, up to the estimator to alter constants utilizing available information developed in structural analyses. In Table IV, suggested constants are given for main propellant tankage. If no detailed information is available, body mass can be structured from the tank mass estimating relationships of this table combined with estimates for intertank adaptors, fairings, cargo bay structure, non-optimums, and the other subelements suggested by equation (20) in the text. Once established for one size of vehicle body structure masses of the same generic design of other sizes can be obtained.

While the method of reporting mass properties follows closely that recommended for subsystems and fluids inventory in reference 11, it has been modified slightly so that it can serve both for Level I weight estimates of subsystems and also for sequential mass estimates. The sequential mass statement accounts for expendables during the mission such as main rocket and reaction control propellant and discharged cargo, etc. In this regard, item 19, Table III, has been changed to "cargo returned" while item (24), net "cargo discharged" has been added. In the more common type of missions, cargo up or down either remains the same, or decreases; and sequentially, vehicle mass is
continually decreasing from liftoff to landing. However, if ascent cargo is less than return cargo, (such as on a retrieve mission), the cargo "discharged" on-orbit, item (24), should be listed as a negative quantity. A separate column can be used adjacent to individual subsystem masses for sequential masses. This mass generally showing a decreasing value from gross liftoff to dry mass as stated above.

V. EXAMPLE STRUCTURE

Example structure is shown for in-house study vehicle EN 155 in Appendix D. Composites and honeycomb were extensively used to obtain low mass. In Appendix E, unit masses of various types of crossections are shown. The sections apply to areas where exterior peak temperatures are approximately 1000° C (1800° F), and liquid hydrogen is stored internally. Two crossections (E-4 and E-5) however, are representative of areas where no fluid containment is required. The unit masses shown tend to be considerably lower than real structure since door cutouts, secondary structure, and other non-optimums are not included. The final figure in this section shows TPS unit masses as a function of peak temperatures for both metallic and ceramic reusable surface insulation. This data could be applied to mass estimation of the TPS on a vehicle for which isotherms for peak temperatures are known as opposed to use of the more general equation (27) in the text.
CONCLUSIONS

Several techniques for estimating the mass properties of Earth-to-orbit transportation systems have been reviewed. Where appropriate, basic equations and related constants have been revised, and estimating procedures given. Based on in-house studies, the following conclusions are offered:

1. The overall vehicle trending technique presented is considered very rapid for resizing a vehicle for which propellant and overall inert mass are already known and no significant configuration changes are involved in trending from one generic-point design to another.

2. Analysis on a more detailed basis by Subsystem Trending as outlined in this report has been found to be very accurate but requires more time and detailed information on individual subsystems.

3. The equations developed for subsystem trending are considered useful for parametric studies wherein the impact of one subsystem parameter on the overall system is being evaluated.

4. No one set of constants can be provided which will be suitable for each design. Constants provided are reasonable baselines from which more mature mass estimates can be developed when more detailed subsystem information is available.
INTERNATIONAL SYSTEM OF UNITS CONVERSION FACTORS, PHYSICAL CONSTANTS, AND PREFIXES

(a) Conversion factors

<table>
<thead>
<tr>
<th>CONVERT FROM</th>
<th>TO</th>
<th>MULTIPLY BY</th>
</tr>
</thead>
<tbody>
<tr>
<td>INCHES</td>
<td>METERS</td>
<td>.025 400</td>
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<tr>
<td>INCHES²</td>
<td>METERS²</td>
<td>.000 645 160</td>
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<tr>
<td>FOOT</td>
<td>METERS</td>
<td>.304 800</td>
</tr>
<tr>
<td>FOOT²</td>
<td>METERS²</td>
<td>.092 903 040</td>
</tr>
<tr>
<td>POUNDS</td>
<td>KILOGRAMS</td>
<td>.453 592 370</td>
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<td>POUNDS/INCHES³</td>
<td>KILOGRAM/METER³</td>
<td>27 679.905</td>
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<td>KILOGRAM/METER³</td>
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<td>FOOT/SEC</td>
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<td>FOOT/SEC²</td>
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<tr>
<td>POUNDS/INCH²</td>
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<tr>
<td>POUNDS/FOOT²</td>
<td>NEWTON/METER²</td>
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<td>(OR PASCALS)</td>
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</table>

(b) Prefixes

The names of multiples and submultiples of SI units may be formed by application of the prefixes:

<table>
<thead>
<tr>
<th>FACTOR BY WHICH UNIT IS MULTIPLIED</th>
<th>PREFIX</th>
<th>SYMBOL</th>
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<tbody>
<tr>
<td>10⁹</td>
<td>giga</td>
<td>G</td>
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<tr>
<td>10⁶</td>
<td>mega</td>
<td>M</td>
</tr>
<tr>
<td>10³</td>
<td>kilo</td>
<td>k</td>
</tr>
<tr>
<td>10²</td>
<td>hecto</td>
<td>h</td>
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<tr>
<td>10</td>
<td>deka</td>
<td>da</td>
</tr>
<tr>
<td>10⁻¹</td>
<td>deci</td>
<td>d</td>
</tr>
<tr>
<td>10⁻²</td>
<td>centi</td>
<td>c</td>
</tr>
<tr>
<td>10⁻³</td>
<td>milli</td>
<td>m</td>
</tr>
</tbody>
</table>

(c) Physical constants

\[ g = 9.8 \text{m/sec}^2 \text{ or } 32.174 \text{ feet/Sec}^2 \]
REFERENCES


## TABLE I
TRENDING SUBSYSTEM GROWTH VERIFICATION

<table>
<thead>
<tr>
<th>SUBSYSTEM</th>
<th>MASS RELATED TO</th>
<th>APPROXIMATE VEH. REF. LENGTH PROPORTIONALLY</th>
<th>APPROX. % OF TOTAL MASS</th>
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<tr>
<td>1.0     WING GROUP</td>
<td>PLANFORM AREA</td>
<td>$L^2$</td>
<td>10</td>
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<td>2.0     TAIL</td>
<td>PLANFORM AREA</td>
<td>$L^2$</td>
<td>2</td>
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<td>3.0     BODY</td>
<td>BODY WETTED AREA</td>
<td>$L^2$</td>
<td>26</td>
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<tr>
<td>TANKAGE</td>
<td>TANK VOLUME</td>
<td>$L^3$</td>
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<td>THRUST STR.</td>
<td>THRUST</td>
<td>$L^3$</td>
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</tr>
<tr>
<td>4.0     TPS</td>
<td>WETTED AREA</td>
<td>$L^2$</td>
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<tr>
<td>5.0     LANDING GEAR</td>
<td>LANDED MASS</td>
<td>$L^2$</td>
<td>3</td>
</tr>
<tr>
<td>6.0     PROPULSION ASCENT</td>
<td>GROSS MASS</td>
<td>$L^3$</td>
<td>22.6</td>
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<td>8.0     PROPUL. AUX. (INCL. PROP.)</td>
<td>INJECTED MASS</td>
<td>$L^2$</td>
<td>1.7</td>
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<tr>
<td>9.0     PRIME POWER</td>
<td>CONTROL SURFACE AREA</td>
<td>$L^2$</td>
<td></td>
</tr>
<tr>
<td>10.0    ELEC. POWER</td>
<td></td>
<td>$L^2$</td>
<td></td>
</tr>
<tr>
<td>11.0    HYDRAULICS</td>
<td>CONTROL SURFACE AREA</td>
<td>$L^2$</td>
<td>1.7</td>
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<tr>
<td>12.0    SURFACE CONTROLS</td>
<td>AREA</td>
<td>$L^2$</td>
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</tr>
<tr>
<td>13.0    AVIONICS</td>
<td>&quot;MISSION DEPENDENT&quot;</td>
<td>CONSTANT</td>
<td></td>
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<tr>
<td>14.0    ENVIRONMENTAL CONTROL</td>
<td>FUNCTION OF PRESSURIZED TOTAL VOLUMES</td>
<td>CONSTANT</td>
<td></td>
</tr>
<tr>
<td>15.0    PERSONNEL PROVISIONS</td>
<td>SIZE OF CREW</td>
<td>CONSTANT</td>
<td></td>
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<tr>
<td>16.0    GROWTH</td>
<td>CONSTANT X DRY WT.</td>
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<td>SUBSYSTEM</td>
<td>WT, LB</td>
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<td>---------------------</td>
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<tr>
<td>1.0 WING GROUP</td>
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<td>6.0 PROPULSION</td>
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<td>7.0 PROPULSION, RCS</td>
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<td>9.0 PRIME POWER</td>
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<td>10.0 ELEC CONV AND DISTRIBUTION</td>
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<td>14.0 ENVIRONMENTAL CONTROL</td>
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<td>15.0 PERSONNEL PROVISIONS*</td>
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<td>16.0 MARGIN</td>
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<td>DRY WEIGHT</td>
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<td>154,739</td>
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</tbody>
</table>

P/L PROVISIONS (467 LB) ARE INCLUDED IN PERSONNEL PROVISIONS BY ROCKWELL INTERNATIONAL, NASA ORBITER PRIME CONTRACTOR.
<table>
<thead>
<tr>
<th>Item Description</th>
<th>R.I. 12/76 Wt, LB</th>
<th>SEMP Program Wt, LB</th>
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<td>DRY WEIGHT</td>
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<td>154,739</td>
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<tr>
<td>17.0 PERSONNEL</td>
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<td>18.0 PAYLOAD ACCOMMODATIONS</td>
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<td>19.0 CARGO (RETURNED)</td>
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<td>20.0 RESIDUAL FLUIDS</td>
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<td>LANDED WEIGHT</td>
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<td>21.0 OMS AND RCS RESERVES</td>
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<td>ENTRY WEIGHT</td>
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<td>22.0 RCS PROPELLANT (ENTRY)</td>
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<td>DESCENT WEIGHT</td>
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<td>24.0 CARGO DISCHARGED</td>
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<tr>
<td>INJECTED WEIGHT</td>
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<td>240,990</td>
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<td>25.0 ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS</td>
<td>2,344</td>
<td>4,454</td>
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<tr>
<td>26.0 INFLIGHT LOSSES</td>
<td>2,753</td>
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<tr>
<td>27.0 ASCENT PROPELLANT</td>
<td>5,206</td>
<td>5,206</td>
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<tr>
<td>GROSS LIFTOFF WEIGHT</td>
<td>255,170</td>
<td>251,313</td>
</tr>
</tbody>
</table>
TABLE III - VEHICLE SUBSYSTEM TRENDING

EQUATIONS AND VARIABLES

\[
m_w = m_l \left(1 + \left[1 + \frac{S_b}{S_w} \right] \frac{1}{T_r} \frac{S_w}{T_r} \right) \cdot 0.386 \cdot W_{m} \cdot \frac{L_w}{L_s} \cdot 0.572 \quad W_{c} \cdot \frac{L_c}{L_s} \cdot 0.572
\]

WHERE:

- \( W_{m} \) = Ultimate normal load factor for the design condition (or 1.4 \( X \) 2.5 g's subsonic maneuver when ascent wing loading is limited to the subsonic maneuver case.

- \( m_l \) = Mass of vehicle at landing.

- \( S_b \) = Body planform area

- \( S_w \) = Exposed wing planform

- \( T_r \) = Exposed wing root chord max. thickness

- \( L_w \) = Exposed total structural wing span

- \( L_b \) = Body width at wing body juncture

EQUATION CONSTANTS

Exposed wing material/configuration constants

- \( W_{m} = 0.286 \) - Aluminum skin/stringer, dry wing, no TPS
- \( = 0.343 \) - same as above but wet wing for storable propellant.
- \( = 0.229 \) - Metallic composite (Boron Aluminum) honeycomb dry wing, no TPS.
- \( = 0.265 \) - 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

- \( W_c = 0.0267 \) - dry carry-thru (integral)
- \( = 0.0347 \) - wet carry-thru (integral)
- \( = 0.100 \) - dry carry-thru (conven.)
- \( = 0.12 \) - wet carry-thru (conven.)

Wing/Body efficiency factor

- \( f = 0.20 \) for conventional vehicle to
- \( 0.15 \) for control configured vehicle.

Tail material/configuration constant

- \( V_t = 1.872 \) - aluminum skin/stringer, no TPS.
- \( = 1.108 \) - Metallic composite structure, no TPS.
- \( = 1.000 \) - graphite epoxy composite structure, no TPS.
### TABLE III (CONT'D)

#### EQUATIONS AND VARIABLES

**全体 (Tail Group (Cont'd))**

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Equation</th>
</tr>
</thead>
<tbody>
<tr>
<td>3.0 BODY GROUP</td>
<td></td>
</tr>
</tbody>
</table>

\[ m_b = B_c \left( N_c \right)^5 + B_o \cdot S_{Wet_b} \cdot (N_z)^{1/3} + B_f \cdot V_2 + B_o \cdot V_1^{1.1} + B_e \cdot \left[ Ne^1 T_{vac} \right] \]

+ \[ N_e^2 T_{vac2} + B_{br} \cdot S_f \cdot 1.15 \]

**Where:**

- \( N_c \): number of crew
- \( N_z \): ultimate normal load factor (same as for wing)
- \( S_{Wet_b} \): wetted area of body structure less areas of main propellant tanks which dual as body shell
- \( V_2 \): volume of main fuel tank
- \( V_1 \): volume of main oxidizer tank
- \( N_e \): number of engines (Type 1 or Type 2)
- \( T_{vac} \): engine vacuum thrust
- \( S_f \): body flap planform area

#### EQUATION CONSTANTS

- \( B_c = 1.500 \): super alloy honeycomb, hot structure.

**CABIN CONSTANTS**

- \( B_c = 2043 \): full windshield aluminum construction.
- \( B_c = 1293 \): above with no windshield.
- \( B_c = 1740 \): full windshield composite construction.
- \( B_c = 1140 \): above with no windshield and composite construction.

**BODY CONSTANTS**

- \( B_o = 2.72 \): composite structure, no TPS
- \( B_o = 3.20 \): aluminum structure, no composite, no TPS
- \( B_o = 3.40 \): hot metallic Ti/Rene HC, no TPS required
- \( B_o = 4.43 \): moldline tankage; tank, body structure, cryogenic insulation integrated.

**TANK CONSTANTS**

- \( B_f = \text{fuel tank constant (see table IV)} \)
- \( B_o = \text{oxidizer tank constant (see table IV)} \)

**BODY FLAP CONSTANTS**

- \( B_{bf} = 1.59 \): hot structure
- \( B_{bf} = 1.38 \): aluminum str. no TPS
### TABLE III (CONTINUED)

#### 4.0 THERMAL PROTECTION SYSTEM

**EQUATIONS**

\[ m_{TPS} = K_T \left( \frac{1}{\tau} \right) \left( \frac{m_v}{S_{CL}} \right) b \cdot S_{wet,y} \]

**WHERE:**
- \( K_T \): material/configuration constant for the TPS
- \( \tau \): equivalent thermal thickness of backup structure (inches)
- \( m_v \): vehicle entry mass
- \( S \): vehicle total entry planform area
- \( C_L \): vehicle average lift coefficient during peak heating
- \( S_{wet,y} \): vehicle wetted area
  - \( a = 0.302 \)
  - \( b = 0.5 \) laminar flow
  - \( b = 0.8 \) turbulent flow

#### 5.0 LANDING GEAR

**EQUATIONS**

\[ m_y = K_L \times m_L + m_{ma} + m_s \]

**WHERE:**
- \( m_L \): landed mass
- \( m_{ma} \): manipulator mass (for shuttle = 805 lb)
- \( m_s \): separation system mass

#### 6.0 PROPULSION

**EQUATIONS**

\[ M_{eng} = \left[ R_p h + R_n \left( c_2 - c_1 \right) + R_{ne} \left( c_2 \wedge c_1 \right) + R_{na} \left( c_2 \wedge c_1 - 1 \right) + R_{pf} \left( c_2 \wedge c_1 \right) + R_{g} \right] \cdot m_{em} \]

**EQUATION CONSTANTS**

**RSI MATERIAL CONSTANT**
- \( K_T = \) .140 RSI (shuttle technology)
- \( = \) .110 RSI Advanced
- \( = \) .145 metallic

**BACKUP STRUCTURE CONSTANTS EQUIVALENT THICKNESS**
- \( \bar{\tau} = .100 \) aluminum skin stringer
- \( = .085 \) Titanium
- \( = .115 \) Graphite/Epoxy

**ENTRY TRAJECTORY**
- \( C_L = .65 \) average to mach 10
  - (shuttle VTO/RSI)
- \( = .52 \) average to mach 10 horizontal take-off metallic TPS

**BASIS FOR SHUTTLE \( m_e = 49.9 \) LB/FT²**
- \( K_L = a \) constant percent age of landed mass for landing gear
- \( = .0330 \) shuttle gear
- \( = .0300 \) advanced composite gear
- \( = .0255 \) composite skid system, or composite wheel system with no brakes

*DOES NOT INCLUDE CARRIER PANELS*
6.0 PROPELLION (CONT'D)

WHERE:

\( m_{\text{eng}} \) = total mass of all generic engine groupings
\( \varepsilon_1 \) = expansion ratio of first expansion
\( \varepsilon_2 \) = expansion ratio of second expansion (when applicable)
\( P_c \) = chamber pressure
\( m \) = mass flow per engine
\( N_e \) = number of engines of any one generic grouping
\( I_{se} \) = vacuum specific impulse (nozzle extended for a two position engine)

EQUATIONS

POWER HEAD CONSTANTS

\( R_{ph} \) = sec, power head mass constant
\( = 2.48 \text{ LOX/hydrocarbon staged combustion} \)
\( = 2.10 \text{ LOX/hydrocarbon LHx generator} \)
\( = 5.34 \text{ LOX/LH}_2 \)
\( = 5.18 \text{ dual fuel engine, } P_c = 3000 \text{ psi} \)
\( = 2.48 \text{ LOX/hydrocarbon staged combustion } P_c = 4000 \text{ psi} \)
\( = 2.10 \text{ LOX/hydrocarbon LHx generator, } P_c = 4000 \text{ psi} \)

NOZZLES

\( R_n \) = sec, basic nozzle mass constant
\( = 0.0194 \text{ LOX/LH}_2 \)
\( = 0.0077 \text{ LOX/hydrocarbon} \)
\( = 0.015 \text{ EN 155 (dual fuel)} \)

NOZZLE EXTENSIONS

\( R_{ne} \) = sec, nozzle extension mass constant
\( = 9.943 \text{ LOX/LH}_2 \)
\( = 6.054 \text{ LOX/HC} \)

NOZZLE EXTENSION ACTUATION

\( R_{na} \) = sec/in. constant for extension mechanism
\( = 60.94 \text{ LOX/LH}_2 \)
\( = 36.16 \text{ LOX/HC} \)

EQUATION CONSTANTS
### 7.0 PROPULSION RCS (ATTITUDE CONTROL SYSTEM)

\[ m_e = R_{RCS} \times M_e \times L_r \]

WHERE:
- \( m_e \) = entry mass
- \( L_r \) = vehicle reference length

### 8.0 PROPULSION, OMS (ORBITAL MANEUVER SYSTEM)

\[ m_0 = M_e \times N_e \times T_{vac} + M_t \times m_{op} \]

WHERE:
- \( N_e \) = number of maneuver engines
- \( T_{vac} \) = vacuum thrust per maneuver engine
- \( m_{op} \) = required maneuver propellant

### EQUATION CONSTANTS

**PRESSURIZATION AND FEED**
- \( R_{pf} \) = lines manifold and pressurization system
  - 1.64 current technology
  - 1.4 composite/metallic feedlines

**GIMBAL ACTUATORS**
- \( R_{ga} \) = .00129

- \( R_{RCS} \) = point design all-up system
  - constant including tanks, pressurization and feed, gimbal actuator, etc.
  - \( 1.36 \times 10^{-4}/\text{FT} \) based on the shuttle storable system
  - \( 1.51 \times 10^{-4}/\text{FT} \) based on a cryogenic system

- \( M_{me} \) = maneuver engine constant LBT/LBm
  - .0863 based on current shuttle storable system
  - .035 based on advanced cryogenic space engine

- \( M_t \) = maneuver system propellant supply system
  - .119 for storable propellants including pressurization
  - .152 for cryogenic propellants including pressurization and feed
### TABLE III (CONTINUED)

#### 9.0 PRIME POWER

- **Equations**
  
  \[
  m_{\text{pow}} = P_{W_c} \times S_c + P_{W_e} \times \Delta T_{\text{vac}} + P_{W_b} \times m_a
  \]

  - \(S_c\) = total surface control area
  - \(\Delta T_{\text{vac}}\) = total vacuum thrust of gimbaled engines
  - \(m_a\) = mass of avionics

#### 10.0 ELEC. CONV. AND DISTR.

- **Equations**
  
  \[
  m_{\text{el}} = E \times m_l
  \]

  WHERE:
  - \(m_{\text{el}}\) = mass of electrical system
  - \(m_l\) = vehicle landed mass

#### 11.0 HYDRAULICS CONVERSION AND DISTRIBUTION

- **Equations**
  
  \[
  m_h = H_{cs} \times \frac{S_c}{S_c} + H_e \times \frac{S_r}{S_r} \times \Delta T_{\text{vac}}
  \]

  WHERE:
  - \(S_c\) = total surface control area
  - \(\Delta T_{\text{vac}}\) = total vacuum thrust of gimbaled engines

#### 12.0 SURFACE CONTROLS

- **Equations**
  
  \[
  m_{sc} = S_{sc} \times S_c + S_p c
  \]

  WHERE \(S_c\) = surface control area

### EQUATION CONSTANTS

- \(P_{W_c}\) = surface control hydraulic pump power demand
  - = .712
  - = .610 (accumulators for peak demand)

- \(P_{W_e}\) = Engine gimbal power demand
  - = .97 \times 10^{-4}

- \(P_{W_b}\) = battery power demand constant
  - = .405

- \(E\) = electrical conversion and distribution system mass constant
  - = .02
  - = .038 (SHUTTLE)

- \(H_{cs}\) = surface control constant
  - = 2.10 shuttle technology base
  - = 1.23 for a 5000 psi system

- \(H_e\) = engine related gimbal actuation
  - = 3.00 \times 10^{-4} shuttle technology
  - = 1.68 \times 10^{-4} for a 5000 psi system

- \(S_{sc}\) = surface control actuator constant
  - = 3.75 for shuttle technology
**TABLE III (CONTINUED)**

**12.0 SURFACE CONTROLS (CONT'D)**

**13.0 AVIONICS**

\[ m_{av} = M_{av} (m_d)^{1/8} \]

**14.0 ENVIRONMENTAL CONTROL**

\[ m_{env} = E_c (V_p)^{0.75} + E_o + N_c (D) + E_a m_{av} \]

WHERE:
- \( V_p \) = total pressurized volume including wheel wells
- \( N_c \) = number of crew
- \( D \) = days on orbit
- \( m_{env} \) = avionics mass
- \( E_c \) = pressurized volume constant
- \( E_o \) = oxygen supply tank constant
- \( E_a \) = avionics heat load constant

**15.0 PERSONNEL PROVISIONS**

\[ m_{pp} = PP_f + PP_s (N_c) \]

WHERE:
- \( N_c \) = number of crew (1 to 4)
- \( PP_f \) = food waste and water management system, 1 to 4 crew.
- \( PP_s \) = seats and other pilot and crew related items

**EQUATION CONSTANTS**

- \( 3.80 \) = for 5000 psi system
- \( 3.32 \) = for 5000 psi system of advanced materials
- \( S_{pc} \) = miscellaneous systems
  - \( = 200 \)
- \( M_{av} \) = avionics mass constant
  - \( = 1350 \) for current technology
  - \( = 710 \) for 1990 technology
- \( E_c \) = pressurized volume constant
  - \( = 5.85 \)
- \( E_o \) = oxygen supply tank constant
  - \( = 10.9 \)
- \( E_a \) = avionics heat load constant
  - \( = .44 \)
- \( PP_f \) = food waste and water management system, 1 to 4 crew.
  - \( = 0 \) for mission \( \leq 24 \) hours
  - \( = 353 \) for missions \( > 24 \) hours
- \( PP_s \) = seats and other pilot and crew related items
  - \( = 167 \)
TABLE III (CONTINUED)

EQUATIONS

16.0 MARGIN

\[ m_{\text{MAR}} = \text{MAR} (m_d - \Delta m_e m_{\text{eng}}) \]

- \( m_{\text{MAR}} \) = MAR (vehicle dry mass - engine mass)
- \( m_{\text{MAR}} \) = engine mass
- \( N_e \) = number of engines

17.0 PERSONNEL

\[ m_{\text{per}} = P_m + P_p (N_c) \]

where: \( N_c \) = no. of crew (1 to 4)

- \( P_m \) = miscellaneous = 400
- \( P_p \) = personnel = 560

18.0 PAYLOAD ACCOM.

Fixed Input

19.0 CARGO (RETURNED)

Fixed Input

20.0 RESIDUAL FLUIDS

\[ m_{\text{RF}} = R_{\text{RF}} (m_p)^{.79} \]

- \( R_{\text{RF}} \) = .05 (includes main propellant tank pressurization gas)

21.0 OMS AND RCS RESERVES

\[ m_{\text{ORR}} = m_{\text{L}} \left[ \left( \frac{\Delta V_0}{\Delta V_{\text{geo}}} \right) + \left( \frac{\Delta V_R}{\Delta V_{\text{ro}} g} \right) \right] - 2 \]

where: \( \Delta V_0 \) & \( \Delta V_R \) = equivalent OMS and RCS \( \Delta V \)'s for the mission.

- \( R_0 \) = .005
- \( R_r \) = .005
- \( g \) = 32.2 ft/sec²

for specific impulse values, see (23)
### TABLE III: (CONTINUED)

#### SUBSYSTEMS

**22.0 RCS PROPELLANT (ENTRY)**

\[ m_{RCS} = m_e \left( \frac{\Delta V_{re}}{I_{s_e} g} \right)^{-1} \]

where:  
- \( m_e \) = vehicle entry mass  
- \( \Delta V_{re} \) = "Delta Vre" equivalent for entry = 40 ft/sec for shuttle

#### EQUATIONS

**23.0 ACPS CONSUMABLES (RCS + OMS) ON ORBIT**

\[ m_c = m_{DESC} \left( \frac{\Delta V_{ro} + \Delta V_{u}}{I_{s_m} g I_{s_ro}} \right)^{-2} \]

where:  
- \( m_c \) = consumable RCS + OMS  
- \( \Delta V_{ro} \) = equivalent ideal \( \Delta V \) for maneuver  
- \( I_{s_m} \) = average specific impulse of maneuver engine with three restarts  
- \( I_{s_ro} \) = average specific impulse (pulsing)

#### EQUATION CONSTANTS

- \( I_{s_m} \) = effective entry avg. specific impulse; \( = 242 \) sec. for shuttle including degradation for back pressure
- \( g \) = gravity constant

- \( I_{s_m} \) = 313 sec. (storable)
- \( I_{s_m} \) = 440 sec. (cryo)
- \( I_{s_ro} \) = 289 sec. (storable pulsing)
- \( I_{s_ro} \) = 398 sec. (cryo pulsing)

Ascent cargo less return cargo

**24.0 CARGO DISCHARGED**

**25.0 ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS**

\[ m_{ARES} = m_{INJ} \left( \frac{\Delta V_{IDEAL} R_{ar}}{I_{s_e} g} \right)^{-1} + R_{ap} (m_p) \]

- \( \Delta V_{IDEAL} \) = ideal  
- \( I_{s_e} \) = main engine vacuum specific impulse  
- \( g \) = gravity constant

- \( R_{ar} \) = .005 ascent reserves constant  
- \( R_{ap} \) = .004 ascent residuals constant  
- \( m_p \) = ascent propellant

**26.0 INFIGHT LOSSES**

\[ m_{inf} = R_{inf} (m_p) \]

- \( R_{inf} \) = .0043 inflight losses constant
EQUATION CONSTANTS

\[ P_{\text{PL}} = 0.001 \text{ to } 0.002 \text{ for pre-launch losses; } \]

\[ m = \text{total propellant} \]

TABLE III (CONCLUDED)

SUBSYSTEMS

27.0 ASCENT PROPELLANT
# TABLE IV - TANK WEIGHT CONSTANTS

<table>
<thead>
<tr>
<th>SOURCE</th>
<th>PROPELLANT</th>
<th>( K ) ( \text{LB/FT}^3 )</th>
<th>VOL ( \text{ FT}^3 )</th>
<th>ULLAGE PRESSURE</th>
<th>INTEGRAL OR NON-INTEGRAL</th>
<th>MATERIAL</th>
<th>GEOMETRY</th>
<th>COMMENTS</th>
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<tbody>
<tr>
<td>SHUTTLE E/T</td>
<td>LH(_2)</td>
<td>0.5918</td>
<td>53.515</td>
<td>36</td>
<td>INTEGRAL</td>
<td>AL2215</td>
<td><img src="image" alt="Diagram" /></td>
<td>DOES NOT INCLUDE INSULATION</td>
</tr>
<tr>
<td>EN-155*</td>
<td>LH(_2)</td>
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<td>60.37</td>
<td>30</td>
<td>INTEGRAL</td>
<td>INC 718</td>
<td><img src="image" alt="Diagram" /></td>
<td>HONEYCOMB SANDWICH ADDED HONEYCOMB FOR THERMAL PROTECTION</td>
</tr>
<tr>
<td>EN-178*</td>
<td>LH(_2)</td>
<td>0.5760</td>
<td>41.666</td>
<td>20</td>
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<td>AL2219</td>
<td><img src="image" alt="Diagram" /></td>
<td>ISOGRID INCLUDES 4,364 LB INSULATION</td>
</tr>
<tr>
<td>SHUTTLE E/T</td>
<td>LOX</td>
<td>0.6458</td>
<td>19.609</td>
<td>38</td>
<td>INTEGRAL</td>
<td>AL2219</td>
<td><img src="image" alt="Diagram" /></td>
<td>DOES NOT INCLUDE INSULATION</td>
</tr>
<tr>
<td>EN-155*</td>
<td>LOX</td>
<td>0.7660</td>
<td>18.355</td>
<td>20</td>
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<td>AL2219</td>
<td><img src="image" alt="Diagram" /></td>
<td>POLYIMIDE HONEYCOMB FOR INSULATION AND STRUCTURAL STABILIZATION</td>
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</tr>
<tr>
<td>EN-155</td>
<td>JP-5</td>
<td>0.7000</td>
<td>4.819</td>
<td>5</td>
<td>NON INTEGRAL</td>
<td>AL2219</td>
<td><img src="image" alt="Diagram" /></td>
<td>CONVENTIONAL SKIN STIFFENED CONSTRUCTION W/O INSULATION</td>
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<tr>
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<td><img src="image" alt="Diagram" /></td>
<td></td>
</tr>
</tbody>
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*NOTE: EN EVALUATIONS REFER TO LARP IN-HOUSE STUDY WIP**.
FIGURE 1.- SINGLE-STAGE-TO-ORBIT VEHICLE (EM 155). PORTIONS OF TOP SURFACE OF FOREBODY LH₂ TANK ARE EXPOSED. THE BOTTOM SURFACE IS PROTECTED WITH FAIRINGS AND METALLIC THERMAL PROTECTION. TWO TYPES OF PROPULSION SYSTEMS ARE INDICATED; THE LOX/RP SYSTEM BEING OPERATED IN PARALLEL WITH THE LOX/LH₂ SYSTEM EARLY IN THE FLIGHT. SUCH A PROPULSION SYSTEM GIVES AN OVERALL SMALLER VEHICLE EVEN THOUGH THE LOX/RP SYSTEM IS LOWER PERFORMING THAN LOX/LH₂.
FIGURE 2.- EFFECT OF A CHANGE IN THE EXPONENTIAL IN THE TRENDING EQUATION ON PREDICTED PERFORMANCE FOR AN OFF-POINT DESIGN VEHICLE.
FIGURE 3.- VEHICLE TRENDING, EN 155 (29,500 Kg PAYLOAD)
FIGURE 4.- PERFORMANCE MASS FRACTION VS. PROPELLANT LOADING FOR VARIOUS PAYLOADS.
GROWTH FACTOR: $\frac{24.5}{1} = 1$ 
(SLOPE)

(MEAN VALUE FOR MASS RANGE SHOWN)

PAYLOAD BAY: 4.6M X 18.3M 
(15 FT X 60 FT)

MASS RATIO: 8.07

FIGURE 5.- EFFECT OF PAYLOAD MASS ON REQUIRED PROPELLANT LOADING.
FIGURE 6 - EFFECT OF A CHANGE IN FIXED MASSES IN THE TRENDING EQUATION ON PREDICTED VEHICLE PERFORMANCE.

NOTE: ALL CURVES BASED ON A TRENDING EXPONENTIAL OF 1/6.
APPENDIX A

OVERALL VEHICLE TRENDING PROCEDURE
AND SAMPLE CALCULATION
1. Given:
   Gross Vehicle mass = 1,634,998 Kg
   Ascent propellant mass = 1,411,150 Kg
   Required m.r. = 8.1 \( \lambda = .8767 \)
   Actual m.r. = 7.3 \( \lambda = .8631 \)

2. Estimate Fixed Mass:
   - Payload = 29,500
   - Crew Compartment = 2,270
   - Avionics = 2,021
   - Manipulator = 349
   - Personnel and Provisions = 1500
   - Cargo Bay Doors and Cargo Bay Structure = 9720
   Total = 45359 Kg

3. Compute Trending Mass Fraction

\[
\lambda = \frac{\text{ascend propellant}}{\text{gross - fixed mass}} = \frac{1,411,150}{1,634,998 - 45359} = .8878
\]

4. Compute vehicle point design characteristic, \( K_{V_1} \) based on point design propellant loading and trending mass fraction.

\[
K_{V_1} = \left( \frac{m_p}{\lambda} \right)^{1/6} \left( 1 - \lambda \right) \lambda
\]

\[
= \left( \frac{1,411,150}{.8878} \right)^{1/6} \left( 1 - .8878 \right) .8878
\]

\[
= 1.3384
\]
5. Compute \( \lambda \) for the new point design by assuming new ascent propellant loadings

\[
\lambda = \frac{1}{1 + \frac{m_f}{m_{p_2}} + \left[ \frac{1}{m_{p_2}} \right]^{1/6}} \frac{K_v}{1}
\]

The required propellant mass is found to be 2,100,000 Kg compared to 1,411,150 Kg in the original vehicle.

or

\[
\lambda = \frac{1}{1 + \frac{45,359}{2,100,000} + \left[ \frac{1}{2,100,000} \right]^{1/6}} \frac{1.3384}{1}
\]

\[
\lambda = \frac{1}{1.0216 + .1183} = .877
\]

and m.r. = \( \frac{1}{1-.877} = 8.1 \)

For changes in the required payload step 5 above is repeated. For example, assume the payload increment desired is 15,900 Kg. Then:

\[
\lambda = \frac{1}{1 + \frac{45,359 + 15,900}{m_{p_2}} + \left[ \frac{1}{m_{p_2}} \right]^{1/6}} \frac{1.3384}{1}
\]

The required \( \lambda \) is still assumed to be = .8767, or by iterative procedure on computer \( m_{p_2} \) is found to be equal to 2,437,500 Kg.

checking:

\[
\lambda = \frac{1}{1 + \frac{61,259}{2,437,500} + \left[ \frac{1}{2,437,500} \right]^{1/6}} \frac{1.3384}{1}
\]

\[
\lambda = \frac{1}{1.0251 + .1153} = .877
\]
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<th>$K_1$</th>
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<th>$K_2$</th>
<th>$1.33840E+01$</th>
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</table>

- $K_1$ corresponds to $m_r$ in the text.
- $K_2$ corresponds to $K_{y_1}$ in the text.
- $WPI$ corresponds to $m_{p_1}$ in the text.
APPENDIX B

SUBSYSTEM TRENDING WITH EXAMPLES FOR THE SHUTTLE ORBITER AND LARC IN-HOUSE STUDY VEHICLE, EN-155
Figure B-1.- Vehicle test case for SEMP program.
SHUTTLE ORBITER
SUBSYSTEM TRENDING RESULTS

### MASS REPORT

| CK | 1.0 UING GROUP | 15657. | LB | 7102. | KG |
| CK | 2.0 TAIL GROUP | 2311. | LB | 1321. | KG |
| CK | 3.0 BODY GROUP | 41934. | LB | 19033. | KG |
| CK | 4.0 TPS | 24385. | LB | 11058. | KG |
| CK | 5.0 LANDING | 3341. | LB | 3647. | KG |
| CK | 6.0 PROPULSION | 26260. | LB | 12791. | KG |
| CK | 7.0 PROPULSION, RCS | 2814. | LB | 1276. | KG |
| CK | 8.0 PROPULSION, OMS | 2976. | LB | 1358. | KG |
| CK | 9.0 PRIME POWER | 3830. | LB | 1374. | KG |
| CK | 10.0 ELEC CONV AND DISTR | 7319. | LB | 3316. | KG |
| CK | 11.0 HYDRAULICS | 1775. | LB | 805. | KG |
| CK | 12.0 SURFACE CONTROLS | 2615. | LB | 1186. | KG |
| CK | 13.0 ALCIONICS | 6911. | LB | 2927. | KG |
| CK | 14.0 ENVIRONMENTAL CONTROL | 5770. | LB | 2299. | KG |
| CK | 15.0 PERSONNEL PROVISIONS | 1021. | LB | 463. | KG |
| CK | 16.0 MAILING | 767. | LB | 348. | KG |
| **DRY WEIGHT** | **154739.** | LB | **70188.** | KG |
| CK | 17.0 PERSONNEL | 2646. | LB | 1197. | KG |
| CK | 18.0 PAYLOAD ACCOM. | 1608. | LB | 720. | KG |
| CK | 19.0 CARGO (RETURNED) | 32030. | LB | 14515. | KG |
| CK | 20.0 RESIDUAL FLUIDS | 1551. | LB | 704. | KG |
| **LANDED WEIGHT** | **192538.** | LB | **87334.** | KG |

| CK | 21.0 OMS AND RCS RESERVES | 77. | LB | 35. | KG |
| CK | OMS | 57. | LB | 26. | KG |
| CK | RCS | 15. | LB | 9. | KG |
| **ENTRY WEIGHT** | **192815.** | LB | **87369.** | KG |
| CK | 22.0 RCS PROPELLANT (ENTRY) | 828. | LB | 376. | KG |
| **DESCENT WEIGHT** | **193444.** | LB | **87745.** | KG |
| CK | 23.0 ACSPS CONSUMABLES (RCS + OMS) ON ORBIT | 14547. | LB | 6598. | KG |
| CK | RCS | 1664. | LB | 756. | KG |
| CK | OMS | 12883. | LB | 5844. | KG |
| CK | 24.0 CARGO DISCHARGED | 23600. | LB | 14969. | KG |
| **INJECTED WEIGHT** | **240999.** | LB | **109311.** | KG |
| CK | 25.0 ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS | 4454. | LB | 2020. | KG |
| CK | 26.0 IMPULSE LOSSES | 862. | LB | 380. | KG |
| CK | 27.0 ASCENT PROPELLANT | 5230. | LB | 2361. | KG |
| **GROSS LIFT OFF WEIGHT** | **251313.** | LB | **113994.** | KG |

SHUTTLE DATA
### SUBSYSTEM TRENDING RESULTS

<table>
<thead>
<tr>
<th>Component</th>
<th>Dry Weight</th>
<th>Dry Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.0 VING GROUP</td>
<td>67326.2 Lb</td>
<td>31656.7 Kg</td>
</tr>
<tr>
<td>2.0 TAIL GROUP</td>
<td>67326.2 Lb</td>
<td>31656.7 Kg</td>
</tr>
<tr>
<td>3.0 BODY GROUP</td>
<td>161175.2 Lb</td>
<td>73363.5 Kg</td>
</tr>
<tr>
<td>4.0 TPS</td>
<td>6315.2 Lb</td>
<td>2895.2 Kg</td>
</tr>
<tr>
<td>5.0 LANDING</td>
<td>50825.2 Lb</td>
<td>2314.4 Kg</td>
</tr>
<tr>
<td>6.0 PROPELLION</td>
<td>35019.2 Lb</td>
<td>1592.4 Kg</td>
</tr>
<tr>
<td>7.0 PROPELLION, RCS</td>
<td>14333.2 Lb</td>
<td>661.1 Kg</td>
</tr>
<tr>
<td>8.0 PROPELLION, LPS</td>
<td>6586.2 Lb</td>
<td>2483.6 Kg</td>
</tr>
<tr>
<td>9.0 PRIME FUEL</td>
<td>3408.2 Lb</td>
<td>154.2 Kg</td>
</tr>
<tr>
<td>10.0 ELECTRONIC AND DISPLAY</td>
<td>10023.2 Lb</td>
<td>454.5 Kg</td>
</tr>
<tr>
<td>11.0 SPACECRAFT</td>
<td>3537.2 Lb</td>
<td>159.3 Kg</td>
</tr>
<tr>
<td>12.0 SURFACE CONTROLS</td>
<td>3952.2 Lb</td>
<td>179.4 Kg</td>
</tr>
<tr>
<td>13.0 INSTRUMENTS</td>
<td>3952.2 Lb</td>
<td>179.4 Kg</td>
</tr>
<tr>
<td>14.0 ENVIRONMENTAL CONTROL</td>
<td>1621.2 Lb</td>
<td>73.2 Kg</td>
</tr>
<tr>
<td>15.0 PERSONNEL PROVIDER</td>
<td>30701.2 Lb</td>
<td>13926.2 Kg</td>
</tr>
<tr>
<td>16.0 MARGIN</td>
<td>43484.2 Lb</td>
<td>196625.2 Kg</td>
</tr>
<tr>
<td><strong>TOTAL</strong></td>
<td>496893.2 Lb</td>
<td>2273558 Kg</td>
</tr>
</tbody>
</table>

#### Entry Weight

<table>
<thead>
<tr>
<th>Component</th>
<th>Entry Weight</th>
<th>Entry Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>22.0 RCS PROPellant (Entry)</td>
<td>531437.2 Lb</td>
<td>227448.5 Kg</td>
</tr>
<tr>
<td>23.0 ACSs CONSUMABLES (RCS + OMS) ON ORBIT</td>
<td>504245.2 Lb</td>
<td>228591.5 Kg</td>
</tr>
<tr>
<td>24.0 CARGO DISCHARGED</td>
<td>26516.2 Lb</td>
<td>1186.9 Kg</td>
</tr>
<tr>
<td><strong>TOTAL</strong></td>
<td>530961.2 Lb</td>
<td>240840.5 Kg</td>
</tr>
</tbody>
</table>

#### Cross Lift Off Weight

<table>
<thead>
<tr>
<th>Component</th>
<th>Cross Lift Off Weight</th>
</tr>
</thead>
<tbody>
<tr>
<td>25.0 ASCENT RESERVES AND ASCENT PROPellant RESIDUA LS</td>
<td>7653.2 Lb</td>
</tr>
<tr>
<td>26.0 INFLIGHT LOSSES</td>
<td>30381.2 Lb</td>
</tr>
<tr>
<td>27.0 ASCENT PROPellant</td>
<td>3681111.2 Lb</td>
</tr>
</tbody>
</table>

**UTOML SSTO Dual-Fueled Vehicle (EN-155) LRF=201.6 ft.**
APPENDIX C

SUBSYSTEM TRENDING SAMPLE COMPUTER INPUT DATA

AND COMPUTER PROGRAM
4(UPERP,UT(15)),
6(BURAR,UT(16)),
6(BURAR,UT(17)),
7(UPVR,UT(18)),
8(UPVRAR,UT(19)),
9(UPYLDR,UT(20)),
EQUIVALENCE (URF,UT(21)),
11(ULAND,UT(22)),
2(UPVRAR,UT(23)),
3(UDSRA,UT(34)),
4(UCRCS,UT(35)),
5(UNTR2,UT(36)),
6(URCSE,UT(25)),
7(UDESC,UT(26)),
8(UDSR,UT(27)),
9(UCRSC,UT(37)),
EQUIVALENCE (URCSR,UT(38)),
11(UPVR,UT(28)),
2(UTNJ,UT(29)),
3(UPR,UT(30)),
4(UFL,UT(31)),
5(UFLP2,UT(32)),
6(UFL,UT(28)),
EQUIVALENCE (UWING,UT(39)),
1(UTAIL,UT(39)),
2(UBODY,UT(40)),
3(UPM,UT(41)),
4(ULDRM,UT(42)),
5(UPVRPM,UT(43)),
6(URCSC,UT(44)),
7(UCM,UT(45)),
8(URFMPM,UT(46)),
9(WE,UT(47)),
EQUIVALENCE (UHCDM,UT(48)),
11(UHCDM,UT(49)),
2(UGAUM,UT(50)),
3(UGRA,UT(51)),
4(UHFM,UT(52)),
5(UNR,UT(53)),
6(UPVRM,UT(54)),
7(URCM,UT(55)),
8(UPVRM,UT(56)),
9(UPYLM,UT(57)),
EQUIVALENCE (URFM,UT(58)),
1(ULANDM,UT(59)),
2(UPDRLM,UT(60)),
3(UDSRA,UT(61)),
4(UPR,UT(62)),
5(UNTR2,UT(63)),
6(URCSE,UT(64)),
7(UDESC,UT(65)),
8(UCRSC,UT(66)),
9(UCM,UT(67)),
EQUIVALENCE (URCSM,UT(68)),
11(UPVR,UT(69)),
2(UTNJ,UT(70)),
3(URNM,UT(71)),
4(ULAND,UT(72)),
5(UPVRAR,UT(73)),
6(UBODY,UT(74)),
DATA VERDAT/
1: VERDAT/
2: R = 0.01.1, RPH = 0.01.2, EPS1 = 0.01.3
3: R = 0.01.2, EPS2 = 0.01.3
5: E = 0.01.2, RPH = 0.01.3
7: RRA = 0.01.3, AISE = 0.01.2
8: A = 0.01.2, 0/1
VERDAT(18) = LOC(RPH)
VERDAT(22) = LOC(RR)
VERDAT(23) = LOC(EPS1)
VERDAT(15) = LOC(RRA)
VERDAT(17) = LOC(AISE)
VERDAT(1) = LOC(ANE)
NAMELIST/HAM1/RH, RN, EPS1, RNE, EPS2, PC, RNA, ANDT, RPH, R
CA, AISE, ANE
C
INPUT VARIABLES
TRACE 888
C IF AICODE=1.0, WILL CALCULATE WPROP USING NEW EQUATION
C
WRITE(1,952) , TITLE
READ(10,900) TITLE
C WRITE(1,950)
C CALL NAM(1,S,VERDAT)
C DO 10 I = 1,77
READ(5,268) VAR(I), XPAR(I)
VAR(I) = XPAR(I)
C WRITE(1,952) VAR(I), XPAR(I)
C 10 CONTINUE
C 800 FORMAT(A10)
900 FORMAT(F15.0,4X,5A2)
IF (ABS(VENTRY-VENTR2)/VENTR2 .LT. .001) GO TO 999
VENTRY = VENTR2
988 CONTINUE
GO TO 1
999 CONTINUE
CALL ERASE
CALL UPRTN
CALL EXIT
END
SUBROUTINE UPRTN
COMMON/UT/UT(100),TITLE(10)
C WRITE(1,954)
C WRITE(1,970) TITLE
C
P50
1F15.0,2X,'KG')
WRITE(1,914) UT(15), UT(53)
914 FORMAT(16X,'15.0 PERSONNEL PROVISIONS',29X,F15.0,2X,'LB',5X,F15.0,12X,'KG')
WRITE(1,915) UT(16), UT(53)
915 FORMAT(16X,'16.0 MARGIN',43X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
WRITE(1,916) UT(17), UT(54)
916 FORMAT(16X,'DRY WEIGHT',34X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
WRITE(1,917) UT(18), UT(55)
917 FORMAT(16X,'17.0 PERSONNEL',40X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
WRITE(1,918) UT(19), UT(56)
918 FORMAT(16X,'18.0 PAYLOAD ACCOM.',35X,F15.0,2X,'LB',5X,F15.0,2X,1X,'KG')
WRITE(1,919) UT(20), UT(57)
919 FORMAT(16X,'19.0 CARGO (RETURNED)',33X,F15.0,2X,'LB',5X,F15.0,2X,1X,'KG')
WRITE(1,920) UT(21), UT(58)
920 FORMAT(16X,'20.0 RESIDENTIAL FLUIDS',34X,F15.0,2X,'LB',5X,F15.0,2X,1X,'KG',//)
WRITE(1,921) UT(22), UT(59)
921 FORMAT(16X,'LANDED WEIGHT',31X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
WORTOT=UT(34)+UT(35)
WORTA=WORTOT*0.4535924
WRITE(1,922) WORTOT, WORTA
922 FORMAT(16X,'21.0 OMS AND RCS RESERVES',29X,F15.0,2X,'LB',5X,F15.0,12X,'KG')
WRITE(1,933) UT(34), UT(61)
933 FORMAT(32X,'OMS',35X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
WRITE(1,934) UT(35), UT(62)
934 FORMAT(16X,'RCS',35X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
WRITE(1,923) UT(24), UT(63)
923 FORMAT(16X,'ENTRY WEIGHT',32X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
WRITE(1,924) UT(25), UT(64)
924 FORMAT(16X,'22.0 RCS PROPELLANT (ENTRY)',27X,F15.0,2X,'LB',5X,1F15.0,2X,'KG',//)
WRITE(1,925) UT(26), UT(65)
925 FORMAT(16X,'DESCENT WEIGHT',30X,F15.0,2X,'LB',5X,F15.0,2X,'KG',//)
WORTMT=UT(36)+UT(37)
WORTMT=WORTMT*0.4535924
WRITE(1,926) WORTMT, WORTMT
926 FORMAT(16X,'23.0 RCS CONSUMABLES (RCS + OMS) ON ORBIT',12X,F15.0,12X,'LB',5X,F15.0,2X,'KG')
WRITE(1,935) UT(36), UT(66)
935 FORMAT(32X,'RCS',35X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
WRITE(1,936) UT(37), UT(67)
936 FORMAT(32X,'OMS',35X,F15.0,2X,'LB',5X,F15.0,2X,'KG')
WRITE(1,927) UT(28), UT(69)
927 FORMAT(16X,'24.0 CARGO DISCHARGED',33X,F15.0,2X,'LB',5X,F15.0,2X,1X,'KG',//)
WRITE(1,928) UT(29), UT(70)
P60      WRITE(1,928) UT(29),UT(70)
928      FORMAT(26X,'INJECTED WEIGHT',29X,F15.0,2X,'LB',6X,F15.0,2X,'KG'/)
         WRITE(1,929) UT(36), UT(71)
929      FORMAT(16X,'ASCENT RESERVES AND ASCENT PROPELLANT RESIDUALS',
         12X,'LB',5X,F15.0,2X,'KG'/)
         WRITE(1,930) UT(31), UT(72)
930      FORMAT(16X,'INFLIGHT LOSSES',34X,F15.0,2X,'LB',5X,F15.0,2X,
         1'KG'/)
         WRITE(1,931) UT(32), UT(73)
931      FORMAT(16X,'ASCENT PROPELLANT',32X,F15.0,2X,'LB',5X,F15.0,2X,
         1'KG'/)
         WRITE(1,932) UT(33), UT(74)
932      FORMAT(26X,'GROSS LIFT OFF WEIGHT',23X,F15.0,2X,'LB',5X,F15.0,2X,
         1'KG'/)
         WRITE(1,988)
988      FORMAT(/,30X,'UOHM SSTO DUAL-FUELED VEHICLE (EN-15S) LREF=291.5
         1FT.'/)
         RETURN
END

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APPENDIX D

EXAMPLE STRUCTURE (EN 155)
Figure D-1.- Forebody tank on in-house design is designed for Inconel 718 honeycomb. Top portions of tank, where heating rates are lower, were exposed. A thermal protection fairing is used on bottom surfaces.
Figure D-2.- Wing is designed for diffusion bonded Inconel 718. Outer layers of honeycomb are nonstructural and used for thermal protection. Lightweight thin-walled honeycomb stabilized tubes are used for internal bracing.
Figure 7-3.– For the LO₂ tank, honeycomb wall duals as insulation and for tank wall stabilization.
Figure D-4.- Provisions were made in the nonintegral LOX tank installations for axial and radial expansions and contraction.
Honeycomb panels were selected to carry by shear loads and provide thermal protection in this region of moderate heat loads above the wing surface.
APPENDIX E

EXAMPLE VEHICLE CROSS SECTIONS
Figure E-1.- A hypothetical body section for hydrogen containment and peak entry temperatures of 1800°F. External and internal insulations, tank, and structure are separated elements.
Figure E-2.- In the above figure, fluid containment and body loads are combined into one functional element.
Figure E-3.—In the above system, the cryogenic insulation function has been added to the tank/body element by utilizing honeycomb.
Figure E-4. Above cross section represents a hypothetical body element in a noncontainment region such as an aft skirt or intertank adapter.
Figure E-5.—The above crosssection might represent an area in a similar application to that shown in the previous figure. No strain isolator is utilized between RSI and structure since the two are strain compatible.
FIGURE E-6.- Thermal protection unit masses versus peak temperatures based on shuttle trajectories.