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High-Performance Composite Material Airframe Weight and Cost Estimating Relations

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Estimates of the weight reduction potential of using advanced composite materials in high-performance aircraft airframes are presented. A conventional, all-aluminum airframe is established as the reference configuration for comparison purpose, with the alternate use of other homogeneous metals, viz, titanium and beryllium, also considered. Advanced composites discussed in detail include boron/epoxy, graphite/epoxy, and an organic filament/epoxy. Cost factors are estimated in order to keep the significance of the various weight reduction factors in proper perspective. Conventional sheet and stringer construction will eventually be modified to take advantage of the unique characteristics of composite materials in achieving maximum structural efficiency. Therefore, the influence of advanced construction technology on weight and cost factors is also considered. Numerical examples representing applications to specific vehicle airframes are presented, indicating the significance of the estimated airframe weight savings in terms of vehicle performance increases.

I. Introduction

THE principal application of advanced composites, i.e., those composites which incorporate high-modulus filaments such as boron or graphite, has been and continues to be to airframe structures of high performance manned aircraft. In addition, there is a rapidly increasing interest in the possibility of replacing manned aircraft with unmanned vehicles, e.g., drones and remotely piloted vehicles (RPV), in certain very high-risk missions.[1]

The present paper is limited to the consideration of high-performance airframes. High performance will be defined here as the ability to operate at *g*-levels near the upper limits of pilot tolerance (and above in the case of unmanned vehicles). The effects of aerodynamic heating will not be considered, limiting maximum vehicle speeds to about Mach 2 and below. These limits are sufficiently broad to include a very high percentage of the types of airframe mission requirements being considered in the industry at the present time.

The primary purpose of the present paper is to provide the conceptual designer with quantitative estimates of the weight-savings potential available when considering advanced composites for use in either manned or unmanned vehicle airframes. These estimates have been established by comparing various material properties, possible differences in construction, and the unique aspects of individual airframe subcomponents. They have been substantiated wherever possible by comparing the generalized estimates with results actually achieved in the airframe industry for selected subcomponents of specific vehicles. This substantiation is obviously very valuable, but is necessarily somewhat limited since only scattered experimental results are available for use.

No consideration is given here to the possibility of resizing the entire vehicle because of the reduced airframe weight, although this is always a distinct possibility. Rather, it is assumed that the airframe weight savings will be utilized to increase the range (by carrying more fuel) and/or payload of the given aircraft configuration.

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Table 1 Candidate high performance airframe structural materials

Material	Density ρ, lb/in^3	Ultimate tensile strength σ_{ult}, 10^3 psi	Elastic stiffness E, 10^6 psi	Specific tensile strength σ_{ult}/ρ, 10^6 in.	Specific stiffness E/ρ, 10^8 in.	Material cost, $/lb
Aluminum alloy 2024-T6	0.100	57.0	10.5	0.570	1.050	0.60
Magnesium alloy HK 31A	0.065	32.7	6.5	0.505	1.003	5.60
Titanium alloy Ti-6Al-4V	0.160	129.6	15.8	0.810	0.988	12.00
Stainless steel 301 XHSR	0.290	251.0	30.0	0.866	1.034	1.15
Beryllium hot rolled	0.067	70.0	44.0	1.050	6.597	350.00
Boron/epoxy composite unidirectional (50% filament volume)	0.073	190.0	30.0	2.603	4.110	105.00
Graphite/epoxy composite unidirectional Thornel 75S (60% filament volume)	0.058	210.0	45.0	3.621	7.759	275.00
Organic filament/epoxy composite unidirectional PRD-49-III (65% filament volume)	0.050	250.0	11.0	5.000	2.200	50.00
Boron/aluminum composite unidirectional (50% filament volume)	0.097	240.0	32.0	2.474	3.300	200.00

II. Candidate High-Performance Structural Materials

Those candidate materials which are of particular interest for the high-performance vehicles being considered here are presented in Table 1. This list is well representative of the wide variety of new materials now being developed. The indicated strength and stiffness properties of the metals are typical of those given in Ref. 2. The properties of the boron and graphite filament-reinforced composites are typical of those given in Ref. 3. The organic filament/epoxy composite is a new development, and not yet found in materials handbooks. The indicated properties were taken from Ref. 4.

Material cost is dependent upon the form in which the material is purchased. For the metals, the prices indicated in Table 1 are representative of sheet stock, the form most commonly required.

The filament-reinforced, epoxy matrix composite materials are typically supplied in the form of tapes, usually 3 in. in width. The filaments are preimpregnated with the epoxy matrix which is then partially cured (B-staged). This prepreg tape is thus the starting material in the hardware fabrication process. The boron/epoxy prepreg tape material cost of $105/lb given in Table 1 is for RIGIDITE 5505/4 prepreg, 3 in. wide tape.[5] The Thornel 75S graphite filament is one of the highest performance graphite filaments available today in terms of mechanical properties (and it is perhaps also the most expensive[6]).

The price of boron filaments has been fairly well established; it is continuing to decline, but slowly. The price of graphite filaments is much more variable, and decreasing fairly rapidly. Other candidate graphite filaments, having lower strength and/or stiffness properties but also a much lower price (i.e., approximately $75/lb), are also available. The organic filaments, when introduced in early 1971, were quoted at $100/lb. This price soon dropped to $50/lb, and is now only $22/lb.[7] The cost of PRD-49-III/epoxy prepreg is currently about $50/lb.

The price of $200/lb for the boron/aluminum composite material is a current quote for boron-acrylic-aluminum preform sheet.[5] This $200/lb. cost could be cut in half if a market volume demand comparable to that for boron/epoxy were to develop.[5]

All of the materials included in Table 1, except the HK 31A magnesium alloy, the 301 XHSR stainless steel alloy, and the boron/aluminum composite will be included in the weight reduction estimates described in Section III. The magnesium alloy and the steel alloy listed in Table 1 are each among the best available within their respective families of alloys. As indicated, magnesium has a low strength and high cost relative to aluminum. Thus, although magnesium received considerable attention in the 1940's, it is not considered as an important airframe material today, aluminum being used instead.

The stainless steel alloy indicated in Table 1 is among the highest strength steels, having a specific strength comparable to titanium. Most other commonly used steel alloys such as the 4000 series (e.g., 4130 and 4340) have specific strengths which are lower than the aluminum alloy. A particular disadvantage of all the steel alloys, however, is their high density. Thus, for an equivalent weight, a steel part (e.g., a stiffener or cover skin), will be thinner than an aluminum or titanium part. This leads to problems of minimum gage limitations and local wrinkling and buckling instabilities. Because of these disadvantages, combined with the fact that the steels offer no outstanding strength or stiffness advantages over aluminum in room temperature applications, steel alloys are not widely used in airframe structures.

The boron/aluminum composite represents a relatively recent development (within the past several years). As indicated in Table 1, it offers a specific strength over four times that of unreinforced aluminum, and a specific stiffness over three times higher. Graphite/aluminum composites are presently being developed also. However, the higher density of the aluminum matrix as compared to epoxy (about twice as high) makes the aluminum matrix composites less attractive than the epoxy matrix composites on a specific longitudinal mechanical properties basis, although the transverse normal and shear properties are typically higher. In addition, the fabrication of metal matrix composites is much more difficult. Their particular attraction at present is in elevated temperature applications, where the epoxy matrix composites cannot be used. Since temperature (aerodynamic heating) is not an important consideration in the present application, the metal matrix composites will not be considered further.

Table 2 Weight estimating worksheet[a]

Material/construction combination	Weight of structural component: Fuselage	Wing	Vertical stabilizer	Horizontal stabilizer or canard	Engine nacelle(s)	Total weight of structure	Total weight reduction	Airframe total cost ratio
A. Aluminum alloy (base case), conventional sheet & stringer construction							...	1.0
B. Titanium alloy (50% Ti, 50% Al), conventional sheet & stringer construction	(0.85)	(0.90)	(0.90)	(0.90)	(0.85)	( )	( )	1.8
C. Beryllium (50% Be, 50% Al, current construction technology	(0.77)	(0.66)	(0.74)	(0.74)	(0.77)	( )	( )	4.5
D. Boron/epoxy composite (50% B/Ep, 50% Al), current construction technology	(0.77)	(0.81)	(0.81)	(0.81)	(0.82)	( )	( )	2.1
E. Graphite/epoxy composite (50% Gr/Ep, 50% Al), current construction technology	(0.71)	(0.69)	(0.69)	(0.69)	(0.76)	( )	( )	2.9
F. Graphite/epoxy composite (80% Gr/Ep, 20% Gl/Ep), advanced construction technology	(0.46)	(0.42)	(0.47)	(0.47)	(0.56)	( )	( )	1.1
G. Organic filament/epoxy composite (50% Org/Ep, 50% Al), current construction technology	(0.77)	(0.87)	(0.87)	(0.87)	(0.82)	( )	( )	1.5
H. Organic filament/epoxy Composite (80% Org/Ep, 20% Gl/Ep) advanced construction technology	(0.55)	(0.70)	(0.70)	(0.70)	(0.60)	( )	( )	0.7

[a] Weight reduction estimates for various material/construction combinations. Weights given in pounds and, in parentheses, as a fraction of the base case weight.

The specific aluminum alloy, 2024-T6, and titanium alloy, Ti-6Al-4V, included in Table 1 are probably the most commonly used airframe alloys of their respective classes at the present time. Other alloys having somewhat higher strength properties (on the order of 10 % higher) are also being used, e.g., 7075-T6 aluminum and Ti-6Al-6V-2Sn titanium, and some further improvements can be expected; however, it is not anticipated that any major increases will be achieved. Thus, the alloys included in Table 1 not only represent the current state-of-the-art, they are also adequately representative of probable near-term developments. The remaining materials and their potential are discussed in Sec. III.

III. Descriptions of the Various Material/Construction Combinations

Eight different material/construction combinations have been selected for detailed consideration here. Only the high-performance structural materials, as described in Sec. II, are considered since weight reduction is the principal goal. Construction methods include variations of the traditional sheet and stringer design, and advanced (unitized) designs, as described below. These are identified in Table 2 as "conventional sheet and stringer construction," "current construction technology," and "advanced construction technology."

Conventional sheet and stringer construction currently dominate the metal (aluminum and titanium) airframe technology. The substructure is built up of many individual small pieces of bent-up metal sheet and extrusions carefully hand fitted and riveted together. This substructure forms the desired shape and is covered with relatively thin metal sheet to obtain the required aerodynamic smoothness. This large amount of fabrication labor required results in a relatively high cost per pound of finished structure ($20.00 to $30.00 per pound being representative values for all-aluminum construction, varying with the complexity of the specific component[8,9]).

The term "current construction technology" as used in identifying Combinations C, D, E, and G in Table 2 refers to the current practice of utilizing new materials on essentially a direct substitution basis, with only minor changes in construction details. This does not take full advantage of the highly anisotropic nature of the composite materials, but since their specific properties are very attractive, even this relatively inefficient method of utilizing them produces favorable results.

Advanced construction technology, as defined here and assumed for Combinations F and H of Table 2, is still in the development stage. It is likely to be 4 or 5 years before widespread production applications are initiated. It involves the use of unitized construction to reduce the number of subassembly parts, and the almost total use of polymer matrix composite materials. Adhesive bonding is the most weight-efficient method of joining composites, and this process is essentially the same as the basic compositing process involved in combining filaments and matrix into a composite material. Thus it is possible to fabricate a complex structure in essentially one step, eliminating most of the riveted and bolted joints which are typical of a conventional structure. The material, used in thin sheet or tape form, has handling qualities not unlike a piece of heavy canvas. It can be readily cut and contoured and laid down by automated, numerically-controlled, tape-laying machines. An entire component can be assembled in this manner, and then fully cured at only a few hundred degrees Fahrenheit into a strong rigid structure. This simplified fabrication procedure results in major reductions in labor costs. The interested reader is referred to Ref. 10 for additional details concerning these various construction methods.

Some of these construction methods are not appropriate for use with all types of materials and thus the number of practical combinations is limited. The eight combinations included in Table 2 are either existing, well-established designs, or clearly viable, probable future configurations. They are believed to be representative of those available now or likely to be developed within the next 4 or 5 years.

Table 2 is presented in the form of a worksheet, with weight estimates to be entered in the appropriate blanks for each specific vehicle configuration to be analyzed. A specific numerical example is given in Sec. V. The information in parentheses in the left column for each material/construction combination indicates the percentage of each type of material assumed in arriving at the weight and cost estimates. These percentages are established as representing a maximum practical utilization of the specific high-performance material being considered, taking into account both performance and cost factors. The remainder of the structure is assumed to be constructed of the lower cost (but lower performance) material indicated, viz, aluminum or a glass/epoxy composite.

Combination A, the all-aluminum conventional sheet and stringer construction, represents the base case configuration. Most existing weight estimating relationships, being empirical or semi-empirical in nature, are based on this material/construction combination. Thus, if the weights of the individual airframe components for the base case are not known for the vehicle configuration to be analyzed, they can be estimated using such relationships. One representative set of relationships is described in Sec. IV, and utilized in the numerical example of Sec. V.

The weights of these same components for material/construction combinations B through H are expressed as a fraction of these base case weights in Table 2 (the numbers in parentheses). The following expressions have been used in arriving at these component weight fraction estimates:

$$WF = \text{component weight fraction} = R\chi_{adv} + 1.0\chi_{Al} \tag{1}$$

where R = ratio of ρ/σ or ρ/E for a given advanced material to the corresponding ρ/σ or ρ/E for aluminum (ρ/σ or ρ/E being used corresponding to whether the particular component is strength critical or stiffness critical), i.e.,

$$R = \frac{(\rho/\sigma)_{adv}}{(\rho/\sigma)_A} = \frac{(\sigma/\rho)_{Al}}{(\sigma/\rho)_{adv}} \text{ (strength critical)}$$

$$R_E = \frac{(\rho/E)_{adv}}{(\rho/E)_A} = \frac{(E/\rho)_{Al}}{(E/\rho)_{adv}} \text{ (stiffness critical)}$$

χ_{adv} = fraction of the original all-aluminum component weight which is to be replaced by the advanced material (values indicated in parentheses in the left column of Table 2). χ_{Al} = fraction of the original all-aluminum component weight which is to be retained.

The ratios ρ/σ or ρ/E for the (isotropic) metals (aluminum, titanium, and beryllium) can be taken directly from Table 1. However, as stated in Table 1, the indicated properties of the composite materials are for a unidirectional laminate. In most applications it is necessary to utilize a laminate of multiple lamina orientations in order to carry the multiaxial design loads. Because the highly anisotropic unidirectional lamina have relatively low transverse and shear strength and stiffness properties, a multilamina orientation composite will have lower composite properties in any given direction than those given in Table 1. It will be assumed here that 50% of the filament reinforcement is oriented in the direction of principal loading, the remaining 50% being oriented as required to provide an adequate balance of strength and/or stiffness along axes of secondary loadings. This is consistent with current laminate construction, as discussed for example in Ref. 11 and 12. Other factors which also influence WF for each specific type of advanced material and/or component include, as examples, the use of sandwich construction in selected areas in place of sheet-stringer design, the influence of minimum gage limitations, temperature influences (a 5% weight penalty when using the epoxy matrix composites in the engine nacelle structure is assumed, for example), and joining penalties (a 5% weight penalty is assumed where extensive joining of composites to metal is required[13]).

The airframe total cost ratio (CR) estimates of Table 2 have been established as follows:

$$CR = \frac{R_{eff}\,\chi_{adv}T_{adv} + \chi_{Al}T_{Al}}{T_A} \tag{2}$$

where χ_{adv} and χ_{Al} are as defined in Eq. (1), and T_{adv} = total per lb cost of a typical structural subassembly if constructed entirely of the advanced material = M_{adv} + F_{adv}; M_{adv} = per lb material cost of the advanced material (expressed as a fraction of F_{Al}) = [per lb material cost of advanced material/per lb fabrication cost of aluminum] × F_{Al}; F_{adv} = per lb fabrication cost of the advanced material (expressed as a fraction of F_{Al}) = [per lb fabrication cost of advanced material/per lb fabrication cost of aluminum] × F_{Al}; T_{Al} = total per lb cost of a typical structural subassembly if constructed entirely of aluminum = M_{Al} + F_{Al}; M_{Al} = per lb material cost of aluminum (expressed as a fraction of F_{Al}) = [per lb material cost of aluminum/per lb fabrication cost of aluminum × F_{Al}; F_{Al} = per lb fabrication cost of aluminum = \$25/lb (assumed on the basis of data contained in Refs 8 and 9). Fabrication costs are influenced, for example, by scrap loss factors, and machining and forming costs.

The term R_{eff} in Eq. (2) represents an effective or average value for the total airframe, analogous to the R of Eq. (1) which was applicable to each individual component. It is evaluated by formulating an equation of the same form as Eq. (1), with R_{eff} replacing R and $(WF)_{avg}$ replacing WF, and then solving for R_{eff} i.e.,

$$R_{eff} = [(WF)_{avg} - \chi_{Al}]/\chi_{adv} \tag{3}$$

where $(WF)_{avg}$ represents an average weight fraction for the total airframe, based on the calculated values for each component using Eq. (1), specific values of which are given in Table 2.

The costs of most of the newer materials are currently subject to constant downward adjustment as production methods are improved and demand increases. This is particularly true of the composite materials. Also, fabrication methods using the newer materials—titanium, beryllium, and the composites—are likewise not yet fully developed. Improved methods of cutting, forming, drilling, welding, etc. are continually being introduced. Since the cost of the all-aluminum base case configuration is well-established and not likely to change significantly with time, while the costs of the other configurations are decreasing, the indicated cost ratios will decrease also. Thus, these estimates, which reflect current industry experience, will tend to become increasingly conservative (overestimate the actual cost ratios with time). As new materials become available, material costs change, and/or new fabrication methods are introduced; Table 2 can be readily updated using these relationships.

The cost ratio estimate derived here [Eq. (2)] is for the total airframe. This could have been extended to the estimation of cost ratios for individual components as well, just as was done in the weight estimate of Eq. (1); however, as stated in the Introduction, cost aspects of utilizing advanced materials in place of aluminum are less well established than factors influencing weight, and will be much more subject to downward revision as material production methods and fabrication technology are continually improved. Since the main thrust of the present paper is the weight reduction potential of high-performance airframes, cost ratios are presented primarily for the purpose of providing a proper perspective for the weight reduction estimates.

Obviously fabrication costs, in addition to varying from component to component (i.e., fuselage, wing, stabilizer, etc.), will also vary with the size of the production run. However, when comparing the use of two different materials but the same method of construction (as is the case in evaluating Combinations B, C, D, E, and G of Table 2 relative to Combination A), then fabrication cost economics of increasing the size of the production run will benefit each material combination approximately equally. Thus, the airframe total cost ratio will be influenced very little.

IV. Aluminum Base Case Component Weight Estimating Relations

Frequently the weights of the individual components of the aluminum base case airframe are not known. This is particularly true for those vehicles which are still in the conceptual design phase. Often only a general configuration and the basic performance requirements have been established. Even with only this information available, however, it is usually possible to estimate the weights of the individual components, and hence the total airframe weight. A number of empirical relationships have been established to do this, most being based upon the extensive data available for manned aircraft utilizing conventional aluminum sheet and stringer construction.

One such set of relationships as suggested by Kirkwood[14] will be summarized here as being typical.

Fuselage

The fuselage weight, in pounds, is estimated as

$$W_f = 5\,(q \times 10^{-2})^{0.25} f^{0.7} D_F^{2.3} (n_u / 6.65)^{0.5} \tag{4}$$

where q = dynamic pressure = $(1/2)\rho v^2$, lb/ft²; ρ = mass density of air at operating conditions, slugs/ft³; v = vehicle velocity, fps; f = fuselage fineness ratio = fuselage length/diam; D_f = fuselage diameter, ft; n_u = structural design load factor (maximum g-loading).

Wing

The wing weight, in pounds, is estimated as

$$W_W = 0.009\left[\frac{W_T n_u S A^{1.5}(1.1 + 0.5\lambda) f_R f_Q^{1.5}}{f_T \cos^{1.5}\Omega}\right]^{0.656} \tag{5}$$

where

$$f_Q = 1 + 31.2[(0.001q)^2/n_u^3]$$

W_T = vehicle gross takeoff weight, lb; S = wing area, ft²; A = wing aspect ratio = total wing span/mean chord length; λ = wing taper ratio = chord length at wing tip/chord length at wing root extended to fuselage centerline; f_R = bending relief factor, dependent upon amount of fuel carried in wing (for wings having little relieving load, use a value of 1.7); Ω = wing sweep angle; and f_T = function of wing thickness ratio (wing root thickness/chord length at wing root), as shown in Table 3. The quantities n_u and q are as defined for the fuselage weight estimate.

Vertical Stabilizer, and Horizontal Stabilizer or Canard

The weight of these components is usually assumed to be a fraction of the wing weight since their sizes are necessarily related. In Kirkwood's work, in which designs utilizing a canard were not explicitly considered, the vertical and horizontal stabilizers together were assumed to weigh 25% that of the wing.

For designs incorporating a horizontal and a vertical stabilizer and no canard, the weight of the vertical stabilizer will be assumed to be 10% that of the wing, the horizontal stabilizer 15%. For delta wing-with-canard designs, i.e., no horizontal stabilizer, the vertical stabilizer will be assumed to weigh 15% that of the wing, and the canard also 15%.

Engine Nacelles

As suggested by Kirkwood, each engine nacelle will be assumed to weigh 7 ½% as much as the engine it encloses.

V. Numerical Example

To demonstrate the use of the method outlined in the previous sections for estimating the weight reduction potential of a vehicle airframe, a specific configuration will be analyzed. The Teledyne/Ryan Aeronautical BQM-34A target drone is an example of a well-designed small vehicle; and an actual component weight breakdown of its aluminum airframe is available. A three-view sketch of the BQM-34A target drone is shown in Fig. 1; its basic characteristics are indicated in Table 4.[15]

Since the weights of the individual structural components are known, it would not be necessary to utilize the weight estimating relations of Sec. IV. However, it is interesting to compute the component weights anyway, to verify the suitability of the estimating relations for an established condition.

The manufacturer utilized a 1.33 safety factor on the 5-g design loading indicated in Table 4. Thus, n_u = 6.65 in Eqs. (4) through (6). For a sea level cruise speed of 550

Table 3 Values of wing thickness ratio function f_T

Wing thickness ratio	f_T
0.03	3.6
0.06	6.0
0.13	12.1
0.18	14.3

Table 4 BQM-34A Target drone characteristics

Performance	
Sea-level cruise speed	550 knots
Maximum range	905 n mi
Design loading	5 g
Dimensions	
Fuselage length	22.97 ft
diameter	28.12 in.
Wing span	12.9 ft
area	36 ft²
aspect ratio	4.63
thickness ratio	0.09
taper ratio	1.0
sweep angle	45 deg
root thickness ratio	0.09
Weights (lb)	
Gross takeoff	2500
Payload	347
Engine	347
Fuel (JP-4)	650
Recovery system	117
Guidance and control systems	73
Structure	601

Table 5 BQM-34A Target vehicle structural component weights, lb

Component	Estimated	Actual
Fuselage	305	220
Wing	196	169
Vertical stabilizer	19.6	29
Horizontal stabilizer	29.4	62
Nacelle group	26	121
Totals	576	601

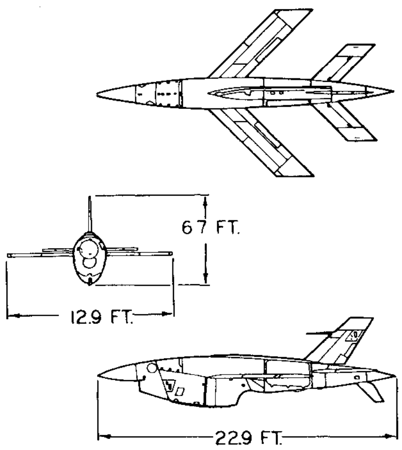

Fig. 1 BQM-34A target drone.

knots, i.e., 882 fps, the dynamic pressure q in Eqs. (4) and (6) is equal to 924 lb/ft^2, assuming an air mass density ρ of 0.002378 slugs/ft^3 for standard sea-level conditions. The fuselage fineness ratio f is equal to 22.97 (12)/28.12 = 9.8. A bending relief factor f_R of 1.7 is assumed, no fuel being carried in the wing. A value of f_T = 8.8, corresponding to the stated wing thickness ratio of 0.09, is obtained by interpolation of the tabulated values given in Table 3. Thus, all of the terms contained in Eqs. (4–6) are defined, and the component weights can be computed as outlined in Sec. IV. The results are given in Table 5, along with the actual component weights stated previously in Table 4.

The total estimated structural weight of 576 lb is only 25 lb less than the actual total weight, a difference of only about 4%. The weight estimates for the individual components do not agree as well. However, this can be accounted for, at least in part. For example, as Fig. 1 indicates, the BQM-34A has an engine nacelle which is integral with the fuselage. Thus, the amount of structure which is attributed to each component, fuselage, or nacelle, is not distinctly defined. That the sum of the estimated weights of these two components, 331 lb, agrees well with the sum of the actual weights, 341 lb, is perhaps more significant. The estimated weight of the horizontal stabilizer is less than one half the actual weight, which may be due to the way in which the weight of the attachments was allocated between fuselage and horizontal stabilizer in the actual weight breakdown. This discrepancy also exists for the vertical stabilizer, although to a lesser degree. Nonetheless, it is clear that the weight estimating relations, even though being very general in nature and relatively simple in form, do give a satisfactory estimate of the structural weight for this specific example. Comparable results have been obtained for other applications also[1,10] providing a high degree of confidence in the general suitability of these approximations.

We will return now to the problem of principal interest, a comparison of the weight estimates for the various material/construction combinations being considered. The actual weights of the structural components, as given in both Tables 4 and 5, can be entered as Combination A, the base case, in Table 2. This is indicated in Table 6, which is a reproduction of the Table 2 worksheet. Weight estimates for Combinations B–H can then be computed directly, as also indicated.

Combinations A, B, D, and E represent well-established configurations, already proven in production applications. Considering the modest difference in estimated cost between the titanium structure and the two composite material structures, the latter two are obviously very attractive, boron/epoxy offering a weight reduction of about 1.5 times that for titanium, and graphite/epoxy over twice as much. Current trends in the airframe industry support this conclusion. Titanium is approaching its maximum utilization as a substitute for aluminum in current airframe designs such as those for the F-14 and F-15 fighter aircraft. Boron/epoxy, the first of the high-performance composites to be extensively developed, is beginning to replace titanium in many applications, including certain components of the F-14 and F-15 aircraft. Graphite/epoxy, a more recent development, is rapidly establishing itself as an improvement over boron/epoxy, having a lower potential cost and better fabrication characteristics.

Beryllium is not a candidate for current applications because of its brittleness characteristics, even though it would otherwise offer a weight reduction equivalent to the current graphite/epoxy composite.

Table 6 Numerical example, Teledyne/Ryan BQM-34A target drone aircraft[a]

	Material/construction combination	Weight of structural component: Fuselage	Wing	Vertical stabilizer	Horizontal stabilizer or canard	Engine nacelle(s)	Total weight of structure	Total weight reduction	Airframe total cost ratio
A.	Aluminum alloy (base case) conventional sheet & stringer construction	220	169	29	62	121	601	...	1.0
B.	Titanium alloy (50% Ti, 50% Al) conventional sheet & stronger construction	187 (0.85)	152 (0.90)	26 (0.90)	56 (0.90)	103 (0.85)	524 (0.87)	77 (0.13)	1.8
C.	Beryllium (50% Be, 50% Al) current construction technology	169 (0.77)	112 (0.66)	22 (0.74)	46 (0.74)	93 (0.77)	442 (0.74)	159 (0.26)	4.5
D.	Boron/Epoxy composite (50% B/Ep, 50% Al) current construction technology	169 (0.77)	137 (0.81)	24 (0.81)	50 (0.81)	99 (0.82)	479 (0.80)	122 (0.20)	2.1
E.	Graphite/Epoxy composite (50% Gr/Ep, 50% Al) current construction tech	156 (0.71)	117 (0.69)	20 (0.69)	43 (0.69)	92 (0.76)	428 (0.71)	173 (0.29)	2.9
F.	Graphite/Epoxy composite (80% Gr/Ep, 20% Gl/Ep) advanced construction tech	101 (0.46)	71 (0.42)	14 (0.47)	29 (0.47)	68 (0.56)	283 (0.47)	318 (0.53)	1.1
G.	Organic Filament/Epoxy composite (50% Org/Ep, 50% Al) urrent cconstruction tech	169 (0.77)	147 (0.87)	25 (0.87)	54 (0.87)	99 (0.82)	494 (0.82)	107 (0.18)	1.5
H.	Organic Filament/Epoxy composite (80% Org Ep, 20% Gl/Ep) advanced constr tech	121 (0.55)	118 (0.70)	20 (0.70)	43 (0.70)	73 (0.60)	375 (0.62)	226 (0.38)	0.7

[a] Weight reduction estimates for various material/construction combinations. Weights given in pounds and, in parentheses, as a fraction of the base case weight.

Combination G, the organic filament/epoxy composite, has been available only a short time. However, because of the extensive experience now available in working with composites in general, no insurmountable difficulties are likely to be encountered in its use. The estimated weight reduction potential for the particular organic filament indicated in Table 6, i.e., DuPont PRD-49-III, is not as great as for the particular graphite filament assumed, i.e., Union Carbide Thornel 75S. However, this organic filament is only the first of what probably will eventually be a whole family of organic filaments, with others having higher stiffness properties. On the other hand, the Thornel 75S filament is among the latest in a series of graphite filaments, and probably is closer (percentagewise) to the ultimate stiffness potential of graphite filaments than PRD-49-III is to the ultimate stiffness potential of organic filaments.

Configurations F and H assume an advanced construction technology which will not be widely available for production applications for perhaps 4 or 5 years. The estimates do indicate the extent of weight reduction clearly forseeable within the relatively near future, however. Expected additional improvements in both graphite and organic filaments during this time period will undoubtedly push the maximum weight reductions actually achieved to beyond the estimates given here.

VI. Summary

The principal results of this study are summarized in Table 2. This table is organized in worksheet form so that it can be used directly in evaluating the weight-savings potential of utilizing various advanced materials in any given vehicle airframe configuration. There is a considerable amount of background information supporting the results of Table 2, some of which is presented in Secs. II and III. The interested reader is referred to Refs. 10 and 16–20 for a more complete discussion. However, the representative set of component weight estimating relations given in Sec. IV make this paper a self-contained package.

This paper is intended primarily for use by systems analysts and preliminary designers, to provide a ready method of estimating airframe weight reduction and cost factors for various advanced material systems and methods of construction. As a minimum, only Table 2 is required for this purpose. However, this paper should also be informative to those not familiar with the use and potential of advanced composite materials. An attempt has been made to clearly present a general overview of the technology, the current state-of-the-art, and materials cost and performance trends.

Weight and cost estimating relations are expressed in equation form so that as new materials are introduced or costs change significantly because of some new development, the factors presented in Table 2 can be readily updated or expanded. That is, the information included here should provide an ample foundation upon which others can develop additional material/construction configurations to satisfy their own special needs.

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