VOL. 17, NO. 9 J. AIRCRAFT SEPTEMBER 1980

ARTICLE NO. 79-0719R
Structures Design Lecture

C80-109

Introduction of New SDM Technology into Production Systems

00029
00030

R. N. Hadcock
Grumman Aerospace Corporation, Bethpage, N.Y.

Foreword

SINCE this is the 20th anniversary SDM Lecture, it would seem appropriate to re-examine past SMD technology to establish a base of reference. This is provided by the structures and materials of WW II fighter aircraft.

I. Introduction

The introduction of new structures, structural dynamics, and materials (SDM) technology into production systems requires the concurrent satisfaction of at least the three preconditions shown diagrammatically in Fig. 1.

First and foremost, there must be a well-defined need for the new technology. This need may be in the form of the inability of existing technology to provide the desired level of performance. It may be in the form of unacceptably high procurement or maintenance costs. It may even be the result of a basic structural deficiency in the design or the material.

Second, the benefits in terms of performance improvement, cost reduction, and risk must be well-defined. This implies that the new technology has achieved a reasonably high level of maturity and availability, and that sufficient hard data are available to perform cost-vs-performance-vs-risk tradeoffs. Risk will almost always be increased by the introduction of new technology. As will be discussed later, it takes between 10 and 30 years for new technology to reach full maturity.

The third precondition for introducing new SDM technology is opportunity. The introduction of at least some new technology into new systems has, in the past, been a very effective way of improving performance while simultaneously advancing technology.

The greatest performance benefits are realized by taking advantage of the interaction of new technologies during preliminary design. Performance can be increased, or vehicle size and associated procurement and operating costs can be reduced, by a significantly greater degree than is possible when individual new technologies are introduced into an existing system. With the reduction of new programs, opportunity is rapidly disappearing for the introduction of new technology.

II. Future Requirements

The next generation of military aircraft, commercial aircraft, helicopters, and spacecraft are very different. However, all will benefit from procurement or operating costs reduction, weight reduction, and, in specific cases, aeroelastic tailoring. In the past, structural weight reductions have been transformed into improved performance, but often at increased cost. Since design is an interactive and evolving process, it is necessary to establish good definitions of possible improvements in SDM technology.

III. Structural Efficiency Improvements

It is difficult to obtain a quantitative measure of improvement in structural efficiency by comparing the structure or airframe weights of different aircraft types over previous years since their components have been designed for many different conditions.

At first sight, the structure weight of fighter aircraft would appear to be a possible guide since most fighter aircraft have been designed to similar ultimate load factor conditions. There is also a large data base available for U.S. Air Force fighters (see Table 1). Over the past 50 years, fighter aircraft have grown significantly in weight and size, as shown in Fig. 2. (The silhouettes are drawn to the same scale.) Size increases should afford more opportunity for design improvement and structural tailoring. It might, therefore, be expected that increased size, in conjunction with SDM technology advances, would reduce the ratios of both the airframe weight [Aircraft Manufacturers Planning Report (AMPR) weight] and the structure weight to design takeoff gross weight (TOGW). As can be seen from Fig. 3, this is not the case. The AMPR weight of WW II fighters was about 40% of design TOGW; this is still true for most current fighter aircraft. The structure weight is still about 33% of the TOGW. It should be noted that both the AMPR weights and TOGW of Navy fighter aircraft are about 5% higher than their Air Force counterparts due to the structural penalties imposed by more severe landing conditions and the catapult and wing fold requirements.

Richard Hadcock (AF) joined Grumman in 1965 as a structural development engineer. He worked on advanced composites development from 1965-70 as structural-analysis and structural-mechanics project engineer and from 1970-75 as advanced-composites program manager. In December 1975 he was appointed director of advanced development, and since then he has been responsible for some 150 government-funded development contracts and IR&D programs covering a broad range of disciplines from design to flight testing to manufacture. Hadcock served as a member of the design-criteria panel on the Air Force/NASA Planning Study "Composites Recast," contributed to the NASA space vehicle design criteria for bucking strength of structural plates, and participated in the NASA monograph advisory planning group for design criteria on composites. He recently chaired an AIA study of critical structures, materials, and process technologies. Before joining Grumman, he worked in the Structures Depts. of the de Havilland Aircraft Co. and Hawker Siddeley Aviation from 1950-65.

Presented as Paper 79-0719 at the AIAA/ASME/ASCE/AHS 20th Structures, Structural Dynamics, and Materials Conference, St. Louis, Mo., April 4-6, 1979; submitted July 5, 1979; revision received Jan. 18, 1980.

Index categories:Structural Design (including Loads); Structural Materials.

Since the small variation in the ratio of structure weight to design TOGW does not reflect the many SDM technology advances made during the past 50 years, the data must be examined in greater depth. During this period, fighter aircraft have become faster, heavier, larger, and more expensive. Overall performance capability has increased by orders of magnitude. Maximum speeds have risen from 234 mph at 7500 ft for the Boeing P-26A, which entered service in 1934, to 1650 mph (M 2.5) at 36,000 ft for such aircraft as the McDonnell Douglas F-15A. A WW II fighter aircraft, the North American P-51B Mustang, that has been used as the baseline for this study, had a maximum speed of 440 mph at 30,000 ft when it entered service in 1943.

Maximum speed of operational fighters almost doubled in the 9 years between 1934 and 1943. By 1959, with the introduction of the F-106A having a maximum speed of 1525 mph at 40,000 ft (M 2.31), maximum speeds had increased by a factor of 6.5 in a period of only 25 years. As shown in Fig. 4, the maximum speed of fighter aircraft leveled off after the F-106A, mainly as a result of SDM complexities and costs associated with structural heating above 350°F and the effects of high dynamic pressures.

IV. Wing Structures

Greater speed requirements have resulted in wings with thinner sections to reduce supersonic drag. Piston-engined fighters such as the P-51 had a thickness-to-chord ratio (t/c) of 15-16%; current fighters such as the F-15 and F-16 have t/c ratios of 4-6%, as shown in Fig. 5.

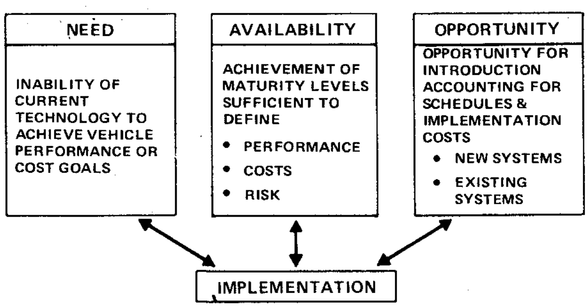

Fig. 1 Preconditions for introduction of new SDM technology.

Since fighter wings are primarily designed by maneuver conditions, it can be shown that the weight of the wing structure required to carry bending loads is proportional to:

$$\frac{N_Z W_{t0} A b \sec^2\Delta}{t/c} \times \frac{\rho}{F} \tag{1}$$

where

N_Z = ultimate load factor
W_{t0} = design TOGW
A = wing aspect ratio
b = wing span
Δ = sweep angle of wing flexural axis (taken at ¼ chord)
t/c = thickness-to-chord ratio at wing root
ρ = density of cover material
F = working stress of cover material

The bending structure (covers plus spar caps) typically accounts for 43-55% of the wing structural group (torque box plus control surfaces).

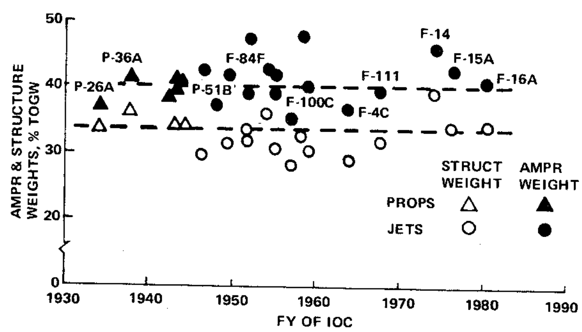

Fig. 3 Airframe (AMPR) and structure weights.

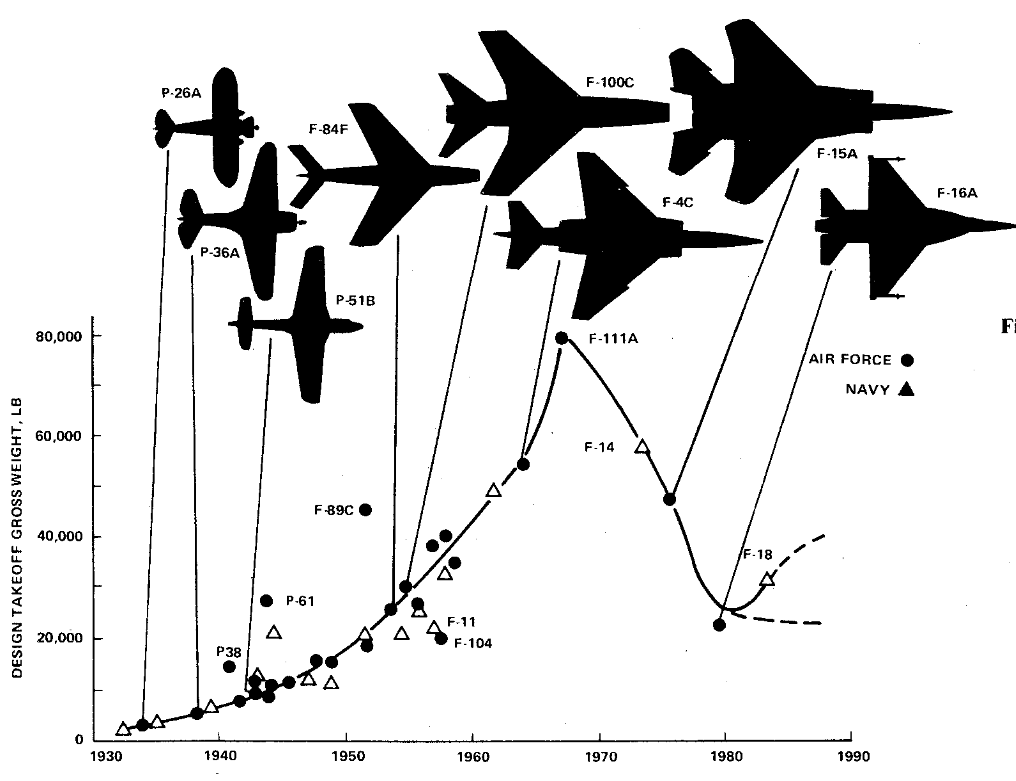

Fig. 2 Fighter takeoff gross weights.

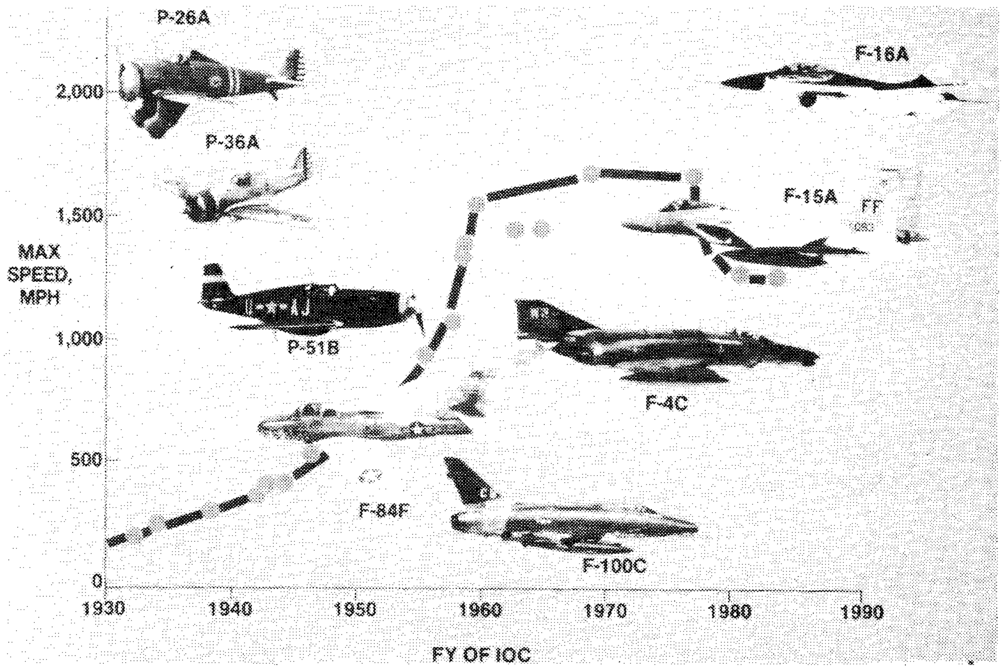

Fig. 4 U.S. Air Force fighters (1930-1980).

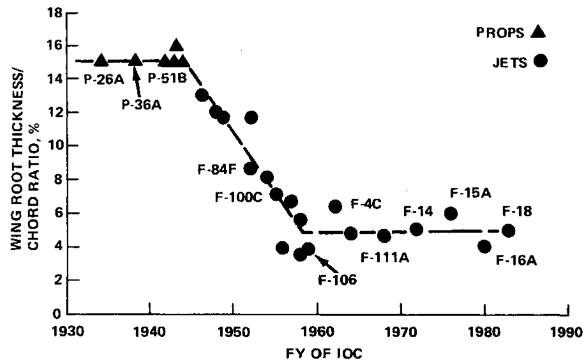

Fig. 5 Wing root thickness/chord ratios.

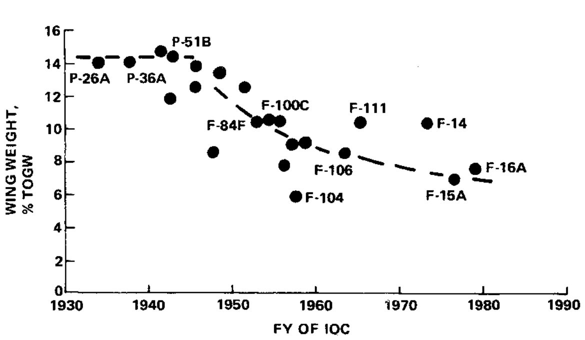

Fig. 6 Fighter wing weights.

Table 1 U.S. Air Force fighter data base (as used for this paper)

1930-39	1940-49	1950-59	1960-69	1970-80
P-26A	P-40N	F-86F	F-105D	F-15A
P-36A	P-47C	F-86D	F-105F	F-16A
	P-51B	F-89C	F-111A	
	P-61B	F-84D	F-4C	
	P-80C	F-100C	F-4D	
	F-84B	F-100D	F-4E	
	F-86A	F-101B		
		F-102A		
		F-104A		
		F-106A		

Maximum speed indirectly influences wing weight since it dictates the maximum acceptable t/c, sweep angle, and maximum structural temperature. Thus speed, in conjunction with TOGW, aspect ratio, and span, has the greatest influence on wing weight changes. Other design considerations such as wing configuration and construction (wing-to-fuselage joint design, exposed wing area, variable sweep, wing fold, landing gear location, taper ratio, integral fuel tankage, high lift devices, and external wing store capability) also affect wing weight, but their influence is less significant than the terms in Eq. (1).

The ultimate load factor has not varied significantly over the past 50 years; nor has the ratio of material density to working stress. The P-51 was designed to an N_Z of 12. Current fighters are generally designed to an N_Z of 11, although there are some exceptions (the F-105B was designed to an N_Z of 13). Material properties have improved over the years (as will be discussed later), but tensile strength increases have generally been offset by reduced allowables to provide a longer service life or sustain increased structural temperatures. Higher cover loadings have permitted the use of higher local working stresses in compression, but these have often been offset by overall stress limitations to preclude flutter or reductions due to higher structural temperatures.

Since the t/c ratio was so drastically reduced between 1945 and 1960 (Fig. 5), it might be expected that the weight of the wing structural group relative to TOGW would increase dramatically. This is not so. As can be seen from Fig. 6, the ratio of wing weight to TOGW has decreased significantly. The wing weight of the F-15A is only 7% TOGW, whereas the wing weight of the P-51 was 14.5% TOGW. The P-51B and F-15A wing structures (with applicable weight and geometrical data) are compared in Fig. 7.

When actual wing weights are normalized to the P-51B baseline span using Eq. (1), weight reductions are even greater. The normalized or effective weights of current fighter wings are about 3% of TOGW, compared to 14.5% of TOGW for the P-51B wing. The normalized weight of fighter wings relative to the P-51B wing is depicted in Fig. 8. Figure 9 shows the transformation of the data in Fig. 8 into the form of relative wing structural efficiencies. As can be seen, advances made during the past 50 years are significant, even recognizing that some of the data may be unconservative since Eq. (1) was utilized to normalize the total wing group weights.

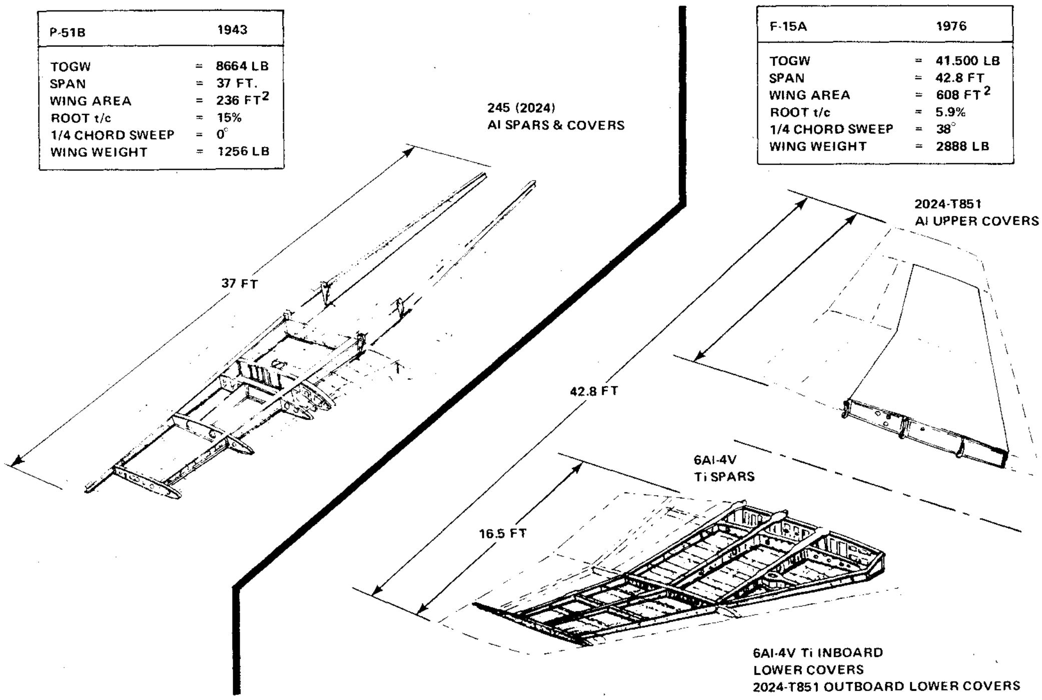

Fig. 7 Wing structural configurations.

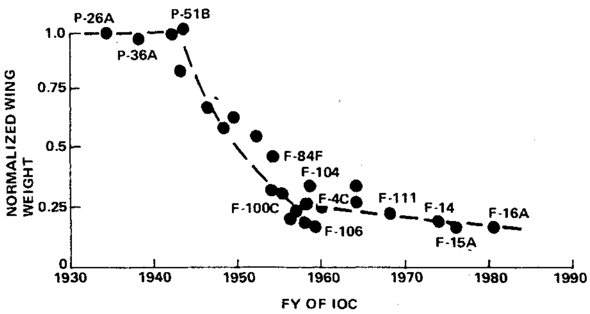

Fig. 8 Normalized wing weights.

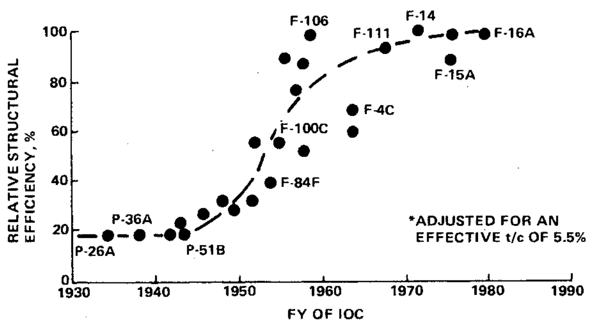

Fig. 9 Relative structural efficiency of fighter aircraft.

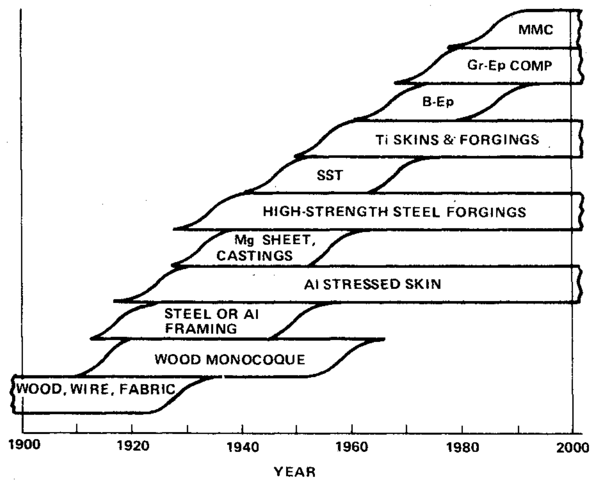

Fig. 10 Aircraft materials application.

Structural efficiency has certainly improved; today's fighter wings could only have been made in almost solid aluminum or steel using the manufacturing and design technology available in 1940.

Performance increases resulting from improved aerodynamics and propulsion technology could not have been obtained without these improvements in structural efficiency. This situation has not been recognized throughout the aerospace community.

It should be noted that the lower value shown in Fig. 9 for the relative structural efficiency of the F-15 wing is based on the actual root t/c ratio of 5.9%. The wing thickness rapidly decreases to a tip t/c ratio of only 3%. The upper value is based on an effective wing root t/c ratio of 5.5%, which is considered more representative.

V. Material Improvements

Since the beginning of the century many different materials have been used for airframe construction. The Wright Brothers used wood, wire, and fabric for the Wright Flyer when they made the first powered aircraft flight 77 years ago. These materials had been successfully used to make gliders and kites during the latter part of the nineteenth century, and were still being used at the start of WW II for training aircraft. Between 1910 and 1925, steel tubes began to replace wood. The first all-metal monoplane, having wings made from pure aluminum, was designed by Hans Reissner in Germany and flew in 1912. This was followed by the various Junkers and Dornier designs produced between 1916 and 1917. It was not until the early 1930's that aluminum-stressed skin became an accepted form of construction for both military and commercial aircraft. Magnesium airframes came and went during the 1930's and 1940's, as did stainless steel during the 1940's and 1950's. By the middle 1950's titanium had become an accepted material, especially for high-

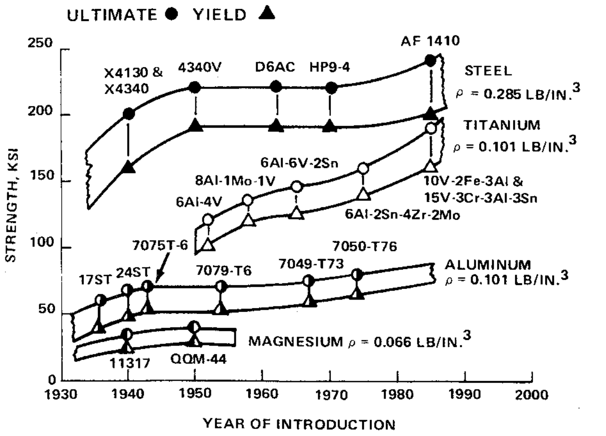

Fig. 11 Tensile strength of materials.

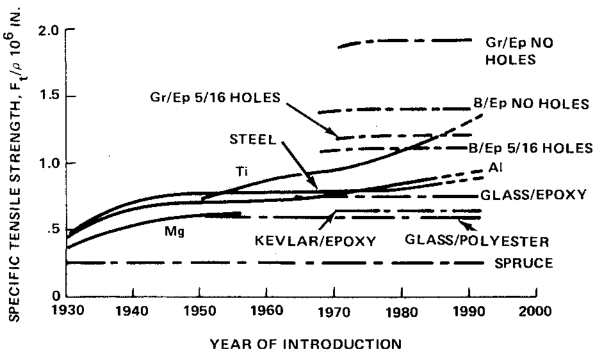

Fig. 12 Specific working tensile strengths of materials.

temperature applications. By the late 1960's organic matrix advanced composites began to appear as viable structural materials. The metal matrix composites (MMC) now being developed should be very effective for future supersonic cruise vehicles if their cost can be reduced significantly. Figure 10 shows materials application during the twentieth century.

Since 1930, aluminum alloys have been the predominant material used on airframe construction for all types of aircraft. Properties have improved during this period, as shown in Fig. 11. This figure includes the properties of titanium alloys and high-strength steel; new alloys of all these materials show considerable promise and are likely to be used in future aircraft.

Figure 12 provides a comparison between advanced composite and metallic specific tensile strengths. Here, the strengths of the composites used are realistic working stress levels for multidirectional laminates as would be used for wing or stabilizer covers. Figure 12 shows that composites are very hole-sensitive relative to metals. However, this penalty is offset by far less sensitivity to tension fatigue. Their effectiveness in compression applications is shown in Fig. 13 where the low density significantly reduces panel weight but, in contrast to metals, compression fatigue now becomes a design consideration.

Materials are available or currently under development which could effect significant weight reductions on future aircraft. It is very probable that all future high-performance aircraft, whether military or commercial, will be made from a mix of aluminum, titanium, steels, and advanced composites. The use of advanced composites is expected to increase with time. Figure 14 shows the material mix of current fighter aircraft, with future projections.

A difficulty with any projection of this type is predicting the time needed for a new structures or materials technology to reach an acceptable level of maturity. Evaluation of the advanced composite development cycle (Fig. 15) indicates that full maturity is reached in about 20 years but that limited military application can begin after 10 years. The full 20-year cycle is probably a minimum for the application of new structures and materials technology to commercial aircraft.

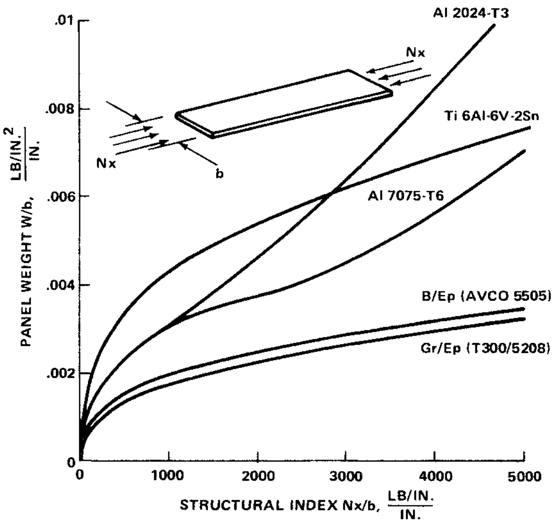

Fig. 13 Weight of long compression panels.

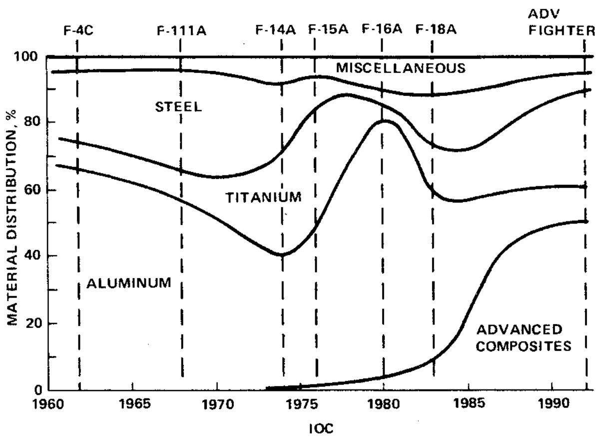

Fig. 14 Material utilization (fighter aircraft).

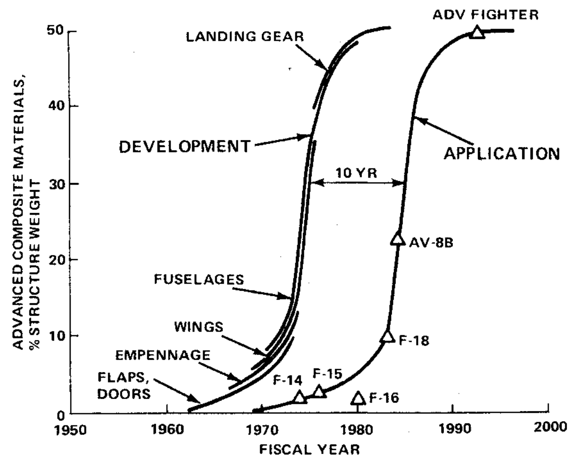

Fig. 15 Advanced composite implementation (aircraft).

VI. Structures and Materials Problems

Past introduction of new SDM technology has resulted in problems from time to time, and some of these have been

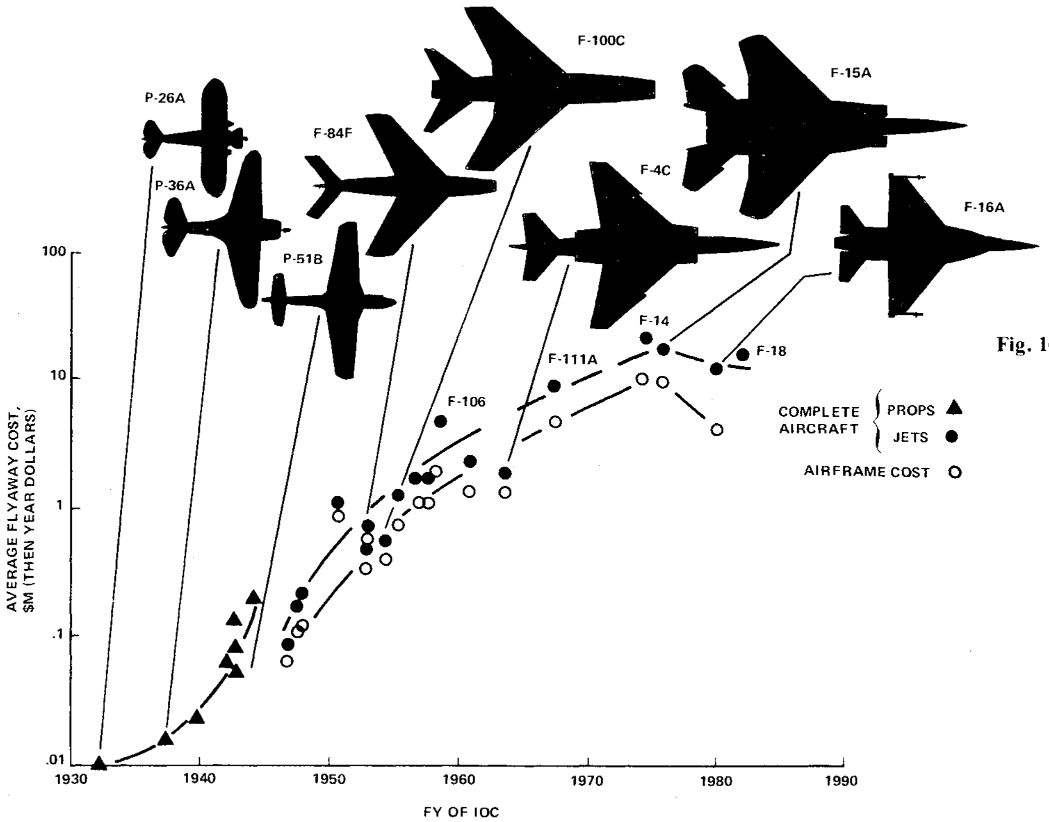

Fig. 16 Fighter aircraft flyaway costs.

major. Perhaps insufficient attention has been given to this history, which can often provide an insight into some of the problems still facing us today.

Consistent quality is a problem with many new material systems. As long ago as 1914, a Dornier engineer wrote: "It had many drawbacks. For instance, it was not produced in consistent quality; more often than not it would exfoliate like the leaves of a book. Impurity inclusions caused frequent embrittlement cracking and, after brief periods of storage, the sheet had the unpleasant tendency to disintegrate in spots to a white powder." This reference was made to Duralumin, the first of many high-strength aluminum alloys which had about double the strength of unalloyed aluminum. Although an airship using Duralumin parts entered service in 1915, Claudius Dornier decided not to use it for the more highly stressed sections of his first all-metal aircraft, the RS-1.

Since that time, aluminum alloys have become the keystone material for aerospace structures. There have been fatigue and stress corrosion problems, but these have been solved by changing alloys or heat treatments, or working to lower stresses.

The problems of exfoliation and inconsistent quality cited by the Dornier engineer are typical of those currently encountered in new material systems. Problems of this type appear from time to time in advanced composites; lately, there has been an additional problem of strength degradation due to humidity.

Humidity has caused several difficulties in the past. One of the earliest major problems occurred on the Western Front in October 1917. The first Fokker Dr 1 triplane, which had a go-ahead in April 1917, was structurally tested on August 11, 1917 to a load factor of 8.38 (108% of design ultimate load). Flight testing of the second and third aircraft began on August 16 and one of these entered service on August 21. By mid-October, 16 triplanes were in service. By October 31, all aircraft were grounded following two fatal accidents. Official investigations concluded that the effects of high humidity in conjunction with unsatisfactory quality control had caused debonding of the wooden outboard tip rib on the upper wing, resulting in loss of the aileron. Modifications were made and the triplanes, re-entered into service during December 1917, experienced no further structural problems.

Fatigue has been a problem in the past, both commercial and military aircraft being designed for longer and longer lives. The P-51 was removed from service in Korea in 1951 by the Australians when extensive wing cracking made repair uneconomical. These aircraft had approximately 1000 h. Current fighter requirements are 4000 h with a factor of 4 by the Air Force, and 6000 h with a factor of 2 by the Navy. In addition, the Air Force requires a damage tolerance capability in conformance with MIL-STD-1530.

Fatigue was a problem in 1930. It was recognized that "the condition of the surface of the material has an immense effect, even small surface cracks producing a marked reduction of the safe limiting alternating stress. Corrosion also has a very marked effect in reducing the safe alternating stress."

This recognition of potential fatigue problems did not prevent the Comet accident in 1954 nor the series of B-47 catastrophic wing failures which occurred in 1958. The B-52 has been reskinned because of fatigue cracks and the C-5A will probably be rewinged. Fighters have not escaped the fatigue problem. An F-100 crash in 1967 led to a program for modifying 682 aircraft to extend service life to 4000 h. The F-111 had problems with the forged steel wing pivot, resulting in grounding in 1970. A nondestructive cold proof test program was an expensive solution to this problem. These are a few of the very costly fatigue problems that are still not well understood.

Fatigue testing accounts for a significant portion of the costs of qualifying a new material or material system. Tests are currently being performed on graphite/epoxy specimens that simulate real-time flight-by-flight fatigue in conjunction with representative humidity and temperature levels. Such tests are needed to generate design data and achieve the level of confidence necessary for introducing these materials into production systems. They are necessary but extremely expensive and, as they are in simulated real-time, it takes many years to obtain the test data.

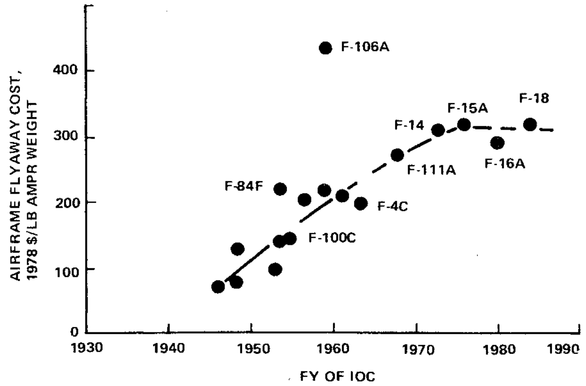

Fig. 17 Airframe costs.

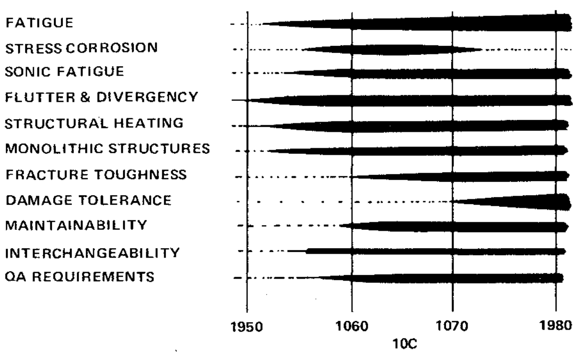

Fig. 18 Parameters causing increases in production costs of fighter aircraft.

The costs associated with composite material qualification and design data generation are becoming prohibitive; millions of dollars may be required for a single fiber-matrix system. Every effort should be made to devise a screening process for selecting only a very few of the more promising new materials for qualification testing. The fully qualified materials should then be given a universal specification.

VII. Costs

As can be seen from Fig. 16, the flyaway costs of fighter aircraft have increased by over three orders of magnitude since 1930. Figure 16 also shows the airframe costs of jet fighters produced after 1945. The airframe cost accounts for more than half of the flyaway cost of most of these aircraft, and the former has increased in the same way as the latter.

Accounting for inflation, the airframe flyaway cost in terms of constant 1978 dollars is shown in Fig. 17. These data were obtained by dividing the airframe cost by the AMPR weight and then correcting for changes in the average labor rate. Escalation factors associated with the labor rate rather than the material were used, since labor has generally accounted for about two-thirds of production costs. Development costs have not been included in the data of either Fig. 16 or 17. The fighter airframe costs, in terms of constant dollars per pound, have escalated by a factor of about three since 1946. Smaller production runs, reduced production rates, increased complexity, and more stringent requirements have been some of the reasons for this escalation.

VIII. Parameters Affecting Costs

Some other parameters which have caused increases in production costs are shown in Fig. 18. The increase in required fatigue life has dictated that more material be placed in critical structural areas to meet target weight.

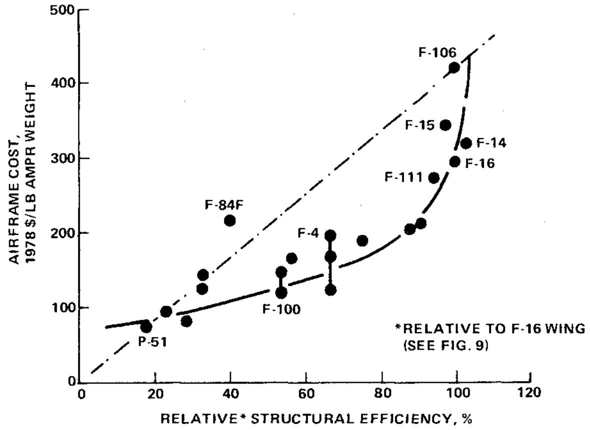

Fig. 19 Effect of increased structural efficiency on airframe costs.

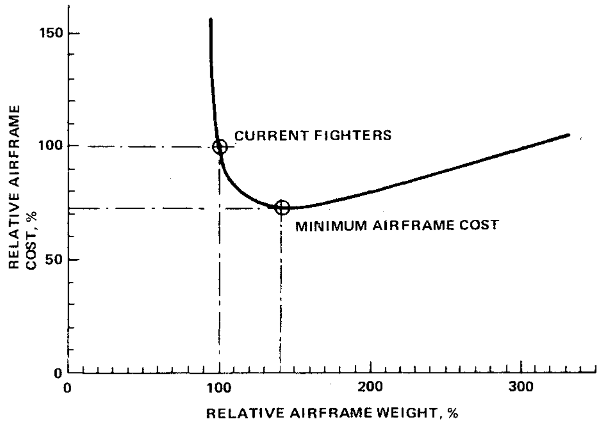

Fig. 20 Fighter airframe weight-cost envelope.

Fatigue, higher temperatures, fracture toughness, and stress corrosion problems have indicated the use of lower-strength aluminum alloys. They have also dictated the use of close-tolerance or interference-fit fasteners, a specified finish for holes and critical surfaces, etc., all of which have increased the cost of design, fabrication, and quality assurance.

Advances in design, materials, manufacturing, and quality assurance technologies have occurred during this time. Tool materials and coatings, numerically controlled machining, automatic feed, chemical milling, etc., have all helped to offset the cost increases resulting from improved performance and more stringent requirements. Computer-aided design and manufacturing will further offset these costs.

It is difficult, if not impossible, to quantify the cost increases or savings associated with each of these effects. It is, however, possible to obtain some estimate of their overall impact in combination.

IX. Airframe Cost Effectiveness

Assuming that the structural efficiency improvements achieved on wings (Fig. 9) would also be realized on the remainder of the airframe (since these are impossible to quantify), and using these data in conjunction with the cost data from Fig. 17, airframe cost per lb can be plotted against structural efficiency. This plot, shown in Fig. 19, indicates that the general trend is for cost to increase as the structural efficiency increases. This result is not unexpected since any structural tailoring has associated cost increases when expressed in terms of dollars per lb.

The data plotted on Fig. 19 show that the cost per lb has increased at a far slower rate than the increase in structural efficiency. The chain dotted line shows constant airframe cost

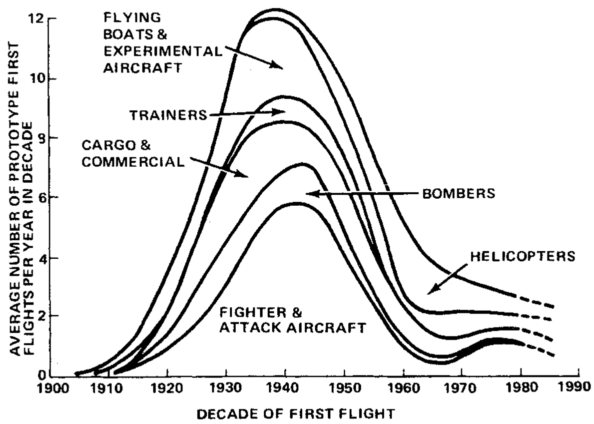

Fig. 21 First flights of prototype U.S. military and commercial aircraft.

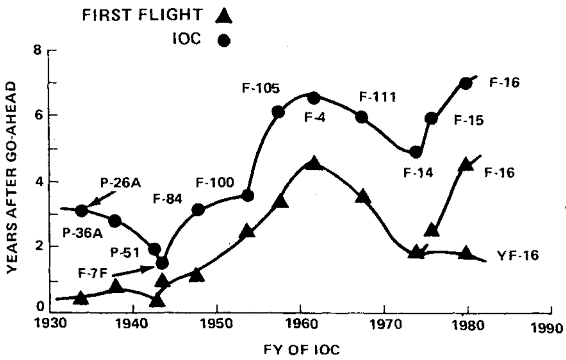

Fig. 22 Years from go-ahead to IOC.

relative to WW II aircraft. The difference between the chain dotted line and the heavier dashed line is an indication of the cost reductions due to manufacturing improvements, which are not insignificant. The heavy dashed line was used to generate the relative cost-vs-relative weight curve in Fig. 20, which shows that cost increases excessively as the limit of structural efficiency is approached. It can also be seen that a 40% weight penalty is associated with minimum airframe costs.

These data are examples of the much more detailed weight/cost information that must be available to make decisions during the preliminary design phases since mission performance and overall program cost are very sensitive to weight change. The data are also needed in more detail by the SDM development community since they reflect today's state-of-the-art. Current technology has reached an extremely sophisticated level and must be used as a baseline to measure projected new SDM technology advantages.

X. Opportunity

The opportunity for introducing new SDM technologies into new systems is fast disappearing. Figure 21 shows historical data as it relates to new military and commercial aircraft. In terms of the number of prototype first flights per year in each decade since the beginning of the century, a peak of 12 aircraft per year was reached between 1930 and 1940. Since then, the number of first flights has steadily declined. For the remainder of this century, the number will probably not exceed two per year and the majority of these will be either helicopters or experimental, commercial, and cargo aircraft. There will be very few opportunities for applying new SDM technology to new aircraft during the next decade, but there will be some opportunity for application to prototype or demonstrator aircraft and spacecraft.

Another problem associated with opportunity is that appropriate new technologies are selected very early in the preliminary design phase of a new military aircraft. Go-ahead to Initial Operating Capability (IOC) takes from 6 to 8 years, Fig. 22, but most major technical decisions are made during the first year. This means that the candidate technologies must be mature at go-ahead.

Since technological maturity requires in-service experience in conjunction with a sound design and manufacturing data base, the only way to achieve the necessary level of maturity will be to use these new technologies on existing systems. This approach is not new; it is being used by NASA to generate a data base for the application of graphite/epoxy structures to commercial aircraft and has been used by the Air Force, Navy, and Army in a similar manner. The problem with this approach is that the user is often not very enthusiastic about using new technology, especially if increased manufacturing, maintenance, and inspection costs are incurred.

XI. Conclusions

Significant improvements in structural efficiency have been achieved during the past 50 years through a series of continuing advances in SDM technology. Today, aluminum and titanium fighter aircraft structures are appreciably more efficient than those of WW II fighters. However, a limit has now been reached and any further improvements in the efficiency of conventional structures will only be achieved at excessive costs.

Some new SDM technologies offer the promise of further major improvements in structural efficiency without associated cost penalties. However, the opportunity to implement these technologies on new systems is rapidly diminishing. When an opportunity does appear, it is improbable that the new technologies will be approved unless steps are taken now to bring them to acceptable levels of development.

These new SDM technologies should be evaluated against the needs of future projected systems using an approach similar to that outlined in this paper. A very few new materials and new technologies that show real promise for improving performance and reducing costs should be selected for a concentrated development effort to generate qualification, design, and processing data for production implementation. Specifications should be generated to define the constituents and properties of specific types of composites or new alloys. Full-scale parts must be ground-tested for certification and introduced, as replacement parts, into service.

Such an approach would require a well-defined gate between research and development. Research should continue in its present form. However, gates must be introduced to limit the number of new technologies which graduate into the full development cycle, where both costs and requirements are escalating.

Finally, there has been a significant advance in the capability to design, analyze, fabricate, and inspect complex aerospace structures during the past 20 years. Much of this advancement is associated with the availability of computers and the development of computer-aided design and manufacturing systems.

Since capability can breed requirements and, since unnecessary requirements can stifle advancement, any new requirement should be thoroughly evaluated and its impact fully defined before it is implemented.

Acknowledgments

The author is grateful to J. Tedesco, J. Raha, P. Schwartz, S. Dastin, R. Witt, R. Messler, S. Specht, and others at Grumman for their assistance in preparing this paper.

References

[1] Nayler, J.L. and Owen, E., *Aviation Today,* Frederick Warne & Co., Ltd., London and N.Y., 1930.

[2] Fokker, A.H.G. and Gould, B., *Flying Dutchman, The Life of Anthony Fokker,* Henry Holt & Co., N.Y., 1931.

[3] Loudy, F., *Metal Airplane Structures,* The Norman W. Henley Publishing Co., New York, N.Y., 1938.

[4] Titterton, G.F., *Aircraft Materials and Processes,* Pitman Publishing Corp., N.Y., revised ed., 1941.

[5] Peery, D.J., *Aircraft Structures,* McGraw-Hill Book Company, Inc., N.Y., 1949.

[6] Gardner, H.H., "Structural Problems of Advanced Aircraft," *Journal of the Royal Aeronautical Society,* April 1952.

[7] Shanley, F.R., *Weight-Strength Analysis of Aircraft Structures,* Dover Publications, Inc., New York, N.Y., 1960.

[8] Haddow, G.W., and Grosz, P.M., *The German Giants—The German R-Planes,* Putnam, London, 1962.

[9] Bruhn, E.F., *Analysis and Design of Flight Vehicle Structures,* Tri-State Offset Company, Cincinnati, Ohio, 1965.

[10] Levenson, G.S., and Barro, S.M., "Cost-Estimating Relationships for Aircraft Airplanes," Rept. RM-4845-pR, The Rand Corporation, Santa Monica, Calif., Feb. 1966.

[11] Hoff, N.J., "Thin Shells in Aerospace Structures," *Astronautics & Aeronautics,* Feb. 1967.

[12] Wagner, R., *American Combat Planes,* Doubleday & Company, Inc., Garden City, N.Y., 1968.

[13] Langley, M., "The History of Metal Aircraft Construction," *Journal of the Royal Aeronautical Society,* Jan. 1971.

[14] White, W.D., "U.S. Tactical Air Power, Missions, Forces and Costs," The Brookings Institution, Washington, D.C., 1974.

[15] Lubin, G., *Handbook of Fiberglass and Advanced Composites,* Van Nostrand Reinhold Co., N.Y., 1975.

[16] Nicolai, L.M., "Fundamentals of Aircraft Design," School of Engineering, University of Dayton, Dayton, Ohio, 1975.

[17] Hedrick, I.G., and Whiteside, J.B., "Effects of Environment on Advanced Composite Structures," Paper 77-463, AIAA Conference on Aircraft Composites, San Diego, Calif., March 1977.

[18] Taylor, J.W.R. and Munson, K., *Jane's All the World Aircraft 1977-78,* MacDonald and Jane's Publishers, Ltd., London, 1977.

[19] Knaack, M.S., *Encyclopedia of U.S. Air Force Aircraft and Missile Systems, Vol. I, Post-World War II Fighters,* Office of Air Force History, Washington, D.C., 1978.

[20] Bannink, G., Hadcock, R., and Forsch, H., "Advanced Design Composite Material Aircraft Study," *Journal of Aircraft,* Vol. 15, Oct. 1978, pp. 661-669.