NASA Contractor Report 3720

Guidelines for Composite Materials Research Related to General Aviation Aircraft

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PREFACE

This report, covering the work done for the NASA Langley Research Center on Contract NAS1-16804 during the period September 30, 1981 through February 13, 1983, was prepared by the Principal Investigator for MSC, Mr. Norris F. Dow, in collaboration with the Program Manager for MSC, Dr. B. Walter Rosen, and other members of the MSC staff, particularly Mr. Edward A. Humphreys. The author wishes to express his appreciation to his collaborators and to Mr. Marvin Dow, who was the NASA Technical Representative, for their many contributions and technical discussions.

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INTRODUCTION

OBJECTIVES

The effective utilization of composite materials in general
aviation aircraft requires the technology to produce structures
that are durable, save weight, and are cost competitive with
metal structures. To some extent, this technology is available
from development programs for transport and military aircraft.
However, the needs of general aviation aircraft are different. To
meet these needs, will require an integrated research program.
At the time this contract was let, NASA planned a general
aviation composites program. However, there is no planned
program now. Therefore, this report is written in the context of
what a program should address, if a program is ever established.
This report presents the results of studies carried out by the
Materials Sciences Corporation to identify and generate guide-
lines for a research program that would be most appropriate for
the general aviation requirements.

The overall objective of such a general aviation composites
program would be to develop the technology that will lead to pro-
gressive introduction of composite materials into production
aircraft. To accomplish this objective, the program definition
phase during which candidate materials, designs and processes
would be selected would be followed by Phase II - Development and
Test, which would explore applications of promising material
combinations and structural configurations. Finally, Phase III -
Demonstration, would demonstrate practical approaches to the
manufacture of composite structures.

A program of this type is strongly influenced by the rela-
tive emphasis upon the differing time scales which could be
considered. As illustrated in figure 1, simple near term
applications (e.g. control surfaces) could be defined today
and would only require the phase III effort. Progressively,

more ambitious applications would require more research and
technology development as suggested in figure 1. During the
course of the present studies, the decision was made to place
emphasis upon the research required for long-range applications.

Thus, the program definition phase had four specific ob-
jectives, as follows.

   (1) Determine, with participation by industry, the long-
        term developments in composite materials and the
        associated manufacturing processes that afford a po-
        tential payoff in terms of technology advancement
        and commercial acceptability.

   (2) Devise and use anlysis methods to evaluate the
        effect of various composite materials and manu-
        facturing processes on airframe efficiency and
        cost.

   (3) Establish design criteria for composite components
        and select the appropriate materials and manufactur-
        ing techniques.

   (4) Identify specific obstacles to the application of
        composite materials in general aviation structures
        and recommend the necessary research and devel-
        opment activities to overcome the deficiencies.

APPROACH

In order to achieve its objectives, the research definition
study was formulated to contain the individual tasks outlined
in figure 2. These tasks may be grouped into three major
categories: definition of needs and requirements; preliminary
design and cost evaluations, and detailed design studies.
The underlying logic here is that a quantitative evaluation of
the merits of specific composite materials and structures must be
conducted in order to define desirable areas of research. This
must be supplemented by interaction with industry in order to

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assess possible, promising technologies which could influence the research tasks. These would be primarily in the area of new materials and processes.

Interface with General Aviation Manufacturers

The definition of needs and requirements drew upon a series of interactions with general aviation manufacturers. (A list of the organizations visited is presented in Appendix A and the authors express their appreciation to those organizations for their cooperation. As agreed, no specific data are attributed to any of these sources.) These discussions were directed toward:

(1) Cost evaluations, both of manufacture and of premiums for performance.

(2) Design criteria: stiffness requirements, surface smoothness, minimum gage requirements.

(3) Trends in design: wing loadings, aspect and taper ratios, overall.

Results of the discussions with regard to these objectives were generally disappointing. Thus, specific quantitative cost data were not obtained. Stiffness requirements were ill defined, probably due to the fact that this was not a serious limiting factor for aluminum structures. Design trends appeared to stem more from modification capability of old designs than from generation of new ones. There were however, certain positive results of these discussions.

It was possible to interpret the discussions with manufacturers to yield important generalities with regard to cost. For aluminum structures, the total manufactured cost runs from $20 - $100/lb., and of that cost 80 - 95% is manufacturing cost. Also, manufacturers do recognize the substantial dollar value of performance ~ assigning it, for example, in many cases, a premium of $100 to $500 per pound of weight reduction.

The general aviation manufacturers conveyed a cautious interest in composites. They generally expressed a need for a comprehensive data base for the specific materials to be used. There did not seem to be an appreciation of the joint role of analysis and experiment in the definition of design allowables. Associated with this was the impression that composites are approached with extreme conservatism, particularly in the area of criteria to be used for design. Nevertheless, the progress of the Lear-Fan composite airplane has had a definite impact. The industry does recognize that composites have an important role to play. Initial advances are likely to be made by utilization of contracts to commercial transport and/or military aircraft manufacturers, for initial fabrication of major general aviation components. They agreed that NASA's role in composites research for the long term is proper.

From these interactions with industry, the study was influenced directly towards a dual emphasis upon manufacturability and upon performance. Thus, raw material costs were seen as less important, while manufacturing cost was more important, and performance, as will be seen subsequently, the most important factor.

Analysis Methods Developed for the Study

To define the recommended research required the utilization of quantitative procedures for evaluation of cost and performance of candidate materials and structures configurations. The range of variables to be considered is enormous. Hence, it was necessary to develop improved methods for carrying out these evaluations. Such developments were necessary for material selection, for structural evaluation, and for cost assessment. These methods are presented in the appendices to this report, but an illustration at this point will indicate the nature of the required developments.

In the area of material selection, a major problem associated with the choice of material configuration is that different materials have the best values of different properties. Thus, it becomes necessary to identify the key grouping of properties for a given application and then screen materials to find the best-possible combinations of these properties or property combinations. For example, in the compression loading of a flat plate, performance is dependent upon both the compressive strength and the buckling resistance of the laminate used for the plate. The buckling resistance is measured by a grouping of all of the four in-plane elastic constants of the laminate. This function of properties is called the plate buckling modulus. Thus, laminates which plot farthest from the origin on a plot of compresive strength vs. plate buckling modulus are candidate laminates for maximum performance in this application. Such a plot is shown in figure 3. In this application, the extremum points were obtained from laminates of the $\pm\theta$ family, a surprising and convenient result. Having defined the governing family, preliminary design trade studies could be conducted efficiently for this family only.

For structural evaluation, the conventional structural efficiency methodology has the shortcoming of representing only structural elements of constant cross-section. Thus, for practical structural configurations having taper, such as the wing box structure, a more precise approach was needed. This approach needs to reflect performance, in terms of overall bending and twisting stiffnesses, as well as in load (or moment) carrying capability. An illustration of the resulting methodology is shown in figure 4, wherein normalized wing weight is plotted as a function of structural index, which for this application depends upon bending moment, chord and depth at the root section. This approach permits the display of wing geometry (aspect ratio and thickness) as well as of stiffnesses. Utilization of this methodology is described in Appendix C.

A range of cost studies was carried out at differing levels of detail. A convenient approach to the summary of cost data was also required. It was found desirable to approach cost from the viewpoints of both buyer and seller. This work is described in Appendices B and C.

Preliminary Design Considerations

A major question which impacts the results of the quantitative design studies is the definition of criteria for the formulation of design allowables. Analysis of laminate strength has been investigated primarily through the approach of translating laminate loads into lamina (and interlaminar) stress states and utilizing those in criteria for initiation of local failure. This is the procedure for the so-called first ply failure criterion. Conversely, experimental strength determination has dealt primarily with overall laminate failure rather than any localized failure. Eventually, because most laminates under static or cyclic loading will experience local failures prior to ultimate failure, an understanding of the entire failure process will be essential for sound design. At the present time, concern with impact damage tends to limit designs to low allowable strain levels.

In time, the design approach will be more ambitious, and will consist of a sensible and conservative analysis supplemented by realistic testing. The major elements of a conservative design analysis are: use of low strain allowables, special analysis of complexities, special treatment (e.g. reinforcement) for problem areas, establishment of inherent defect acceptance criteria, and the use of representative defects for design analyses to increase reliability. Realistic testing involves both coupon testing for environmental effects, statistical data base, etc., and sub-component testing for verification of failure modes and levels. Both the analysis and the experiments will address the basic

problems of laminates with holes and with disbonds. Design
allowables will be generated for basic loading conditions. These
results will be used to address actual design complexities by
drawing upon treatment of: combined stresses, utilizing inter-
action curves; load spectrum effects, utilizing cumulative damage
hypotheses; environmental effects, utilizing reduced allowables
and revised stress analyses; and lifetime calculations, utilizing
residual strength distributions at moderately long times, in
conjunction with effective crack growth laws. All of these
methodologies are in a continuing state of development. Hence,
it is necessary to be conservative. Thus, this basic design
procedure must be supplemented by testing.

The testing will generally involve a full-scale structure
under fatigue. It should be recognized, however, that fatigue
effects can be very profoundly influenced by environmental
effects, and realistic full scale environmental testing can be
prohibitively expensive. Therefore, it is necessary to establish
an estimate of mean lifetime with a full scale fatigue test in an
ambient environment. Then coupon and element testing are used to
obtain a definition of material and structural variability
effects. It is necessary to utilize a procedure whereby test
coupons with holes establish the statistics of failure. Such
coupons are also used to treat the environmental effects. At the
second level, representative small components that contain
realistic stress fields should be tested to demonstrate the
possibilities of unanticipated complexities and their influence
upon failure.

For an approach consistent with the conservative present day
philosophy, first ply failure has been selected for the trade
studies.

OUTLINE OF THE REPORT

The results of the supporting studies, which have been con-

ducted to aid in the formulation of the recommended research plan, have been relegated to Appendices to this report. The supporting investigations for this definition study have been grouped into the following categories: materials and structural elements (see Appendix B); structural components (see Appendix C); composite aircraft structures (see Appendix D); manufacturing technology (see Appendix E); and cost (see Appendix F).

The body of the report is the summary of the major elements of the recommended research plan. The research tasks are presented separately for each of the following subject areas: mechanics of materials; material development; mechanics of structural elements; development of structural elements; mechanics of structural components; and manufacturing technology development.

RECOMMENDED RESEARCH PLAN

CONCEPT

The research plan derived from the various inputs and studies, as described in the foregoing section, is a comprehensive one encompassing both scientific and engineering technologies relating to composite materials and structures. Indeed initialy the comprehensiveness may appear overwhelming. The tasks can be addressed in an orderly fashion, however, to yield a growing body of results, - with early pay-offs as well as cumulative benefits. Thus the concept has been insofar as possible to identify researches providing both fundamental and readily applicable results. Further, effort has been made to provide appropriate evaluations of the potential contributions of the researches in terms of early and long-term needs in general aviation aircraft, to provide directions for emphasis in the execution of the plan.

AREAS FOR RESEARCH

In the following presentation, recommended researches have been divided into eight categories, as follows:
1. Mechanics of materials
2. Material development
3. Mechanics of structural elements
4. Structural element development
5. Mechanics of structural components
6. Development of components
7. Development of composite structures
8. Development of design methodology

Each category is first discussed in a uniform, repetitive format. This format, adopted in an effort to make the presentations uniformly objective, -

1. Identifies and defines the Problem Area.
2. Describes the Effort Required to resolve the problem.
3. Evaluates the Potential Merit of successful research.

Details of the analyses and experiments required in the
individual categories are expanded as appropriate in the Appendices.
Finally, the evaluations are reviewed and compared to obtain
bases for directing the research emphasis.

Mechanics of Materials

Problem Areas

Despite the progress that has been made in understanding
the mechanics of composite materials, three key problem areas
remain in which research is needed to provide sound, fundamen-
tal bases for the determination of properties and failure criter-
ia, and guidelines for developments to attain the true poten-
tials of advanced materials. These areas comprise:

1. Inadequate knowledge of transverse properties and fail-
   ure mechanics, of the relationship between them, and of

2. The differences in composite strengths and modes of
   failure if woven fabrics are used in place of straight,
   unidirectional filaments, and also

3. Inadequate capability to correlate composite strengths
   with matrix properties and insure the achievement of
   the potentials of the reinforcement/matrix combina-
   tion.

The first of these problem areas relates to the problem
of the development of sufficient understanding of the mechanics
of failure of composites under realistic stress states so that:
(1) reliable methods of strength analysis can be developed,
and (2) so that failure criteria which do not impose undue penal-

ties on the use of composites can become acceptable. The other
two areas relate more to extending and applying enhanced under-
standing of failure mechanics to develop improved reinforcement
forms and matrix materials. Discussion of the present state of
understanding in all three problem areas is given in Appendix B.
Researches directed toward the solutions of the problems are
outlined in the following section.

Research Effort

The research effort needed for the mechanics of materials
problem areas is both analytical and experimental. Three essen-
tially separate but partially interacting and overlapping re-
search studies for this effort are discussed in a summary fashion
in this section and related technical detail is given in Appen-
dix B.

RESEARCH STUDY 1 - Transverse Properties and Failure
Mechanics of Unidirectionally Reinforced Composites

Objectives

Provide an adequate, reliable data base for development of
analytical models, and analysis methods based thereon, capable
of predicting modes of failure and strengths of unidirection-
ally reinforced advanced composites under loadings realistically
representative of those encountered in general aviation aircraft.

Approach

The study involves extensive tests designed specifically
to evaluate properties - not adequately available in the liter-
ature, such as transverse tension, compression, and shear, -
which can cause or contribute to premature failures (i.e. failure
not utilizing the potentials of advanced fiber reinforcements).
Particular emphasis should be upon instrumentation and test design
to enable indentification and correlative of failure modes and
stresses. Accordingly, specimen and test design should be a colla-
borative effort of analyst and experimentalist to insure pertinent
results. Material used should be a typical advanced composite ma-
terial suitable for end application in general aviation aircraft.

Data, as developed in the study, should be analyzed by avail-
able and developing methodology, serving as a basis for rede-
sign of both methodology and subsequent tests. A substantial
initial program to yield needed basic data can be laid out in ad-
vance, however, based on the problem areas discussed in Appendix B.

Desired Results

Data and methodology for strength prediction for advanced
composites, of validity demonstrated adequate as a basis for:

(1)   Acceptable failure criteria.
(2)   Comparisons, and extensions for developments, in
        Studies 2 and 3.

RESEARCH STUDY 2 - Comparison of Strengths and Failure
Modes of Unidirectionally Reinforced and Woven-Fabric
Reinforced Composites

Objectives

Determination of modes of failure and stengths of woven-
fabric reinforced advanced composites for comparison with the
results for unidirectional.

Approach

Based on the results from the unidirectionally reinforced
specimens, critical experiments should be designed to explore the
changes in failure modes produced by the woven reinforcements.
Then quantitative data and failure criteria should be developed
for wovens as for unidirectionals, if the critical experiments
are favorable to the wovens.

While detail designs for the crucial experiments for this
study must await the results of Study 1, background data needed
for those designs can be projected to be concentrated in areas
for which failure modes are in-plane, transverse type failures,
and/or minimum gage restrictions apply.

Desired Results

Definitive determination of load conditions for which woven
or unidirectional reinforcements are better suited. Develop-
ment of failure criteria for wovens for improved performance.

RESEARCH STUDY 3 - Correlation of Composite Strengths
with Matrix Properties

Objectives

Develop capability to correlate composite strength as well
as stiffness properties with properties of constituents, particu-
larly to account properly for the role of the matrix character-
istics. Derive guidelines for development of matrix materials
having improved performance potentials or ease of manufacture.

Approach

The study requires a coordinated analytical/materials de-
velopment effort for specimen design, as follows:

(1)   A parametric analytical study, based on the results
        of Studies 1 and 2, to define ranges of matrix proper-
        ties of interest (see Appendix B).

(2)   Trial formulation of resins to approximate the de-
        sired properties.

(3)   Coordinated selection of experimental matrix mater-
        ials, fabrication and test of specimens.

(4)   Repeat the sequence (1) - (3) with appropriate modi-
        fications in light of the results.

The magnitude of the effort required for this study relates
to the developing analysis methodology discussed in Appendix B.
Details of tests and specimens are not attempted.

Desired Results

Quantitative data on: (1) ranges of matrix properties of
interest as capable of favorably influencing performance, also
(2) corresponding ranges of practical interest from the stand-
point of capability to be formulated readily. Most valuably,
the results should provide guidelines for assessing potentials,
such as, for example, the performance of thermoplastics.

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Material Development

Areas for Research and Development

Three material development areas in which potential for improvement can be reasonably quantified, in terms of research and development effort required, have been identified as corollary results of various studies in this program. The three are somewhat different in nature, one from another, and are directed toward quite different types of improved characteristics - toughness, cost effectiveness, and structural efficiency. These three areas are:

1. Development of multi-directional and through-the-thickness running reinforcement configurations.
2. Development of suitable reinforcement/thermoplastic resin combinations.
3. Development of medium-low density core materials for plate and shell structures.

The first of these areas is directed toward the utilization and furtherance of the growing technology of multi-directional fabric formation to achieve economical and efficiently reinforced composites. The second involves the dual development (deriving in part from Study 3 of the foregoing section) of (1) a composite system having reinforcements suitable for forming in place, and (2) a thermoplastic resin combining formability with good structural performance. The third, an output of the trade-off studies reported in Appendix E, is directed toward part count reduction with superior performance.

The specific tasks associated with these three development programs are summarized in the following sections and detailed in the associated Appendices.

Development Research Effort

The effort required in all three material development areas is primarily of a research nature, both analytical and experimental, even though its end objective is material development, - as the following summarizations demonstrate.

DEVELOPMENTAL RESEARCH STUDY 4 - Multi-Directional
Reinforcement for Enhanced Toughness

Objectives

Develop multi-directional reinforcement configurations for filamentary composites combining ease of handling and formability with good in-plane reinforcement efficiency, which provide improved properties, especially enhanced transverse impact damage resistance compared to that of conventional composite laminates.

Approach

As discussed in Appendix B, a multi-phase study is appropriate, involving:

(1)   Analytical design of reinforcement configuration to optimize both in-plane and through-the-thickness characteristics, as well as providing ease of manufacture both of the reinforcements and of the resulting composite.

(2)   Trial manufacture and selection of promising reinforcement designs.

(3)   Trial manufacture and selection of promising composites utilizing these reinforcements.

(4)   Determination of composite properties achieved.

(5)   Evaluation of results for both manufacturability and performance, and generation of guidelines for final development.

Desired Results

Evaluations of potentials and trade-off of various multi-directional reinforcements as regards both performance and manufacturing. Development of preferred configurations for general aviation applications.

DEVELOPMENTAL RESEARCH STUDY 5 - Reinforcement/Thermoplastic
Resin Combinations

Objectives

Develop more cost effective composites, by providing in-
creased formability through the use of thermoplastic matrix
materials, while retaining good performance through the use of
adequately stable, formed reinforcement configurations.

Approach

There are two parts to the problem, namely: (1) the provi-
sion of a thermoplastic matrix material capable of good perfor-
mance (competitive with epoxy), and (2) the provision of a rein-
forcement capable of retaining adequate integrity of configura-
tion to yield a high performance composite while accommodating
the shape changes of manufacture to end-product shape. First, a sat-
isfactory resin must be demonstrated. As noted in Appendix E,
promising enough candidates have appeared to suggest that a po-
tential exists that such a material is accessible. Reformulation
of selected materials should be directed by the results of Study 3.

Part two of the study does not have to wait for resin de-
velopment. Performance of the reinforcement as regards forma-
bility can and should be evaluated in available matrix materials
which may not meet the demands of performance sought for the final
composite. Candidate reinforcement configurations, as noted in
Appendies B and C should include both biaxial and triaxially
woven fabrics, in view of configurational stability of wovens
and of the demonstrated formability of the triaxial weaves.

The final part of the program involves the marriage of se-
lected resins and reinforcements, with construction and test
of adequate numbers of specimen for full evaluations.

Desired Results

Development of an advanced composite material combining
ease of manufacture for general aviation aircraft applications
with advanced performance characteristics.

DEVELOPMENT RESEARCH STUDY 6 - Medium-Low Density Core Materials

Objectives

Development of a medium-low density core material for use as the central ply in composite laminate plate and shell structures to utilize the substantial benefits of minor reductions in density for achieving minimum weight with minimum part count as described in Appendices B and C.

Approach

Various methods for the production of a resin core material should be evaluated, with emphasis upon the use of low density fillers such as microballoons or hollow organic fibers, even though they lead to low strengths and stiffnesses (because, as shown in Appendix B, core material properties need only be adequate to stabilize the face plies, as in conventional, low density core sandwich construction). Target values of core properties are discussed in Appendix B. Guidelines for material formulation will be generated in Studies 3 and 5.

Promising core materials should be fabricated into medium-low density core plates and tested for plate buckling and ultimate strength characteristics as discussed in Appendix C.

Desired Results

Development and demonstration of a cheap, medium-low density core material for use in general aviation aircraft wing and fuselage structures of high structural efficiency with minimal stiffening.

Mechanics of Structural Elements

Problem Areas

Because of the vastly different properties of advanced composites compared to those of ductile, isotropic metals, available extensive studies of aluminum-alloy plate and shell buckling and ultimate strengths contribute little toward the design of composite plate and shell structural elements. Herein "plate and shell structural elements" are taken to mean the basic, building block units of which structural components such as stiffener sections, spar caps, etc. are made up. Studies to facilitate design of composite elements are not required to be as extensive as those for aluminum, however. Knowledge of methodology developed for metals, both of an analytical and of an experimental nature is in part transferable to composites.

The mechanics of composite buckling, and particularly of post-buckling and failure are sufficiently different from those for metals, however, so that both the fundamentals thereof and specifics of various composites need to be explored if aircraft structural designs utilizing composites are to be made economically and with confidence. In the studies reported herein, four problem areas are identified as those for which such explorations would be most profitable for general aviation aircraft applications. These are:

1. The fundamental one of the plate buckling and ultimate strength characteristics of the "building block" composite laminate configurations, supplemented by

2. The related characteristics of laminates utilizing more economical reinforcement forms such as wovens.

3. Extensions of 1 and 2 to include evaluations of influences of material variations (such as the use of thermoplastic resins, as from Study 5, or medium-low density core materials, as in Study 6).

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4.   Problems of material response not encountered in me-
      tals such as interlaminar shear and other through-
      the-thickness characteristics as in Study 4.

The researches needed in all four of these areas are funda-
mental in nature, - by necessity. To obtain data directly
usable in design for all likely candidate material combinations
and configurations would be an overwhelming task. Just as re-
sults in the related area for the mechanics of aluminum-alloy
structural elements provide guidelines and methodology for compos-
ites, however, so results for the baseline composites as des-
cribed here can provide needed guidelines for variants therefrom.
Researches toward the needed results are summarized in the fol-
lowing sections.

Research Effort

The research effort on the mechanics of structural elements,
while desirably of a general nature, practically needs a specific
composition to use as a baseline. Review of available materials
(see Appendix B) nominates a medium-high modulus carbon fiber
(circa 200 $\frac{GN}{m^2}$ [30 msi]) in an epoxy matrix as a representative
material for general aviation applications. The research effort
in each of the following areas is designed to utilize this base-
line material at a volume fraction reinforcement of 0.6.

RESEARCH STUDY 7 - Flat Plate Compressive Properties of
Advanced Composites

Objectives

Experimental determination of buckling, ultimate, and failure modes of baseline unidirectional composite laminate flat plates with selected reinforcement configurations. Use of experimental results to develop methods of strength prediction.

Approach

The approach follows procedures used for aluminum alloys, comprising tests of flat plate elements in V-groove fixtures or as elements in square tube or stiffener - section components. A wide range of proportions should be tested to provide a wide ratio of ultimate to buckling stresses. Correlations are to be made of calculated and measured buckling stresses, ultimate loads, and unit shortenings. New influences to be evaluated compared to metals are: (1) effects of anisotropies as established by changes in reinforcement configuration, - particularly for the near-minimum gage laminates of interest for general aviation aircraft, (2) correlations of failure modes with those of Study 1, and (3) relative magnitudes of ratios of buckling and ultimate stresses at various stress levels.

The program benefits from pioneering studies in aluminum alloys, so it becomes of manageable proportions. Specimen configuration, numbers of tests, etc. are discussed in Appendices B and C.

Desired Results

Experimental verification of methods of prediction of failure characteristics of typical laminate configurations. Determination of relative performance of various configurations, to provide bases for selection of the better ones for exploitation. Confirmation of predicted performance gains compared to aluminum-alloy flat plates.

RESEARCH STUDY 8 - Comparison of Woven and Straight,
Unidirectional Filamentary Reinforcement for Composite Flat Plates

Objectives

Experimental evaluation of the actual performance as plate
elements in compression of advanced composite laminates having
straight, unidirectional reinforcements compared to ones having
woven-fabric reinforcements.

Approach

The approach follows directly the procedures used in Study
7, but would be limited in scope, as described in Appendix B
to proportions and configurations for which calculations suggest
either: (1) significant differences in performance for the woven
or straight reinforcements, or (2) critical regions in which
differences in performance depend crucially upon the actual
failure characteristics encountered.

A preliminary layout of an experimental program for this
study would suggest the use of only three woven configurations as
identified in Appendix B. If the results of Studies 2 or 4 so
suggest, the program should be expanded to evaluate further
weaves derived from those studies.

Desired Results

Definitive measures of relative performance of unidirection-
als and wovens as reinforcements. Guidelines for further de-
velopment of reinforcements having favorable handling charac-
teristics for ease of manufacture.

RESEARCH STUDY 9 - Evaluations of Effects of Curvature on
Buckling and Ultimate Strength of Composite Plate Elements

Objectives

Experimental evaluation of the effectiveness of curvature
for the stabilization of composite plate elements in compression.
Derivation of guidelines for curvature requirements for optimum
performance and comparison with predictions. Development of
curved element stiffener sections.

Approach

Determine buckling stress vs. plate width and curvature re-
lationships. With the curved plate buckling characteristics
defined, investigate curved element stiffener sections beginning
with hat-type sections for which twisting failure is not apt to
be encountered. Studies in this area should be exploratory, and
the size of the effort dependent upon the results. Also to be
explored are Zee-type sections having curved elements. Here the
possibility of twisting is evident. Twisting boundaries should
be established by preliminary tests as discussed in Appendix C,
and further investigations made dependent on the findings of
these preliminary experiments.

Desired Results

Verification and quantification of the improvements in per-
formance accessible by the replacement of flat plate composite
compressive elements with curved plate elements wherever feasi-
ble.

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RESEARCH STUDY 10 – Evaluation of Buckling Performance of
Laminated Composite Plates Having Medium-Low Density Cores

Objectives

Evaluation of performance of medium-low density core ma-
terial developed in Study 6. Verification of structural effi-
ciency indicated by analysis in the current studies.

Approach

The approach follows again that of Study 7, to provide
compressive buckling and ultimate strength values under
conditions similar to those for the conventional laminates of
Study 7. Particular attention should be paid to the failure modes
encountered to determine whether the core material contributed to
premature failure. Data collected should be correlated with the
results of Study 7 and to the parametric predictions of the
current studies.

Desired Results

Demonstration and quantification of the performance and merit
of medium-low density core plate construction for providing
structural efficiency with minimal amounts of supporting struc-
ture.

RESEARCH STUDY 11 - Studies of Interlaminar Shear
Characteristics of Advanced Composite Laminates

Objectives

Basic evaluations of interlaminar shear stresses at bolted
or riveted joints in advanced composites. Quantitative deter-
mination of effectiveness of woven-fabric reinforcements at
such joints to reduce the interlaminar stresses. Demonstration
of improved fatigue lifetimes of joints through interlaminar
shear reduction.

Approach

Development with the aid of finite element analyses of
representative model joint designs for experimental investi-
gation of characteristics of joints in advanced composites.
Utilization of these models to evaluate static and fatigue pro-
perties of various approaches to reinforcement design in the
vicinity of the joint. In particular, as discussed in Appen-
dix B, evaluations are to be made analytically by exercising
the finite element analysis program, of the relative effec-
tiveness of unidirectional and woven-fabric reinforcements.
Improved reinforcement designs developed by this program are
then to be tested in fatigue to evaluate increases in resistance
to crack propagation and for enhanced fatigue lifetimes.

Desired Results

Development of improved, standardized approaches to joint
design. Increase in fatigue lifetimes of joined structures.

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Development of Structural Elements

Areas for Research and Development

Because of the vast number of combinations of materials and configurations of reinforcements possible with composites, a great need exists for concentration of developmental effort on the few good rather than on many acceptable possibilities for composite structural elements. First attempts to define guidelines for selection of elements for exploitation are described in Appendix B. The candidate elements should have characteristics leading to:

1. Economy via ease of manufacture.
2. Minimum weight with minimum part count.
3. Surety of achievement of predicted properties.

To contribute to meeting such requirements, two rather broadly based areas for research and development of structural elements have been defined. The areas defined are:

1. Research, primarily experimental, directed toward the establishment and quantification of specifications for, and measurement of, characteristics of generic families of "standard" structural elements.
2. Fabrication and test of advanced, prototype elements as above utilizing materials and concepts developed in Studies 1-10.

The effort needed in each of these areas is summarized in the following sections.

DEVELOPMENTAL RESEARCH STUDY 12 - Characteristics of
Generic Families of Composite Structural
Elements for General Aviation Aircraft Applications

Objectives

Extension of results of Studies 7 and 8 to evaluate charact-
eristics of the better composite configurations, for various
thicknesses from minimum gage to at least 5 mm (0.2 in.) thickness,
and utilizing the medium-low density core materials of Study 6;
similarly, of post-formed curved elements utilizing the thermoplas-
tic resin development of Study 5, in various plate and shell struc-
tures and loading conditions. Confirmation of methods of analysis
developed, and validation of performance of selected elements over
the anticipated range of general aviation aircraft applications.
Development of analyses of fabrication costs as baseline values.

Approach

On the basis of trade-off studies as in Appendix B and con-
firming results in Studies 7 and 8, select 3 or 4 reinforcement
configurations for exploitation. (The results herein, as shown
in Appendix B, nominate the following configurations: $\pm15^{\circ}$, $\pm30^{\circ}/90^{\circ}$,
and if the weave inhibits in-plane shear, a $\pm45^{\circ}$ woven-fabric
reinforcement.) For these configurations obtain extensive data
on buckling and ultimate strength for plate and shell structures
under various loading conditions. Correlate results with pre-
dictions based on methodology developed in previous studies. An
important part of the approach is assessment during fabrication
of the costs involved, and of projections to production.

Desired Results

A defined family of advanced composite structural elements
useful either (1) for employment in general aviation aircraft
structures with minimal qualification problems, or (2) as a basis
for comparison with alternate materials and configurations.

Mechanics of Structural Components

Problem Areas

When structural elements are joined with other structural elements to form structural components, the interactions among them can result in response characteristics, particularly failure modes, substantially different from those for the individual element. Stiffener sections, such as hat- and zee- sections, are typical examples. Extensive studies of such interacting responses have been made for aluminum-alloy construction. The vastly different properties of composites, however, make those studies of little value for interpreting the responses of composite components. Modes of failure and failure loads require new evaluations for the composites with particular emphasis in the following areas:

(1) Determination of compressive buckling and failure modes as a function of proportions and reinforcement configuration of typical, basic stiffener shapes such as zees, channels, and I- or H- sections of a baseline composite material to establish a data base for extensions to and correlations with other materials.

(2) Studies of influences of attachment design on strengths of stiffened plate components to determine if attachment design criteria developed for the aluminum alloys can be extended to composites.

(3) Studies of failure modes associated with more complex assemblies than simple stiffener sections, including stiffened compression panels, shear webs, and multi-web beams in bending. Correlations insofar as possible with results for metals.

Research Efforts

In this research area, attention needs to be directed toward selection of loads and dimensions pertinent to those encountered in general aviation aircraft. Discussion of these constraints will be presented in later sections herein, - in Appendix C in particular. The minimum gage problem is a dominent one in this research area.

Summaries of appropriate research follow.

RESEARCH STUDY 13 - Evaluation of Compressive Buckling
and Failure Characteristics of Composite Stiffener Sections

Objectives

Development of basic data on the stability and failure of
composite stiffener sections. Correlations with analytical
predictions for buckling and semi-empirical methods for ulti-
mate strengths. Development of guidelines for balanced propor-
tions for stiffeners to avoid premature failure modes.

Approach

The approaches to this problem take guidance from the
corresponding researches in the aluminum alloys. The effort
needs to be both analytical and experimental. For buckling,
first the restraint coefficients analytically derived for buck-
ling of isotropic sections need to be extended to account for
the anisotropies of composites. Second, experiments to confirm
the predictions are required. For ultimate strengths, the experi-
ments come first to provide information on failure modes and
data for semi-empirical analyses, following in general the ap-
proaches found appropriate for the aluminum alloys.

Desired Results

Methodology for predicting compressive buckling and failure
stresses for typical stiffener sections. Development of generic
"standard" stiffener proportions. Baseline data for a family of
carbon fiber/epoxy composites for comparisons with other mater-
ials.

RESEARCH STUDY 14 - Development of Criteria for the Design of Joints for Composite Stiffened Components

Objectives

Development of data and criteria to facilitate the design of the attachment of stiffeners for composites stiffened plates and shells. Evaluations of various approaches to attachment design both for cost and performance.

Approach

The approaches to be used in this study require a combined effort, - analytical, developmental, and experimental. Analytically, extensions are needed to the methods derived for riveted joints on stiffened panels (see Appendices C and D). Developmentally, various joint configurations need to be explored both for ease of manufacture and performance. Finally, the experimental effort concentrates on the promising joint designs and provides data thereon for correlation with the analysis.

The experimental data are to define the relationship between joint characteristics and ultimate strength of the stiffened component. While results in riveted aluminum-alloy structures suggest that joints which produce the highest stresses are uneconomical, the possibility needs to be explored that bonded composite joints can be more effective.

Desired Results

Guidelines for effective, economic joint design for stiffened composite components.

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RESEARCH STUDY 15 - Properties of Advanced
Stiffener Sections for Composite Materials

Objectives

Determination of favorable proportions for advanced composite stiffener sections incorporating curved elements and medium-low density core materials, so that premature failure modes are avoided and potential performance gains due to curvature and/or materials are accessible.

Approach

The approach draws upon previous results from Studies 7 and 9 in which flat and curved plate elements have been evaluated. Those results provide a basis for the initial proportioning of stiffener sections designed to utilize either curvature, medium-low density core materials or both. Based on the results, sections having systematically varying proportions over the ranges of interest for application should be designed and tested, both as stiffeners alone and as attached to sheet, to determine buckling and failure modes and stresses. Emphasis should be toward the definition of boundaries to the proportions leading to the highest stresses.

Desired Results

Results should lead to the development of families of stiffener sections having demonstrated and defined properties which effectively exploit the properties of advanced composites.

RESEARCH STUDY 16 - Strength and Deflection

Characteristics of Multi-Web Beams

Objectives

Definization of the strength and stiffness properties of composite multi-web beams. Determination of the role the webs play in the stabilization of the compression cover and in delaying or precipitating ultimate failure. Development of baseline data for use in extensions to and correlations with advanced configurations.

Approach

This research area is the first in the recommended plan for which the effort is more pioneering than derived from previous research relating to aluminum-alloy construction. Accordingly, the approach is directly toward the goals of cost-effective composite construction. The initial effort should be the development of baseline data for representative composite laminate construction. A first series of beams utilizing the quasi-isotropic $\pm 30^{\circ}/90^{\circ}$ layup throughout, with a series of web spacings (as discussed in Appendix C) is recommended. Such beams offer most direct possibilities for correlations with predictions, and for establishing basic data for extension to higher performance reinforcement configurations. These beams should be tested for: (1) initial buckling strengths in both positive and negative bending, (2) stiffnesses in bending and torsion, and (3) ultimate strength in positive bending. The results should be used to provide fundaamental data on multi-web beam characteristics.

Desired Results

Basic data on composite multi-web beam response characteristics. Development of specimen design and test techniques for multi-web beams. Guidelines for advanced design and evaluation studies.

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DEVELOPMENTAL RESEARCH STUDY 17 - Cost-Effectiveness and Performance Evaluations

Objectives

Demonstrate and quantify on the structural component level, cost-effectiveness and performance gains associated with applications of technological advances in materials and structural elements.

Approach

Select a family of components suitable for use in actual general aircraft applications. Include prototype stiffened panels, multi-web beams, and fuselage shells. Utilizing developments from the foregoing research (including, for example: a medium-low density core material for laminate construction, curved plate construction for stiffener elements, and optimally oriented reinforcement configurations) design components to meet the requirements for those applications. Fabricate and test the resulting components, and evaluate the results. Evaluations should consider: (1) part count, (2) weight, and (3) structural performance.

Desired Results

Demonstrations of magnitudes of improvements achievable from technological advances in materials and structures.

DEVELOPMENTAL RESEARCH STUDY 18 – Fabrication and Test of Prototype Advanced Structures

Objectives

Evaluations of practical problems encountered in trans-
ition from structural components to actual aircraft structures.
Realistic determination of fabrication costs. Demonstration
and quantification of actual structural performance improvements
resulting from technological advances.

Approach

Select typical general aviation aircraft structures for a
range of applications and carry forward design and fabrication of
advanced composite structures suited for those applications.
Make full evaluations of those structures, including costs,
unanticipated characteristics, and measured structural per-
formance. Where appropriate, conduct competitive design ap-
proaches for comparisons with other materials including aluminum.
Ranges of applications investigated should include full (semi-
span) wings and stabilizers and fuselage sections, as discussed
in Appendix D.

Desired Results

Development of quantitative measures of potentials of ad-
vanced composites for general aviation aircraft structures.
Development of practical solutions to truly representative fabri-
cation problems and quantification of costs. Discovery of
errors and gaps in research results requiring further study.

RESEARCH STUDY 19 - Development of Integrated Aerodynamic -
Structural Design Computer Program for General Aviation
Advanced Composite Aircraft

Objectives

Provide a methodology for design of general aviation air-
craft in which trade-offs among design characteristics (gross
weight, wing area, etc., etc.) aerodynamic design, structural
design, and performance are readily effected and overall configur-
ation choices and design decisions can be made on an economical,
rational basis.

Approach

The approach is via the development and integration of com-
puter programs relating the various aspects of aircraft design,
in similar fashion to the way the BOZO program generated with
this study integrates wing structural design with loads and
configurations.

Desired Results

A comprehensive, automated procedure by which various air-
craft design approaches to a given set of performance require-
ments can be evaluated, and preferred ones selected.

Manufacturing Technology Development

In this section manufacturing technology development is recommended that would provide improvements in key, "generic" manufacturing areas of particular importance for general aviation aircraft. The overall development represents a multi-year effort, and it is comprised only of long-term projects, on the basis that short term ones are logically carried out by industry. The thrust is in part toward process development leading to improved products, and in part toward automation to help reduce composite manufacturing costs below those for metal structures.

MANUFACTURING DEVELOPMENT PROGRAM 1 – Development of Robotics to Reduce Hand Operations in Manufacturing of Composites for General Aviation Aircraft Applications

Objectives

Develop machinery and methodology for reducing or eliminating hand layup of advanced composite elements, components and structures, for cost reduction, improved accuracy of placement, and improved uniformity.

Approach

Approach should follow three paths: (1) refinement and/or development of filament winding equipment to make it better suited to general aviation aircraft structures including both fuselage and empennage components; (2) automated tape layup e-quipment suitable for minimum gage constructions; and (3) development of automated equipment for layup of woven-fabric rein-forced prepreg. Filament winding appears to be a cost-effective manufacturing method for fabricating the aft portions of fuselages on general aviation aircraft. Development should be pursued to extend the method to other portions of fuselages.

In order for tape laying up equipment to be appropriate for general aviation aircraft, it should incorporate the capability to lay up scrim reinforcement along with the tape to provide moderate biaxiality to the minimum thickness material. Along with the machine development should go studies of actually achieved properties of such minimum gage materials.

Woven-fabric layup machinery should be versatile to accommo-date to the varied uses for which fabric reinforcements are suited (see Appendix E). Important to the development is accur-acy of placement.

Desired Results

Versatile machinery capable of reducing to a minimum the amount of hand work used in fabrication.

MANUFACTURING DEVELOPMENT PROGRAM 2 - Development of Pultrusion
Equipment for General Aviation Aircraft Applications

Objectives

Develop machinery and methodology capable of manufacturing
economically the minimum thickness, high width/thickness ratio
structural elements of importance in general aviation aircraft.

Approach

The results of research on stiffener sections (Study 12)
should be used to provide guidelines for configurations and sizes
of pultrusions of interest for general aviation aircraft appli-
cations. Manufacturing limits should be determined in early
trials and compared with those desired, and compromises effected.
Emphasis should be upon minimum-gage constructions, and manu-
facturing development directed toward the closest practical
approaches to the limiting thicknesses in shapes, sizes, and
materials defined in preceding research.

Desired Results

Development of economical fabrication techniques for the
thin sections needed for general aviation aircraft.

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MANUFACTURING DEVELOPMENT PROGRAM 3 - Development of Fabric
Forming Machinery for Advanced Composites

Objectives

Develop versatile, large braiding machinery for forming
multi-directional reinforcements in advanced composites for gen-
eral aviation applications.

Approach

Adapt braiding technology to provide formed-fabric con-
structions for reinforcements. Braiding machinery has inherent
multi-directional capability but the fabrics are of limited
size. (It takes large machinery to braid a small fabric.) De-
velopment of the required large machines should extend the range
of usefulness of multi-directional reinforcements.

Desired Results

Capability to braid 3-D fabrics one meter (39 in.) wide in a
variety of advanced materials and weave configurations.

MANUFACTURING DEVELOPMENT PROGRAM 4 - Development of High Speed, Fully Automated NDI Equipment for Advanced Composites

Objectives

Develop fully automated thermography equipment for non-destructive inspection of composites capable of scanning at the rate of 50 m^2/hr. (500 ft^2/hr.).

Approach

Approach is primarily through advances in computer-scan analysis. Correlating studies will be required to develop and confirm sensitivities and reliability. Other advances in the areas of transition from laboratory instrument to commercial production operation will also be required.

Desired Results

Reduction in costs of quality control.

CONCLUDING DISCUSSION

A comprehensive research program directed toward the advancement of the technology of composites for general aviation aircraft applications has been presented, together with supporting studies, data, and evaluations. Obviously, some of the researches in the program offer greater promise for contributing to the desired advancement than others. Accordingly, discussion of relative merits and consequent directions for emphasis in terms of various criteria is desirable. This discussion will be separated into four categories:

(1) Researches yielding primarily <u>understanding</u>, such as analysis methodology, or experimental data. These researches are identified in Table 1.

(2) Researches yielding new materials and structural forms. These researches are identified in Table 2.

(3) Researches leading to improved structures and design methods. These are listed in Table 3.

(4) Researches in manufacturing technology. These are listed in Table 3.

In addition to the directions for emphasis developed in the various studies in each of these four categories, general guidelines also emerged applicable to the conduct of the research program as a whole. All of these are summarized in the following sections.

RESEARCHES LEADING TO IMPROVED TECHNICAL UNDERSTANDING

The researches in this category that appear to offer the possibility of providing the greatest contribution in the shortest time are in the area of strength property evaluations and failure criteria. The crux of the problem in this area is assoc-

iated with the in-plane weakness in the transverse direction
of advanced filamentary composites. Transverse failure mechanics
are ill defined; unduly conservative failure criteria are the
rule. If composites are to fulfill their promise, they must
be utilized fully. Failure criteria must not be compromising
due to lack of knowledge or understanding of strength character-
istics. (Most pertinent Studies are 1, 2, 3, 7, and 8, - see
Table 1).

For general aviation aircraft applications, the need for
economy leads to emphasis on reinforcement forms such as woven
fabrics that contribute to ease of manufacture. The fact,
brought out in the supporting studies herein, that such rein-
forcement offer promise of outstanding performance in many cases
(as in the reduction of interlaminar shear problems - Study 11)
bears out the need for early improved understanding of their
characteristics. Most importantly, if woven constructions can
be found to improve the in-plane transverse properties, substan-
tial performance improvements can become accessible, as noted
in the next section.

RESEARCH YIELDING NEW MATERIALS AND STRUCTURAL FORMS

Of the five studies recommended in this category, two are
materials related, two are structures related, and the fifth
is a combination of the two technologies (see Table 2). The
materials developments have the differing objectives of improve-
ment in ease of manufacture (Study 5 - Thermoplastic Development),
and improvements in performance (Study 6 - Medium-Low Density
Development), but Study 5 will not be a success if performance
is poor and Study 6 will fail if it's product is hard to handle.
Either Study can provide a substantial contribution if success-
ful, and no technical basis is apparent for preference between
them. (Appropriateness for NASA research is another matter to
be considered later).

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Study 10 (to demonstrate the performance of the material from Study 6) must, of course, be deferred pending the successful completion of Study 6.

The Study 9 (Effects of Curvature) offers more immediate pay-off than Study 4 (3-D Reinforcements), because background data (in aluminum) are available showing the effectiveness of curved plate elements. Study 9 has simplicity and immediacy of applicability to recommend it. In the long run, however, if properly optimized (optimized for ease of production as well as performance) 3-D reinforcements are achieved, Study 4 by eliminating through-the-thickness failures may be the more important Study.

RESEARCH LEADING TO IMPROVED STRUCTURES AND DESIGN METHODS

The first five of the studies in this category (see Table 3) represent a carrying forward into composites technology of the research done by NASA (and NACA) on aluminum-alloy structures. The task would indeed be difficult if that pioneering work had not been done, because of the added complexities of composite material anisotropy. Because of the variety of materials that can be generated from given constituents by changes in reinforcement configuration could still become enormous, however, selection of and concentration upon a few combinations at the outset is desirable. Further discussion of the selection process will be made in the final section of this Concluding Discussion.

Important to the introduction of composites to general aviation aircraft is the avoidance of the replication of effort that would occur if each manufacturer attempted to carry out the equivalents of Studies 12 - 16 for his own purposes. Centralization of the effort to provide working knowledge of the characteristics of composite structural elements suited for general aviation aircraft applications in the NASA is vital to the industry.

The final three studies in the category (see Table 3) relate to actual aircraft design and production problems. Whether these should be NASA or industry sponsored is perhaps still open to investigation. Of the three the most appropriate for performance by NASA is Study 19, The Development of an Integrated Aerodynamic - Structural Design Computer Program for General Aviation Aircraft.

RESEARCH IN MANUFACTURING TECHNOLOGY

Of the four Study areas in Manufacturing Technology identified herein (see Table 4) only the first (Development of Robotics to Reduce Hand Operations) addresses the key production fact of general aviation that, compared to military and transport aircraft which are built at the rate of dozens per year, general aviation aircraft are more apt to be in the hundreds per year range. The other three studies are all concerned with basic questions. For Study 2 the question is "How does one make pultrusion equipment for the thin gages needed for general aviation aircraft?" For Study 3 the question is "How does one make fabric forming machinery to generate economically the complex reinforcement patterns needed for advanced composites?" For Study 4 it is "How does one make high speed, fully automated NDI equipment?" In these latter three cases NASA's role to do the research to solve the basic questions is perhaps better defined than for the design of robotics. Especially in the definition of design requirements _ whether for pultrusions, complex fabrics, or NDI equipment, NASA studies can lead the way.

GENERAL IMPLICATIONS OF SUPPORTING STUDIES CONDUCTED IN THIS PROGRAM

Several general results emerged from the present program that have broader implications than to the design of research

studies. These include:

(1) Premiums for performance more than offset high raw-material costs of advanced composites.

(2) Research emphasis should center accordingly on medium-high modulus carbon fibers in epoxy resins as representative of the present most promising type of composite system for general aviation aircraft applications.

(3) Density reduction is one of the most powerful material characteristics offering potential for further increases in performance.

(4) Hybrids are not promising.

(5) Woven fabric reinforcements have potentials for
   (a) Economy
   (b) Minimum gage applications
   (c) Performance for plates and shells in compression and shear
   (d) Reduction in interlaminar shear stresses

(6) Thermoplastics have potentials for economy.
   (a) Need development in collaboration with analysis

(7) Curved-plate elements offer additional performance improvements

(8) Development of an integrated aerodynamic structural design computer program for the utilization of the advanced composites offers promise of maximum realization of true potentials for general aviation aircraft applications.

Table 1. – Research Leading to Improved Technical Understanding.

Study No.	Title
1.	Transverse Properties and Failure Mechanics of Unidirectionally Reinforced Composites.
2.	Comparison of Strengths and Failure Modes of Unidirectionally Reinforced and Woven-Fabric Reinforced Composites.
3.	Correlation of Composite Strengths with Matrix Properties.
7.	Flat Plate Compressive Properties of Advanced Composites.
8.	Comparison of Woven and Straight, Unidirectional Filamentary Reinforcements for Composite Flat Paltes.
11.	Studies of Interlaminar Shear Characteristics of Advanced Composite Laminates.

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Table 2. - Research Yielding New Materials and Structural Forms

Study No.   Title

4.   Multi-Directional Reinforcement for Enhanced Toughness.

5.   Reinforcement/Thermoplastic Resin Combinations.

6.   Medium-Low Density Core Materials.

9.   Evaluations of Effects of Curvature on Buckling and Ultimate
          Strangth of Composite Plate Elements.

10.   Evaluation of Buckling Performance of Laminated Composite
          Plates Having Medium-Low Density Cores.

Table 3. - Research Leading to Improved Structures and Design Methods.

Study No.   Title

12.   Characteristics of Generic Families of Composite Structural
       Elements for General Aviation Aircraft Applications.

13.   Evaluation of Compressive Buckling and Failure Characteristics
       of Composite Stiffener Sections.

14.   Development of Criteria for the Design of Joints for Composite
       Stiffened Components.

15.   Properties of Advanced Stiffener Sections for Composite Materials.

16.   Strength and Deflection Characteristics of Multi-Web Beams.

17.   Cost-Effectiveness and Performance Evaluations.

18.   Fabrication and Test of Prototype Advanced Structures.

19.   Development of Integrated Aerodynamic-Structural Design Computer
       Program for General Aviation Advanced Composite Aircraft.

Table 4. - Research in Manufacturing Technology

Study No.   Title

1.   Development of Robotics to Reduce Hand Operations in Manu-
        facturing of Composites for General Aviation Applications.

2.   Development of Pultrusion Equipment for General Aviation
        Aircraft Applications.

3.   Development of Fabric Forming Machinery for Advanced Composites.

4.   Development of High Speed, Fully Automated NDI Equipment for
        Advanced Composites.

Figure 1. Program Tasks for Study of Composites for General Aviation.
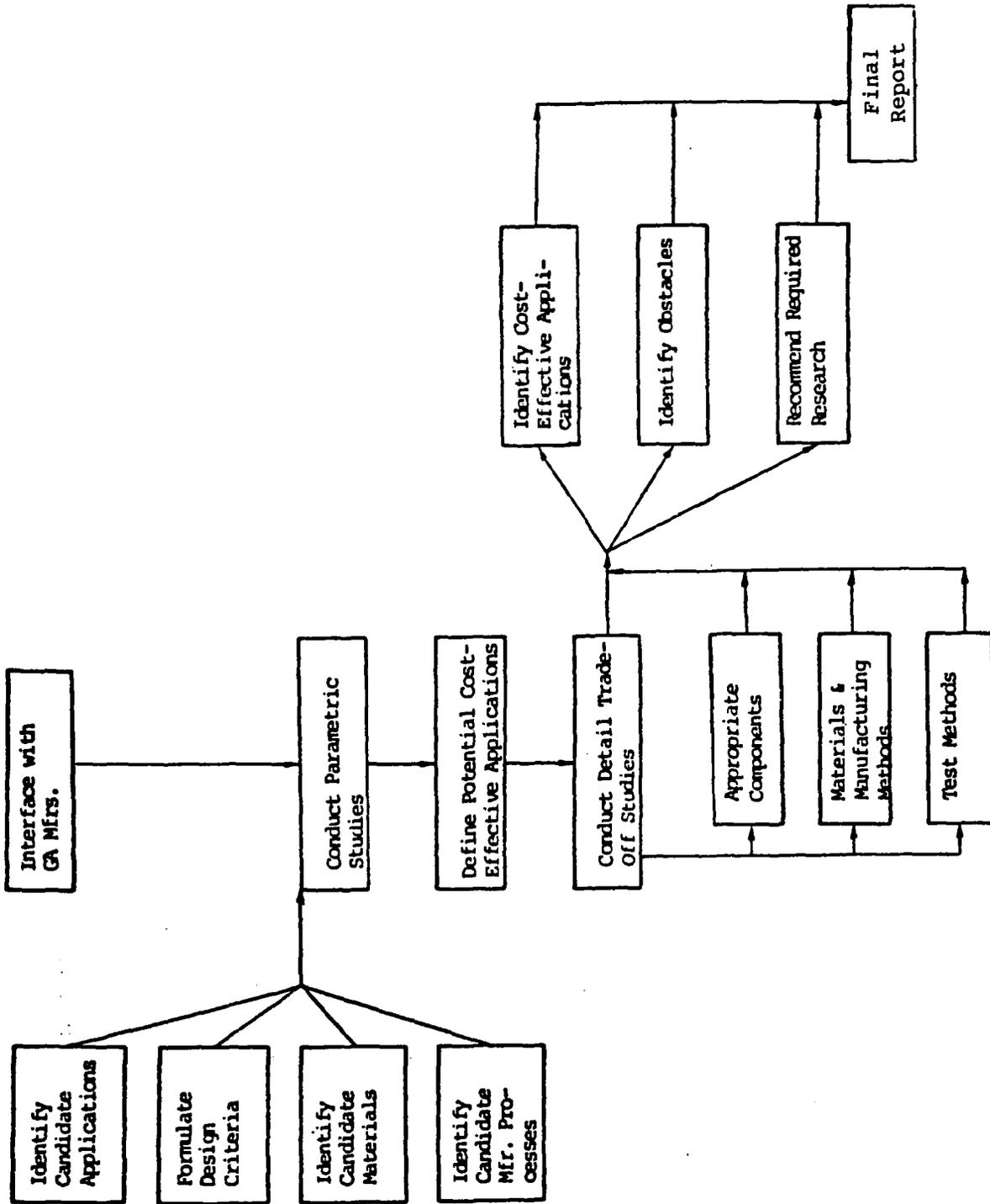
Figure 2. Principal Tasks of the Research Definition Study.

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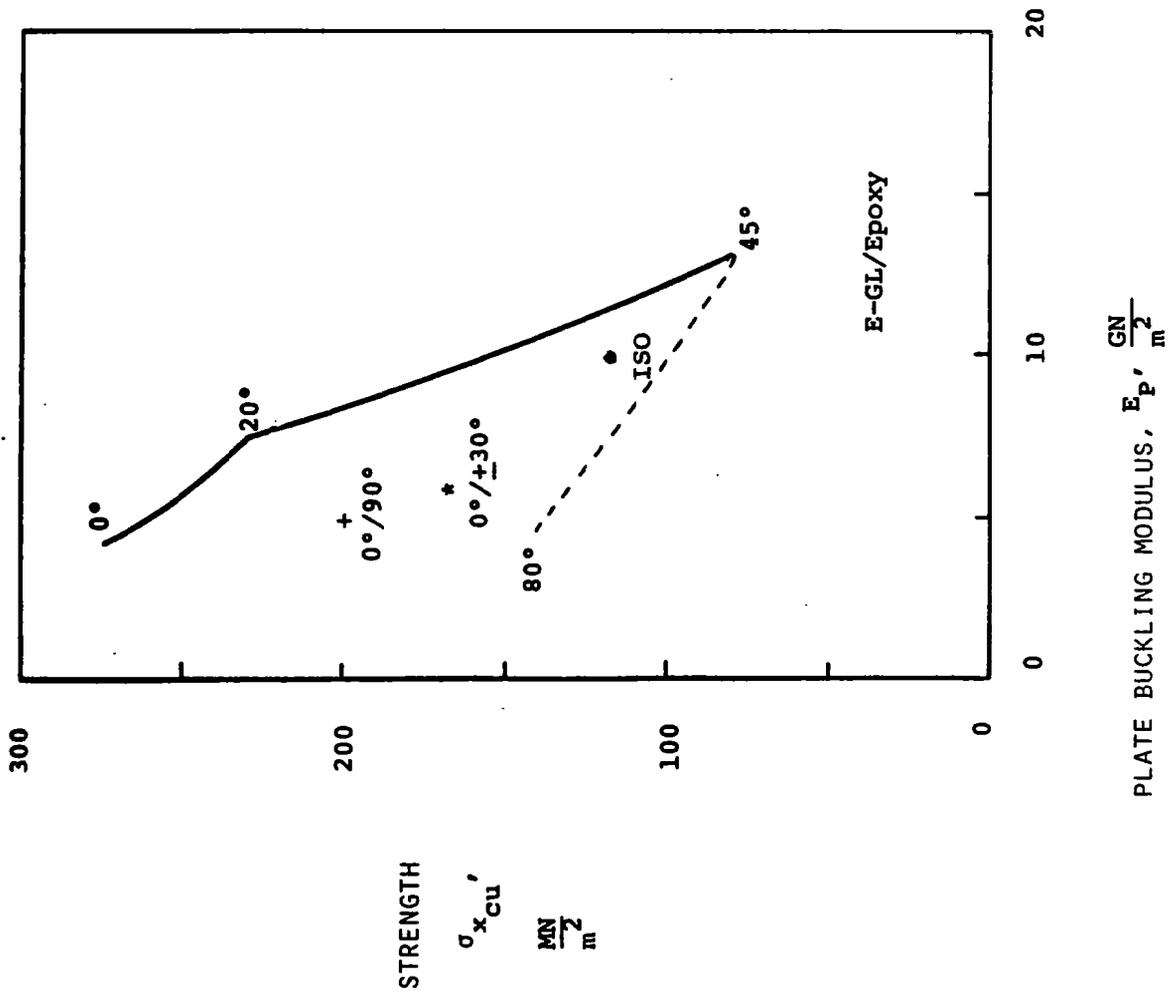
Figure 3. Illustrative Definition of Property Envelope for Material Selection.
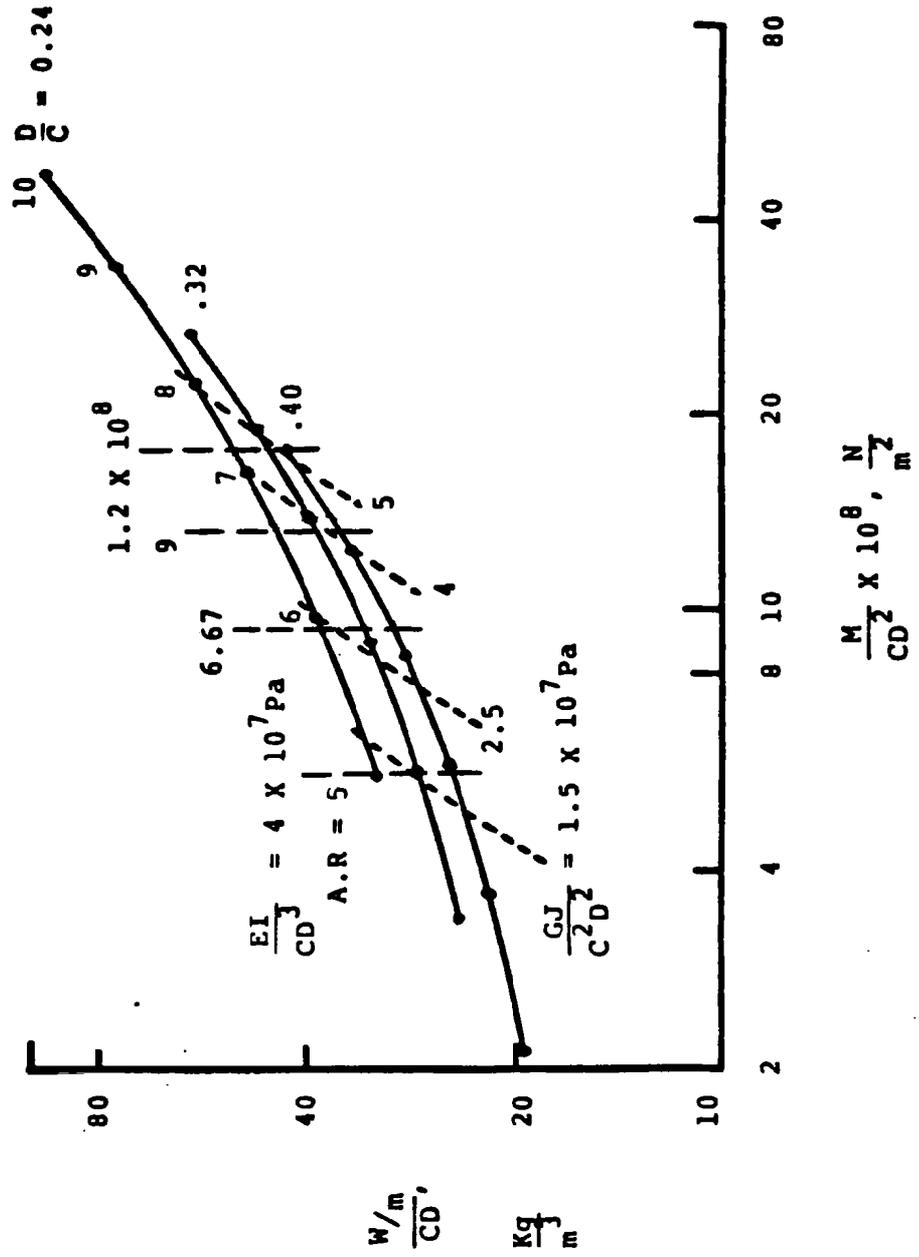
Figure 4. Illustrative Definition of Weight Plot Wing Design Parameters.

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APPENDIX A - DISCUSSIONS WITH MANUFACTURERS

Discussions were held with the following general aviation manufacturers regarding factors influencing the directions for research to be developed in this study.

- Beech Aircraft Corporation, Wichita, Kansas

- Cessna Aircraft Company, Wichita, Kansas

- Gulfstream American Corporation, Savannah, Georgia

- Piper Aircraft Company, Lakeland, Florida

- Gates Learjet, Wichita, Kansas

- Lear Fan, Reno, Nevada

These organizations are listed to acknowledge their assistance. These organizations have not reviewed this report prior to publication and they do not necessarily concur with the recommendations made herein.

APPENDIX B - STUDIES OF MATERIALS AND STRUCTURAL ELEMENTS

LIST OF SYMBOLS AND ABBREVIATIONS

b	plate width
r	shell radius of curvature
v_f	volume fraction reinforcement
D_x	bending stiffness in x-direction
D_y	bending stiffness in y-direction
D_{xy}	twisting stiffness about x-axis
E	Young's Modulus
E_L	extensional stiffness in longitudinal direction
E_P	modulus for use in plate buckling equation
E_S	modulus for use in shell buckling equation
E_T	extensional stiffness in transverse direction
G_{LT}, G_{xy}	in-plane shear stiffness
$I_{P_c}^*$	indicator number for plate efficiency in compression
$I_{P_s}^*$	indicator number for plate efficiency in shear
$I_{S_c}^*$	indicator number for shell efficiency in compression
K_P	constant in plate buckling equation
K_S	constant in shell buckling equation
N_x	compressive load per unit width
N_{xy}	shear load per unit width
W	weight
ε	extensional strain
μ_{xy}	Poisson's ratio for bending
ν_{LT}, ν_{TL}, &	
ν_{xy}, ν_{yx}	Poisson's ratio for extension

ρ	densitiy
σ	direct stress
τ	shear stress

Subscripts

c	compression
cu	compressive ultimate
f	fiber
L	longitudinal
m	matrix
P	plate
S	shell
s	shear
su	shear ultimate
T	transverse
t	tension
tu	tensile ultimate
x	x-direction (longitudinal)
y	y-direction (transverse)

Abbreviations

KEV	Kevlar
HI-C	high modulus carbon fiber
MED-C	medium-high modulus carbon fiber

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REVIEW OF STATE OF THE ART: SELECTION OF BASELINE MATERIAL

Supporting studies began with a review of the state of the art of composite materials appropriate for use in general aviation aircraft applications. The objectives of this review were to:

(1) Identify candidate fibers and matrix materials for exploitation for general aviation applications.

(2) Evaluate the state of knowledge of properties and mechanics of the materials and the relationship of that knowledge to that required if the materials are to be readily applicable.

(3) Select a baseline material (or baseline materials) as most suited for use in research and development of composite structures for general aviation aircraft.

Identification of Candidate Materials

Five candidate fiber materials and one matrix material were identified in the review as appropriate for detailed evaluations for general aviation aircraft applications. These materials meet the criteria of:

(1) Availability and manufacturability.

(2) Data on properties adequate for evaluations.

(3) Demonstrated performance potential competitive with aluminum.

The materials selected were:

E-Glass

S-Glass

Kevlar (49)

MED-C

HI-C

5208

where MED-C designates a medium-high modulus carbon fiber,
such as AS-1 or T-300

HI-C designates a high-modulus carbon fiber,
such as P-55 or HM-1000

and 5208 is an epoxy resin, similar to several available epoxies, but selected as a baseline here because more data appear to be available on its properties (especially those from ref. 1) than for its peers.

Properties of these fibers in the 5208 resin at a volume fraction v_f of 0.6 are listed in table B-1. These properties were used throughout for all calculations in this report.

Attention is called particularly to the values of transverse tensile strength σ_{Ttu} and transverse shear strength τ_{LTsu} used. These values, are somewhat lower than other values in the literature. The choice of these low values was made because of lack of reliable data. Basic properties of available composites for failure modes other than simple tension or compression along the filaments have not been adequately measured and reported.

Calculation of Element Properties

In the studies herein, as in normal use, the candidate materials are generally employed with reinforcements in more than one direction to provide enhanced transverse properties. These elements may be made up as laminates having unidirectional plies at various angles to one another or may have woven-fabric reinforcements. Herein, all properties for multi-directional reinforcements are calculated by the MSC CLAM computer program — a conventional laminate program for stiffnesses with various options for strength. For the purposes of this study, first ply failure was the strength criterion employed. It is a conservative criterion, but approximately equally conserva-

tive for all the materials considered. Its applicability for
special cases like fabrics will be individually discussed as
the cases are encountered.

Evaluations of Materials and Effects of Reinforcement Configurations

Various procedures were used to evaluate thoroughly the
properties of the candidate composites most pertinent to gener-
al aviation applications, beginning with simplistic considera-
tions of strengths, stiffnesses and densities and carrying through
to fairly detailed designs for complete wing structures. Because
of the variety of design conditions encountered in different
portions of aircraft structures, no one material or composite
reinforcement configuration was found to be universally superior.
As will be shown, however, the MED-C material and two or three
reinforcement configurations were found to have superiority for a
wide range of applications.

Preliminary Screening of Candidate Materials

First screening of the candidate materials were made as shown
in figures B-1 to B-3, representing angle-ply laminates with the
angle of reinforcement θ varying. Figure B-1 represents the
decrease in tensile strength of the materials with increasing
shear stiffness as θ varies from 0° to $\pm45^{\circ}$. Figures B-2 and
B-3 are corresponding curves for compressive strength vs. the
plate buckling modulus E_P and shell buckling modulus E_S, where

$$E_P = 1/2 \left[\frac{\sqrt{E_L E_T}}{1 - \sqrt{\nu_{LT} \, \nu_{TL}}} + 2 \, G_{LT} \right] \qquad (B-1)$$

$$E_S = \left[\frac{\sqrt{E_L E_T}}{1-\sqrt{\nu_{LT}\nu_{TL}}} \; (2\,G_{LT}) \right]^{1/2} \quad \text{or}$$

<div align="right">whichever is less</div>

$$= \left[\frac{E_L E_T}{1-\nu_{LT}\nu_{TL}} \right]^{1/2} \tag{B-2}$$

(as shown in reference 2, E_P and E_S replace the E for isotropic materials in the well-known buckling equations for plates and shells, thus

$$\text{for plates} \quad \sigma_{cr} = \frac{K_P \pi^2}{12} E_P \left(\frac{t}{b}\right)^2 \tag{B-3}$$

$$\text{and for shells)} \quad \sigma_{cr} = K_S E_S \left(\frac{t}{r}\right) \tag{B-4}$$

Also shown on figures B-1 to B-3 is a point representing the corresponding strength/weight/stiffness/buckling properties of 7075-T6 aluminum alloy.

This first simple screening reveals much about the characteristics available in these materials.

(1) They all have great tensile strength/weight potentials compared to aluminum.

(2) All except Kevlar have great compressive strength/ weight ratios compared to aluminum.

(3) The longitudinal strengths decrease rapidly as the transverse properties are increased by an increase in the angle of reinforcement from 0°. At ±45° (the bottom ends of the curves) all the composites have less strength than aluminum.

(4) Shear stiffnesses of the glass composites are less than aluminum for all reinforcement configurations. For configurations having the same longitudinal strength/density ratio as aluminum, all three other

composites have much greater shear stiffnesses than aluminum.

(5) Buckling properties of the angle-ply composites can be better than aluminum for plates but not for shells.

(6) Kevlar appears only slightly superior to aluminum in shear stiffness/weight properties and its buckling properties are correspondingly disappointing. Both carbon fiber composites show large superiorities to aluminum except for shell buckling.

Clearly from this first simple screening, it is evident that the direction for performance is via carbon rather than through the use of the other reinforcements. This conclusion will be verified in detail in the following sections.

Evaluations of Structural Efficiencies of Elements

In order to insure that the tentative conclusion reached in the preliminary screening that only the carbon fiber reinforcements offer substantial, across the board potentials for performance improvements, extensive calculations were made of structural efficiencies of all the candidate materials for the three structural element applications of prime interest for general aviation aircraft, namely:

(1) Flat plates in compression, as in wing and tail structures.

(2) Flat plates in shear, as in the webs of beams.

(3) Shells in compression, as in critical areas of the fuselage.

Calculations employed the following equations:

For plates in compression (ref. 3)

$$\frac{N_x}{b} = \frac{\pi^2}{b^3} \left\{ (K_P - 2) \left[\sqrt{D_x D_y} + D_{xy} + 2\nu_{xy} D_y \right] \right\} \tag{B-5}$$

For plates in shear (ref. 3)

$$\frac{N_{xy}}{b} = \frac{\pi^2}{b^3} \left\{ \left[3.4 + 1.9 \left(\frac{\mu_{xy} D_y + 2D_{xy}}{\sqrt{D_x D_y}} \right) \right] \left[D_x D_y^3 \right]^{1/4} \right\} \tag{B-6}$$

and for shells (ref. 2)

$$\frac{N_x}{r} = 2\sqrt{3} \left\{ \left[\sqrt{D_x D_y} \right] \left[2D_{xy} + \mu_{xy} D_y \right] \right\}^{1/2} \phi$$

$$\tag{B-7}$$

where

$$\phi = \left[\frac{2G_{xy} \left(1 + \sqrt{\nu_{xy} \nu_{yx}} \right)}{\sqrt{E_x E_y}} \right]^{1/2} \qquad \text{or 1, whichever is smaller}$$

Evaluations of Influence of Reinforcement
Configuration on Efficiency

Results of the first of these calculations are presented in figures B-4 to B-18 to show the effects of changes in reinforcement configuration on efficiency. Figures B-4 to B-8 show the results for flat plates in compression; figures B-9 to B-13 for flat plates in shear; and figures B-14 to B-18 for shells in compression. In all figures corresponding results for 7075-T6 aluminum alloy are shown for comparison.

In all these plots, the characteristic presentation is that of the unit weight of structural element ($\frac{W}{b}$ or $\frac{W}{r}$) against the loading intensity to be carried ($\frac{N_x}{b}$, $\frac{N_{xy}}{b}$, or $\frac{N_x}{r}$). As usual in structural efficiency analysis, the weight represents the value for failure at the given load, and the curves indicate minimum weights for the chosen proportions and materials.

To supplement the pictorial comparisons of figures B-4 to B-18, calculations were also made of the "indicator numbers" $I_{P_C}^*$, $I_{P_S}^*$, and $I_{S_C}^*$, for plates in compression and shear, and shells in compression, respectively. These indicator numbers are defined by the equations

$$I_{P_C}^* = \left[\frac{\left(\frac{N_x}{b}\right)^{2/3}}{\frac{W/cm^2}{b}} \right]_{I_{P_C}^*} , \quad \frac{\left(\frac{MN}{m^2}\right)^{2/3}}{\frac{gm}{cm^3}} , \quad \left[\left(\frac{in^5}{lb}\right)^{1/3} \right] \tag{B-8}$$

$$I_{P_S}^* = \left[\frac{\left(\frac{N_{xy}}{b}\right)^{2/3}}{\frac{W/cm^2}{b}} \right]_{I_{P_S}^*} , \quad \frac{\left(\frac{MN}{m^2}\right)^{2/3}}{\frac{gm}{cm^3}} , \quad \left[\left(\frac{in^5}{lb}\right)^{1/3} \right] \tag{B-9}$$

$$I\overset{*}{s}_C = \left[\frac{\left(\frac{N_x}{r}\right)^{2/3}}{\frac{W/cm^2}{b}} \right]_{I\overset{*}{s}_C} , \frac{\left(\frac{MN}{m^2}\right)^{2/3}}{\frac{gm}{cm^3}} , \left[\left(\frac{in^5}{lb}\right)^{1/3} \right] \qquad (B-10)$$

define the "corner" points on the curves of weight vs. loading of the structural efficiency plots ⌣ points representative of equality between buckling and failure stresses ⌣ as measures of maximum efficiency. Calculated values for the Indicator Numbers corresponding to curves of figures B-4 to B-18 are given in tables B-2 to B-4.

The following characterisitcs are brought out by the plots of figures B-4 to B-18.

(1) Optimum reinforcement configurations are essentially the same for all the materials. Specifically -

Flat Plates in Compression

For flat plates in compression (figs. B-4 to B-8) the optimum configuration varies in an orderly fashion with decreasing angle of reinforcement from $\pm 45^\circ$ for low loading intensities (failure by elastic buckling) to 0° for high loading intensities (failure at the ultimate compressive strength of the unidirectional material), - i.e. $\pm 45^\circ$ is most effective for buckling, 0° for strength. Small angle reinforcements such as $\pm 15^\circ$ exhibit nearly as high strengths as 0° but with improved buckling characteristics. The quasi-isotropic $\pm 30^\circ/90^\circ$ configuration has a good combination of strength and buckling properties. The maximum values of $I\overset{*}{p}_C$ for these four configurations are (from table B-2) all for the MED-C material, with the following values:

	$\dfrac{\left(\frac{N}{m^2}\right)^{2/3}}{\left(\frac{gm}{cm^3}\right)}$	$\left[\left(\frac{in^5}{lb}\right)^{1/3}\right]$
0°	1,287,000	[98,240]
$\pm 15^\circ$	1,301,000	[102,800]
$\pm 45^\circ$	563,300	[43,000]
$\pm 30^\circ/90^\circ$	1,120,000	[85,520]
compared to	589,200	[44,980] for the

7075-T6 aluminum alloy

Flat Plates in Shear

For flat plates in shear (figs. B-9 to B-13) the optimum
configuration varies from $\pm 60^\circ$ for buckling (c.f. plywood, ref.
6) to $\pm 45^\circ$ for strength. Small reinforcement angles are not
in contention, but the quasi-isotropic $\pm 30^\circ/90^\circ$ configuration
again exhibits a good balance of properties. The maximum values
of $I_{P_S}^*$ for these three configurations are (from table B-3) again
all for the MED-C material, with the following values:

	$\dfrac{\left(\frac{N}{m^2}\right)^{2/3}}{\left(\frac{gm}{cm^3}\right)}$	$\left[\left(\frac{in^5}{lb}\right)^{1/3}\right]$
$\pm 45^\circ$	1,054,000	[80,450]
$\pm 60^\circ$	1,033,000	[78,880]
$\pm 30^\circ/90^\circ$	833,400	[63,620]
compared to	504,200	[38,490] for the

7075-T6 aluminum alloy.

Shells in Compression

For shells in compression (figs. B-14 to B-18) the optimum
configuration is the quasi-isotropic $\pm 30^\circ/90^\circ$, with the $\pm 15^\circ$
a close competitor for buckling, – and the 0°, again with the
$\pm 15^\circ$ a close competitor, is the optimum for strength. The $\pm 45^\circ$
configuration in this case is not really in contention either

as regards strength or buckling. The values of $I_{S_C}^*$ for the three best configurations once more are found to be a maximum for the MED-C material, though here by only a small margin over the HI-C values. Numerical values (from table B-4).

	$\left(\frac{N}{m^2}\right)^{2/3} \Big/ \left(\frac{gm}{cm^3}\right)$	$\left[\left(\frac{in^5}{lb}\right)^{1/3}\right]$
±30°/90°	1,324,000	[101,100]
±15°	1,314,000	[100,300]
0°	856,500	[65,380] far exceed
the	755,200	[57,650]

values for 7075-T6 aluminum alloy

Evaluations of Material Independent of Configurations

The results of the calculations employing equations (B-5) - (B-7) were re-plotted in figures B-19 to B-22 to bring out more clearly the influence of material on efficiency as contrasted with the influence of reinforcement configuration. The implications as regards material of choice for further research and over-all development do not change but an important auxiliary area for emphasis is evidenced, as will be discussed later.

Flat Plates in Compression

Figure B-19 shows envelope curves of $\frac{W/cm^2}{b}$ ($\frac{W/in^2}{b}$) vs. $\frac{N_x}{b}$ for the various candidate materials representative of the minimum weights for the optimum configurations at all values of loading intensity. As shown, the weights of the glass reinforced composites are essentially identical to those for 7075-T6 aluminum alloy up to the load intensity at which their superior strength/density ratios can be utilized. At these high loadings, they do offer a potential for weight saving however, general aviation aircraft are characterized by low load intensities. The curves for Kevlar, MED-C and HI-C materials all exhibit potentials

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for substantial weight saving, the Kevlar at the lower loadings, and the carbons over the entire range of loadings. The curves for the carbon fiber reinforced materials are everywhere below those for Kevlar, and that for the HI-C material is slightly lower than that for the MED-C at low loads.

For plates in compression, the greatest potential for improvement is seen to be through the use of carbon fiber reinforcements.

Flat Plates in Shear

Figure B-20 presents similar curves to those in B-19 but for shear instead of compression loadings. The implications are identical, except that the glasses appear even less competitive, and the HI-C fiber has a slightly greater range of loads for which it is superior to the MED-C than in compression.

Shells in Compression

For shell elements in compression (figures B-21 and B-22) the isotropy of aluminum makes it hard to surpass (as noted in ref. 3, isotropic reinforcements configurations appear to yield minimum-weight for shell buckling). The specific points representing quasi-isotropic reinforcements are plotted in figure B-22, and they provide a summary picture of the relative merits of the materials for shells in compression, specifically:

   (1)   The E-Glass and S-Glass materials are not competitive with 7075-T6.

   (2)   The Kevlar reinforcements have the potential to provide shell elements somewhat (∿20%) lighter than 7075-T6 at medium-low load intensities.

   (3)   The carbon reinforcements offer potentials for approximately twice the weight savings over 7075-T6 as

Kevlar. The HI-C material is slightly the better at the lower loading. The MED-C is better at the higher loadings.

Selection of Baseline Material

In none of the evaluations described above have either E-Glass or S-Glass shown comparable potentials to the other three candidate materials. Kevlar has shown a potential for improvement compared to aluminum-alloy, - a potential that is substantial for plates at low loadings. The carbon fibers surpass the Kevlar in potential for all types of elements, all reinforcement configurations, and all loading conditions. Clearly, the baseline material for emphasis for research and development for general aviation aircraft should utilize the carbon fibers, - if their costs are not prohibitive. (It will be shown elsewhere that the fiber costs per se are not prohibitive.)

The choice between a medium modulus and high modulus fiber is not so clearly defined on a quantitative basis. In general, the differences in performance found between them in the several evaluations were not substantial. The HI-C was better at low loadings, the MED-C at high. For intermediate loading intensities (importantly, loading intensities such that efficient aluminum-alloy construction would not be stressed to the yield point) the two fibers are essentially identical in performance.

Because of this near identity, other bases than efficient performance become dominant. Here, the MED-C prevails. It is easier to handle, particularly to weave. It has had substantially more development emphasis to date, covering many areas of qualification. It has the practical merit of a higher strain to failure than the HI-C fiber, with various associated favorable implications regarding strain compatibilities. The MED-C is the material of choice for exploitation as the baseline material for general aviation aircraft applications.

HYBRID ELEMENTS

In addition to the basic reinforcements evaluated in the preceding sections, possibilities of two types of hybrids need to be explored. These are:

(1) Hybrid configurations having outer plies (faces) of one type of reinforcement configuration and a central ply or plies of a difference configuration. For example, $\pm 45^\circ$ faces on a 0° central ply (relative thicknesses optimized) would appear to combine the buckling resistance properties of the $\pm 45^\circ$ configuration with the strength properties of the unidirectionals.

(2) Hybrid materials, again having differing characteristics in different directions and/or laminae. A similar example to the one cited above would put Kevlar faces on a glass core, or HI-C faces on a MED-C core. A priori, the possibilities appear attractive.

Unfortunately, hybridization turns out to be disappointing. The results of extensive calculations of the properties of hybrids are presented in summary fashion in figures B-23 to B-25.

Hybrid Configurations

For the angle-ply faces on a 0° core, the results are shown in figure B-23 as the short curves at the right side of the plot. The configurational variation along each of these curves is from 95% of the overall laminate thickness in the 0° core (the top, right ends of the curves) to 100° of the material as the face configuration. For comparison, the envelope curve from figure B-19 is reproduced as the dashed curve. While there is some slight penetration of the 15°, 45°, and 60° curves below the envelope for simple all-cross-ply construction, as can

be seen, it is not substantial, and hardly worthy of substantial
developmental emphasis. Similar results (not shown) were found
for the other four candidate materials. For plates in shear,
the best configuration is $\pm 45^{\circ}$ for strength and $\pm 60^{\circ}$ for buck-
ling, and the differences between them are so insubstantial
that a hybrid combination of the two is not warranted. For
shells, the merit of the quasi-isotropic configuration can not
be surpassed by any hybrid, as illustrated in figure B-24.

Hybrid Materials

For more buckle resistant material faces on higher material
strength cores the results are as summarized in figure B-25
for Kevlar faces on an E-Glass core and HI-C faces on a MED-C
core. In both cases the hybrid does provide a region of effici-
ent performance, to fill the gap between the regions of effici-
ent performance of the individual materials. In the limited
region, however, at most, the gain is small.

Similar results were found for a limited number of other
hybrid combinations. Only a limited number were evaluated be-
cause the results were consistently unencouraging.

MATRIX PROPERTIES

Dependence of Strength on Matrix Properties

While not as dominant a factor as for the reinforcement,
matrix properties are also important to composite performance.
If the failure mode is in the matrix material, the laminate
strength can be enhanced by increasing matrix strength or decreas-
ing matrix modulus, as illustrated in figure B-26. Recent studies
have expanded the analysis of the influence of the matrix to
provide a detailed picture of the interacting factors in-
volved. Figure B-27 illustrates a typical case, again, point-

ing out the merit of decreasing matrix stiffness for increased
tensile properties of the composite (not necessarily the same
for compression). The significance here is that methodology
is developing to guide matrix property development.

Medium-Low Density Core Material

Matrix materials can be formulated not only for strength
and stiffness characteristics but also for density. A micro-
balloon filled syntactic epoxy material is available with
approximately one-half the density of the usual epoxies.
Foamed plastics are well-known low density materials of gen-
erally poor structural properties. The obvious question
arises regarding the quantitative merit of matrix density
reduction.

The studies of structural efficiencies of elements were
extended to provide answers to this question. The model used
was the same as that of the 0° core hybrid with the 0° core
replaced with a hypothetical epoxy having an arbitrary density
equal to 1, 1/2, or 1/4 that of normal epoxy density. The mod-
uli of these hypothetical materials were assumed to be 1, 1/5,
and 1/10 those of the usual epoxy, respectively, and the strains
to failure were assumed adequate so that the core did not fail
below collapse load for the composite. For simplicity, quasi-
isotropic face plies were used for this model, and varied from
5% to 66.7% of the total material thickness and the plate buck-
ling efficiencies calculated. Results of the calculations are
shown in figure B-28.

Results show that even the 1/2 density core produces apprec-
iable increases in efficiencies. Gains for the 1/4 density
core (approximately the density of a light-weight mahogany)
are equal to or greater than that for the MED-C composite com-
pared to 7075-T6 aluminum-alloy. Density changes of this mag-

nitude are indeed effective. Their possible use for part count
reduction for general aviation aircraft structures will be con-
sidered later.

Load Intensities Encountered in General Aviation Aircraft

Because the magnitude of the load intensities encountered
in general aviation aircraft structural applications is of prime
importance to the development of appropriate structural and
material approaches, a brief summary of their magnitudes is pre-
sented in figure B-29 with minimal explanation here. Detailed
discussions of load and stiffness requirements will be given
later in Appendix D.

Figure B-29 indicates that load intensities encountered
in plate elements representative of the compressive covers of
box beams for the root sections of the wings of representative
general aviation aircraft. They all encounter load intensities
in the range 1-2 $\frac{MN}{m^2}$ (150-300 psi) as shown; by reference, more
than one decade below those that would utilize 7075-T6 aluminum
alloy to its yield stress. The importance of these low
loadings (even though they are the wing root loadings which
are among the highest in the aircraft) will be emphasized in-
creasingly in the following sections.

CANDIDATE ELEMENTS FOR RESEARCH AND DEVELOPMENT

As a result of the many studies of the effectiveness of
various materials and reinforcement configurations, a few were
found to stand out as logical candidates for exploitation via
research and development, particularly in the light of the low
loading intensities of the general aviation aircraft applica-
tions. These are identified as:

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Materials for Candidate Plate and Shell Elements

A medium-high modulus carbon fiber reinforced composite was identified as the "baseline material" for research and development, and it remains the baseline material for studies at the element level. In view of the low loadings for general aviation applications, however, the HI-C fibers should not be totally neglected. They showed most promise for lightly loaded shells in compression (see figures B-18, B-21, and B-22), and should be considered for shell elements along with the MED-C.

Configurations for Candidate Plate and Shell Elements

Three reinforcement configurations stand out:
(1) $\pm15^\circ$ for maximum strength with moderate transverse strength properties.
(2) $\pm45^\circ$ for maximum shear strength and nearly maximum plate shear buckling resistance.
(2A) $\pm45x^\circ$, if woven constructions can be found to inhibit in-plane shear failures, - for maximum plate compressive buckling resistance with good strength properties.
(3) $\pm30^\circ/90^\circ$ (or $0^\circ/\pm45^\circ/90^\circ$) for maximum shell buckling resistance with good strength, and for near maximum plate buckling resistance in both compression and shear, again with good strength properties.

As shown previously, these same configurations apply regardless of the material used.

To provide a measure of the potential for improvement, calculations were made of the plate and shell efficiencies of these configurations incorporating the medium-low density core material previously considered with the MED-C reinforcements. Results of these calculations are shown in figures B-30 to B-32.

Figure B-30 shows the outstanding potential of the $\pm 45°$
configuration for plate compressive applications if the in-plane
shear failure can be inhibited by a woven-fabric reinforcement.
The maximum weight saving potential for such composite elements
compared to solid 7075-T6 aluminum alloy as shown is 75% at a
value of the compressive load index $\frac{N_x}{b} = 2.75 \frac{MN}{m^2}$ (400 psi).
The $\pm 30°/90°$ configuration provides (without the question about
inhibition of in-plane failure) nearly as much potential weight
saving (73% @ $\frac{N_x}{b} = 2.07 \frac{MN}{m^2}$ [300 psi]).

Figure B-31 shows the corresponding potential for plate
elements in shear. Here the $\pm 45°$ configuration is clearly sup-
erior overall (and the question of in-plane shear failure does
not arise), but at low loadings, the difference in weight saving
potential compared to that for the $\pm 30°/90°$ configuration is
negligible (again, approximately 75% over 7075-T6, - at $\frac{N_{xy}}{b} =$
$0.69 \frac{MN}{m^2}$ [100 psi]).

For shell elements in compression (fig. B-32) the quasi-
isotropic configuration is supreme, offering a potential weight
saving of 58% at an $\frac{N_x}{r} = 1.73 \frac{MN}{m^2}$ (250 psi). For shells, the
$\pm 15°$ configuration is superior to the $\pm 45°$ configuration.

Woven-Fabric Reinforcements

Woven fabric reinforcements have additional performance
potentialities for general aviation aircraft applications to
those cited for plate and shell elements. (Their merit for
ease of handling will be discussed in Appendix E.) By and large,
however, these potentialities have not been quantitatively e-
valuated, as they need to be to provide guidelines for research.
Two aspects of their performance potential were accordingly
explored in the present studies: (1) the magnitude of the sacri-
fice in performance associated with the yarn out-of-straightness
(crimp) in a woven configuration, and (2) the magnitude of the
reduction in interlaminar shear stresses for laminates composed

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of woven-fabric reinforced plies compared to those having uni-
directionally reinforced plies.

Effects of Crimp

The model used to evaluate the effect of crimp utilized
adequately twisted filament bundles to provide circular yarn
cross-sections throughout the weave and exaggerate the magni-
tude of crimp induced in weaving. The method of analysis used
the same approach developed in reference 5 (since programmed
as MSC XCAP code), sub-dividing the yarns into segments of ap-
propriate directionalities wherever crimped.

Results are shown in figure B-33. Plotted for various yarn
spacings are the ratios of stiffnesses for the various moduli
of the composite if reinforced with simple biaxially woven-fabric
compared to cross-ply $0^o/90^o$ laminates of unidirectionally rein-
forced plies of the same volume fraction. The plot shows that
for yarn spacings four or more times the diameter the composite
stiffnesses for wovens and unidirectionals are not very different.
This ratio of four, representative of the ply thickness to yarn
spacing ratio of fabrics with untwisted yarns is generally ex-
ceeded in composite reinforcements.

Interlaminar Shears

The model used to evaluate interlaminar shear was essenti-
ally that of reference 6 in which the interlaminar stresses
reach a peak near the edge of a laminate stretched in tension due
to the "lazy tongs" action between plies. The analysis used
was an in-house finite element code.

Results are shown in figure B-34 for $0^o/90^o$ unidirectionally
reinforced, and plain weave reinforced laminates having their
principal axes at the indicated small angles to the direction
of extension. As shown, the interlaminar stresses are substanti-

ally less for the woven reinforcements.

Not shown are results for $\pm30°/90°$ triaxially woven-fabric reinforcements, inasmuch as the isotropy thereof eliminates the interlaminar stresses altogether.

Candidate Woven-Fabric Reinforcement Configurations

Both conventional biaxially woven and triaxially woven-fabric reinforcements must be considered candidates for research and development for general aviation aircraft applications. The biaxial fabrics will be most applicable if the woven configuration can be found to inhibit the in-plane shear mode of failure; even if no such inhibition can be found, numerous applications for $0°/90°$ reinforcement in a single ply exist. Not only does the fact that biaxial properties are developed in one ply reduce the interlaminar shear stresses as shown, the fact that only one ply is required is important for the light load intensities of general aviation aircraft applications, as will be emphasized in subsequent sections.

Comments on the suitability of biaxial fabrics apply more emphatically to triaxials. The quasi-isotropic $\pm30°/90°$ configuation was shown to have the widest range of efficient application of any considered, - in many cases by a substantial margin. The fact that it combines the three plies required with unidirectionals for isotropy into one is again desirable for general aviation aircraft applications.

Multi-Directional Reinforcements

The extension from triaxial (three directional, in-plane fabrics) to three-dimensional (multi-directional) fabrics is primarily exemplified in current technology by carbon-carbon composites for high temperature applications. Problems in-

volved with extensions to the third dimension are threefold:
(1) problems associated with manufacture of the 3-D reinforce-
ment itself, (2) problems associated with the formability of
a 3-D prepreg, and (3) problems associated with the structural
performance of the three-dimensionally reinforced composite.

In this section only the performance aspects are considered.
(Manufacturing aspects will be covered in Appendix E.) The
difficulty with performance is that any reinforcements running
through-the-thickness subtract from the volume fraction left
in-plane, if the total volume fraction reinforcement is main-
tained constant, - and actually generally interferes with the
in-plane yarns so that the total volume fraction reinforcement
is less than for 2-D reinforcement. The magnitude of the per-
formance reduction due to reduction of in-plane reinforcement
is suggested by figure B-35.

In figure B-35 are shown the stiffnesses of a family of MED-C
reinforced composites all having a total volume fraction rein-
forcement equal to 0.6. If that reinforcement is all in the load
direction (0°), the stiffness is 132 $\frac{GN}{m^2}$ (19.2 msi). For equal
stiffness two ways, the stiffness is approximately halved, to
72 $\frac{GN}{m^2}$ (10.4 msi). (Less than 7075-T6.) For in-plane isotropy
E = 51 $\frac{GN}{m^2}$ (7.4 msi), and for 3-D isotropy E = 30 $\frac{GN}{m^2}$ (4.3 msi),
substantially less than half that of aluminum.

The need for 3-D isotropy is an extreme case, but the warn-
ing of figure B-35 is clear, - namely, keep the volume fraction
in the through-the-thickness direction to the minimum necessary
to provide through-the-thickness properties.

As discussed in Appendix E, an attractive approach to 3-D
reinforcement is through the addition of through-the-thickness
running yarns in conventional multi-layer weaving.

DEVELOPMENT OF DIRECTIONS FOR RESEARCH AND DEVELOPMENT ON MATERIALS AND STRUCTURAL ELEMENTS

The studies of materials and structural elements reported in this Appendix have generated the following directions for research and development -

(1) The most appropriate material combination to use as a baseline for research and development of composites for general aviation aircraft is a medium-high modulus carbon fiber in an epoxy resin.

(2) Matrix material development should be directed by extended studies to correlate the failure of the composite with the matrix material properties.

(3) Prime areas for research are studies of transverse properties and failure mechanics of both straight, unidirectionally reinforced, and woven-fabric reinforced composites.

(4) Three reinforcement configurations stand out as offering best performance:

   (1) The quasi-isotropic ($\pm 30^\circ/90^\circ$) configuration; best overall.

   (2) The $\pm 45^\circ$ configuration; best for shear; if in-plane shear failures inhibited as by an appropriate woven construction, best for plate elements.

   (3) The $\pm 15^\circ$ configuration; for near unidirectional strength with moderate transverse properties.

(5) Develop a medium-low density core material for plate and shell elements.

(6) Woven-fabric reinforcements, both biaxial and tri-axial, are appropriate for development for minimum gage applications.

Table B-1. Properties of the Five Candidate Materials as Unidirectional Composites with $v_f = 0.6$, as Used for Studies in this Report, - and Corresponding 7075-T6 Aluminum Alloy Properties.

MATERIAL		E-GLASS	S-GLASS	KEVLAR	MED-C	HI-C	7075-T6
Density	g/cm^3 (pci)	1.97 (0.071)	1.94 (0.070)	1.30 (0.047)	1.47 (0.053)	1.63 (0.059)	2.80 (0.101)
E_L	GN/m^2 (msi)	45. (6.5)	52. (7.5)	76. (11.)	132. (19.2)	229. (33.2)	72.35 (10.5)
E_T	GN/m^2 (msi)	12. (1.7)	13.8 (2.0)	5.5 (0.8)	10.3 (1.5)	6.9 (1.0)	72.35 (10.5)
G_{LT}	GN/m^2 (msi)	4.4 (0.64)	4.5 (0.65)	2.1 (0.3)	4.3 (0.62)	5.5 (0.8)	27.5 (4.0)
ν_{LT}		0.25	0.25	0.34	0.25	0.25	0.315
σ_{Ltu}	MN/m^2 (ksi)	1030. (150.)	1380. (200.)	1380. (200.)	1240. (180.)	1170. (170.)	480. (70.)
σ_{Ttu}	MN/m^2 (ksi)	41. (6.)	41. (6.)	41. (6.)	41. (6.)	41. (6.)	480. (70.)
σ_{Lcu}	MN/m^2 (ksi)	550. (80.)	660. (96.)	280. (40.)	830. (120.)	690. (100.)	410. (60.)
σ_{Tcu}	MN/m^2 (ksi)	138. (20.)	138. (20.)	138. (20.)	138. (20.)	138. (20.)	410. (60.)
τ_{LTsu}	MN/m^2 (ksi)	55. (8.)	55. (8.)	55. (8.)	55. (8.)	55. (8.)	410. (30.)

Table B-2. Indicator Numbers for Plate Compressive Efficiencies of Five Candidate Materials Used for Studies in this Report, and Corresponding Values for 7075-T6 Aluminum Alloy.

$$\left(\frac{N}{m^2}\right)^{2/3} \Big/ \frac{gm}{cm^3} , \left[\left(\frac{in^5}{lb}\right)^{1/3}\right]$$

REINFORCEMENT CONFIGURATION	E-GLASS	S-GLASS	KEVLAR	MED-C	HI-C	7075-T6
0°	730,100. [55,730.]	810,400. [61,860.]	758,800. [57,920.]	1,287,000. [98,240.]	957,200. [73,070.]	589,200. [44,980.]
±15°	701,200. [53,530.]	816,700. [62,430.]	787,000. [60,080.]	1,301,000. [99,280.]	1,028,000. [78,460.]	
±30°	482,100. [36,800.]	507,800. [38,760.]	816,900. [62,360.]	837,200. [63,910.]	795,000. [60,690.]	
±45°	353,000. [26,950.]	361,600. [27,600.]	578,800. [44,180.]	563,300. [43,000.]	550,000. [41,980.]	
±60°	377,400. [28,810.]	385,900. [29,460.]	586,700. [44,790.]	560,000. [42,740.]	481,000. [36,720.]	
0°/90°	599,300. [45,750.]	657,800. [50,210.]	610,600. [46,610.]	1,028,000. [78,490.]	800,000. [61,050.]	
±30°/90°	520,600. [39,740.]	560,000. [42,730.]	659,800. [50,370.]	1,120,000. [85,520.]	865,300. [66,050.]	

Table B-3. Indicator Numbers for Plate Shear Efficiencies of Five Candidate Materials Used for Studies in this Report, and Corresponding Values for 7075-T6 Aluminum Alloy.

$$\left(\frac{N}{m^2}\right)^{2/3} \Big/ \frac{gm}{cm^3}, \quad \left[\frac{(in^5)^{1/3}}{\left(\frac{lb}{}\right)}\right]$$

REINFORCEMENT CONFIGURATION	E-GLASS	S-GLASS	KEVLAR	MED-C	HI-C	7075-T6
0°	233,600. [17,830.]	235,700. [17,990.]	326,600. [24,930.]	320,700. [24,480.]	280,500. [21,410.]	504,200. [38,490.]
±15°	290,300. [22,160.]	325,400. [24,840.]	650,500. [49,660.]	600,900. [45,870.]	552,400. [42,170.]	
±30°	337,100. [25,730.]	363,100. [27,720.]	586,500. [44,770.]	914,400. [69,800.]	762,700. [58,220.]	
±45°	382,000. [29,160.]	390,600. [29,820.]	680,200. [51,920.]	1,054,000. [80,450.]	901,100. [68,790.]	
±60°	358,000. [27,330.]	388,400. [29,650.]	664,700. [50,740.]	1,033,000. [78,880.]	883,500. [67,440.]	
0°/90°	251,500. [19,200.]	256,600. [19,590.]	403,600. [30,810.]	393,100. [30,010.]	382,300. [29,180.]	
±30°/90°	311,000. [23,740.]	366,900. [28,010.]	533,300. [40,710.]	833,400. [63,620.]	702,600. [53,630.]	

Table B-4. Indicator Numbers for Shell Compressive Efficiencies of Five Candidate Materials Used for Studies in this Report, and Corresponding Values for 7075-T6 Aluminum Alloy.

$$\left(\frac{N}{m^2}\right)^{2/3} \Bigg/ \frac{gm}{cm^3} \quad \left[\left(\frac{in^5}{lb}\right)^{1/3}\right]$$

| REINFORCEMENT CONFIGURATION	E-GLASS	S-GLASS	KEVLAR	MED-C	HI-C	7075-T6
0°	673,700. [51,430.]	730,300. [55,750.]	704,800. [53,800.]	856,500. [65,380.]	922,400. [70,410.]	755,200. [57,650.]
$\pm 15^\circ$	717,200. [54,750.]	785,700. [59,980.]	857,800. [65,480.]	1,314,000. [100,300.]	1,107,000. [84,500.]	
$\pm 30^\circ$	581,400. [44,380.]	550,700. [42,040.]	781,700. [59,670.]	881,100. [67,260.]	850,200. [64,990.]	
$\pm 45^\circ$	404,500. [30,880.]	415,000. [31,680]	578,200. [44,137]	634,000. [48,400.]	643,600. [49,130.]	
$\pm 60^\circ$	445,300. [33,990.]	458,800. [35,020.]	627,000. [47,860.]	673,700. [51,430.]	609,200. [46,500.]	
$0^\circ/90^\circ$	597,900. [45,640.]	645,200. [49,250.]	637,300. [48,650.]	1,033,000. [78,840.]	882,200. [67,340.]	
$\pm 30^\circ/90^\circ$	598,000. [45,650.]	645,700. [49,290.]	853,700. [65,170.]	1,324,000. [101,100.]	1,199,000. [91,560.]	
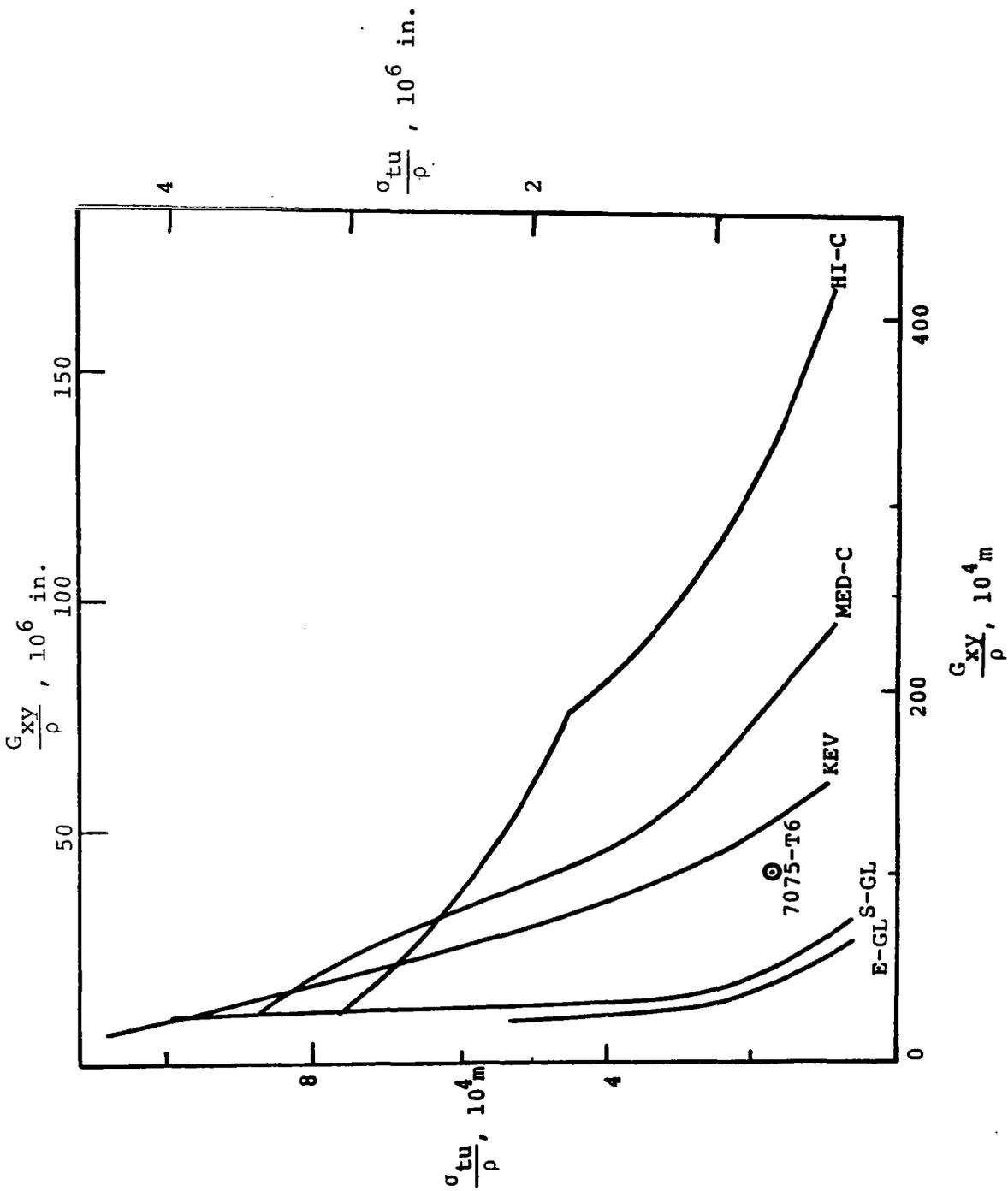
Figure B-1. Tensile Strength/Density and Shear Stiffness/Density Ratios for Candidate Composite Materials Compared to 7075-T6 Aluminum Alloy.
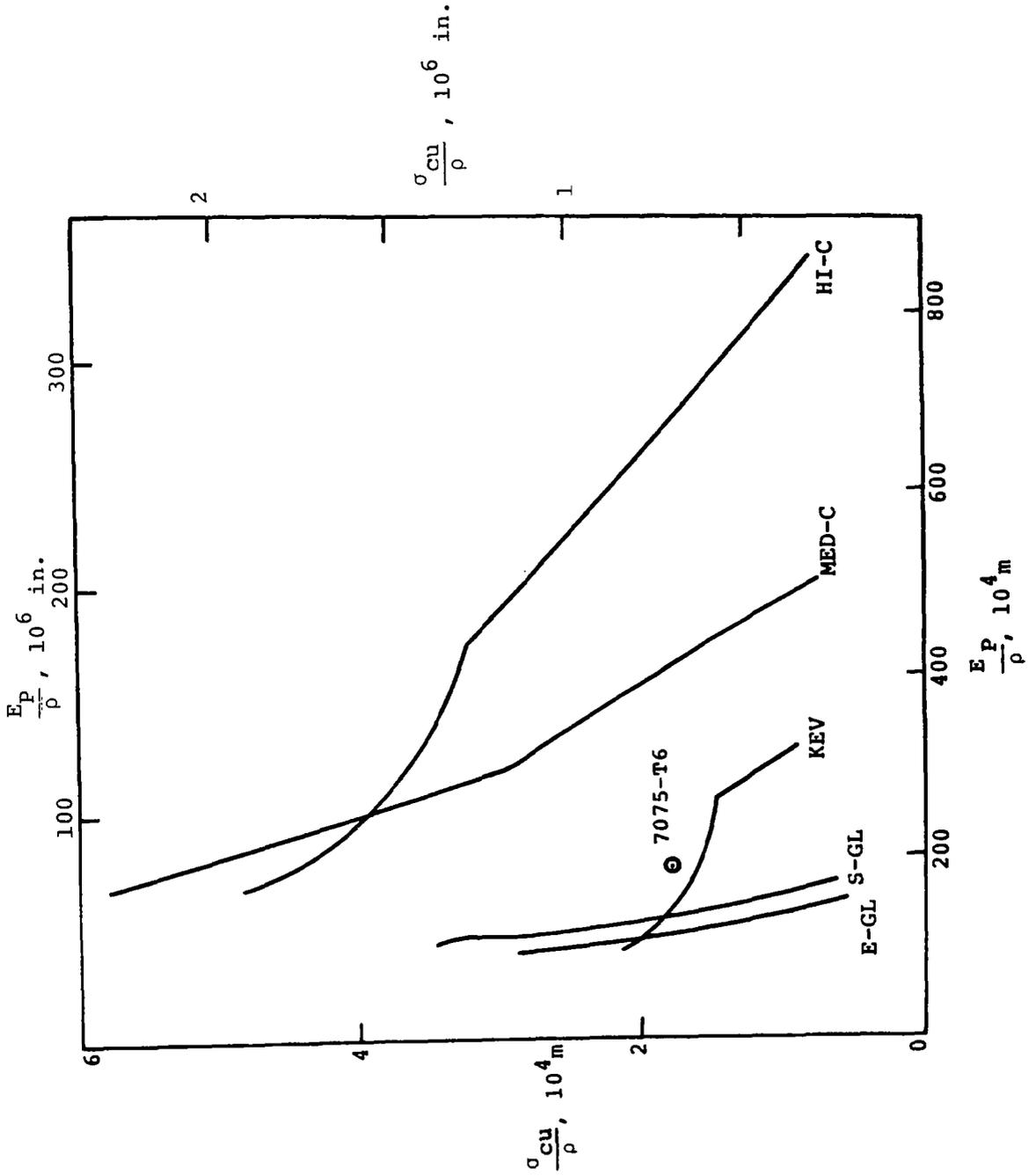
Figure B-2. Compressive Strength/Density Ratios and Plate Buckling Modulus/Density Ratios for Candidate Materials Compared to 7075-T6 Aluminum Alloy.
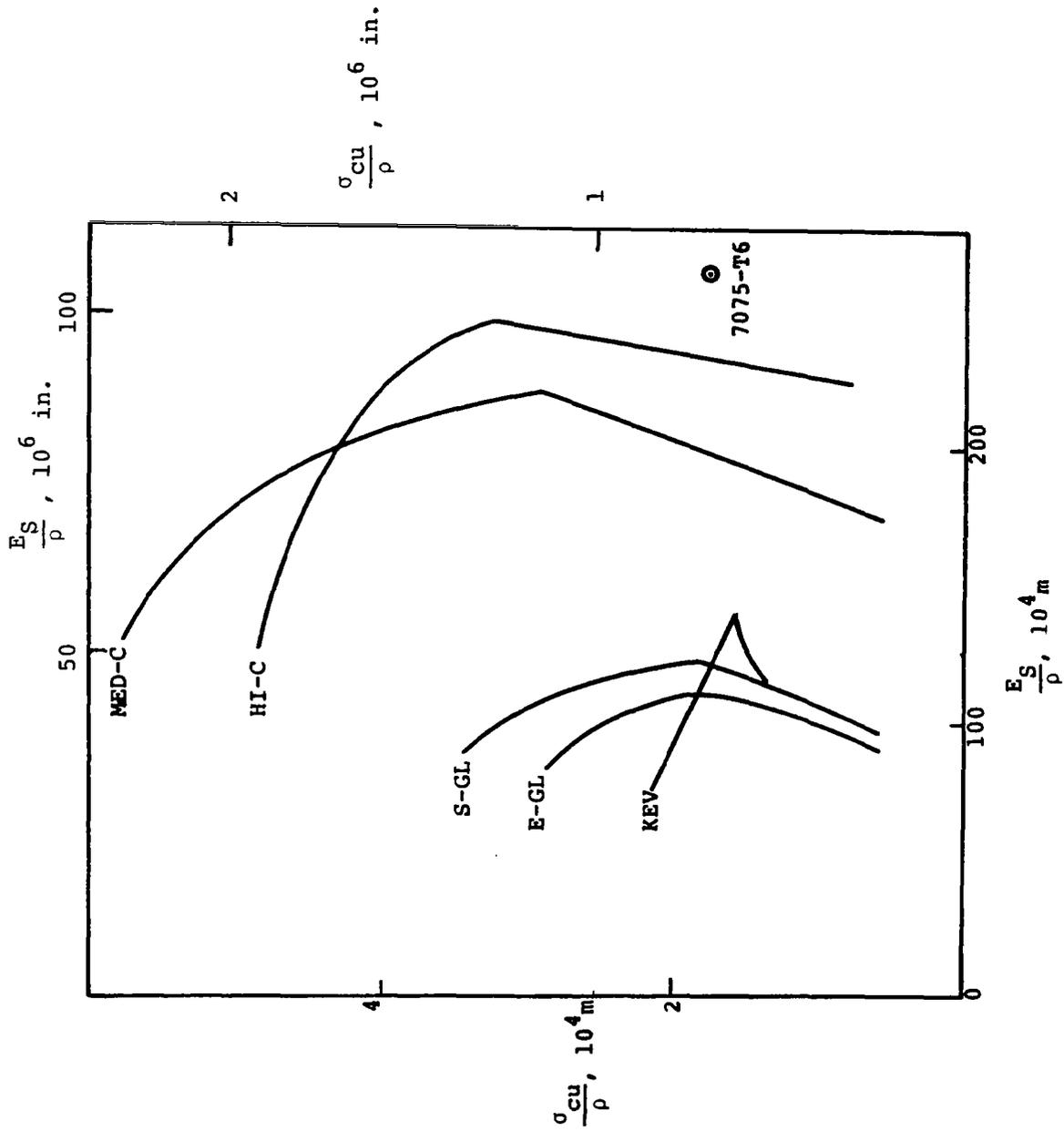
Figure B-3. Compressive Strength/Density Ratios and Shell Buckling Modulus/Density Ratios for Candidate Materials Compared to 7075-T6 Aluminum Alloy.
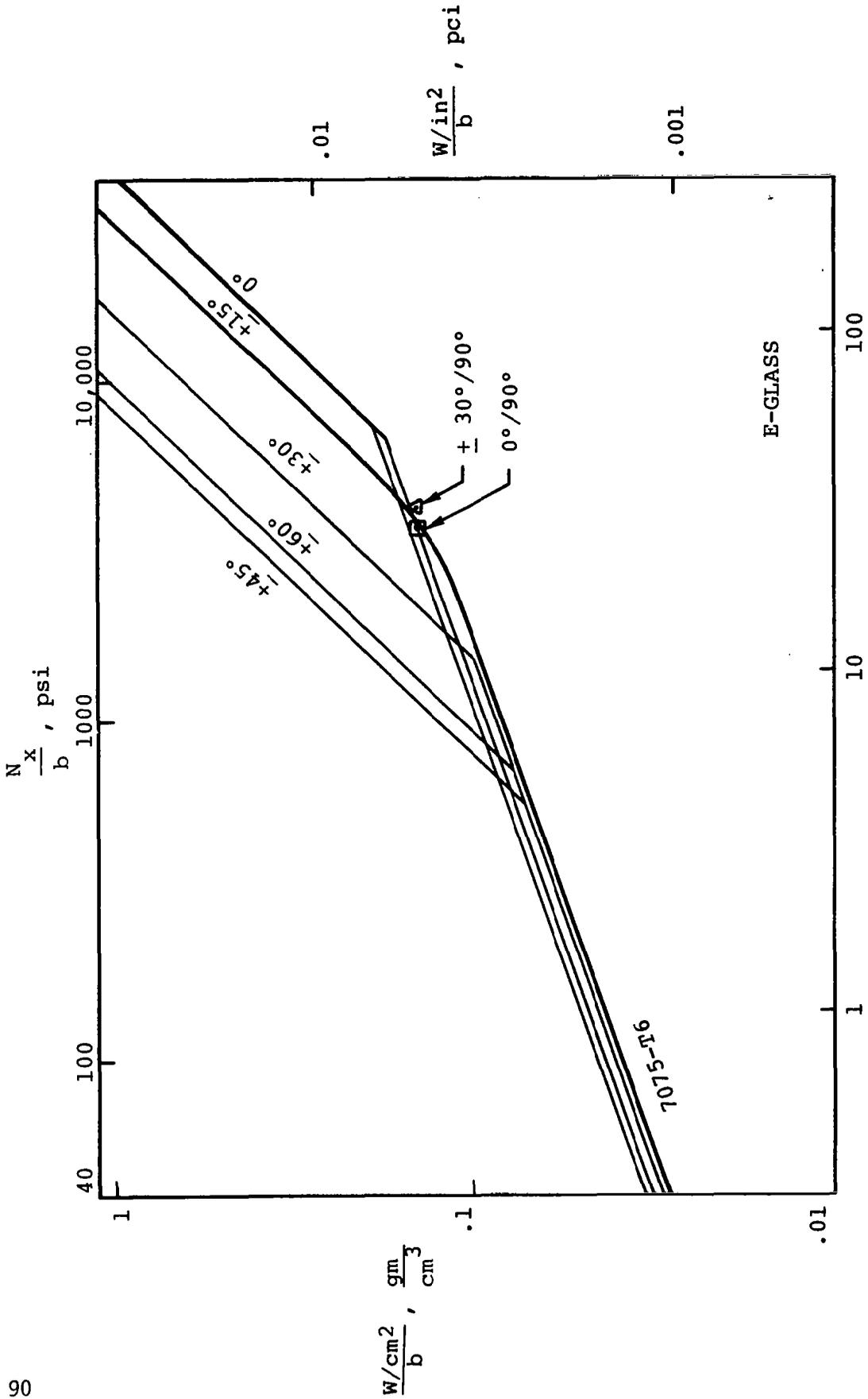
Figure B-4. Efficiencies of Various Reinforcement Configurations for Flat Plates in Compression; E-Glass/Epoxy Compared to 7075-T6 Aluminum Alloy.
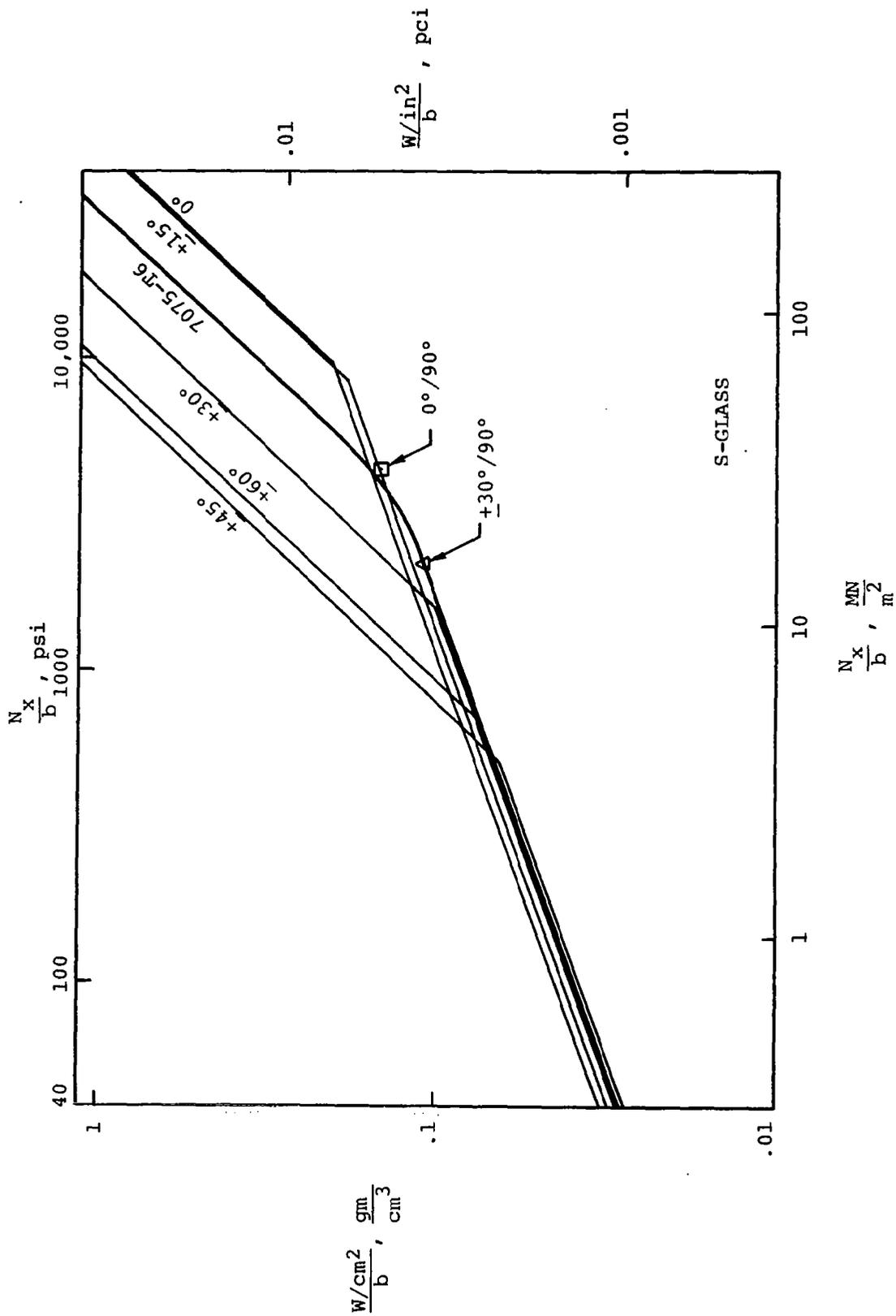
Figure B-5. Efficiencies of Various Reinforcement Configurations for Flat Plates in Compression; S-Glass/Epoxy Compared to 7075-T6 Aluminum Alloy.
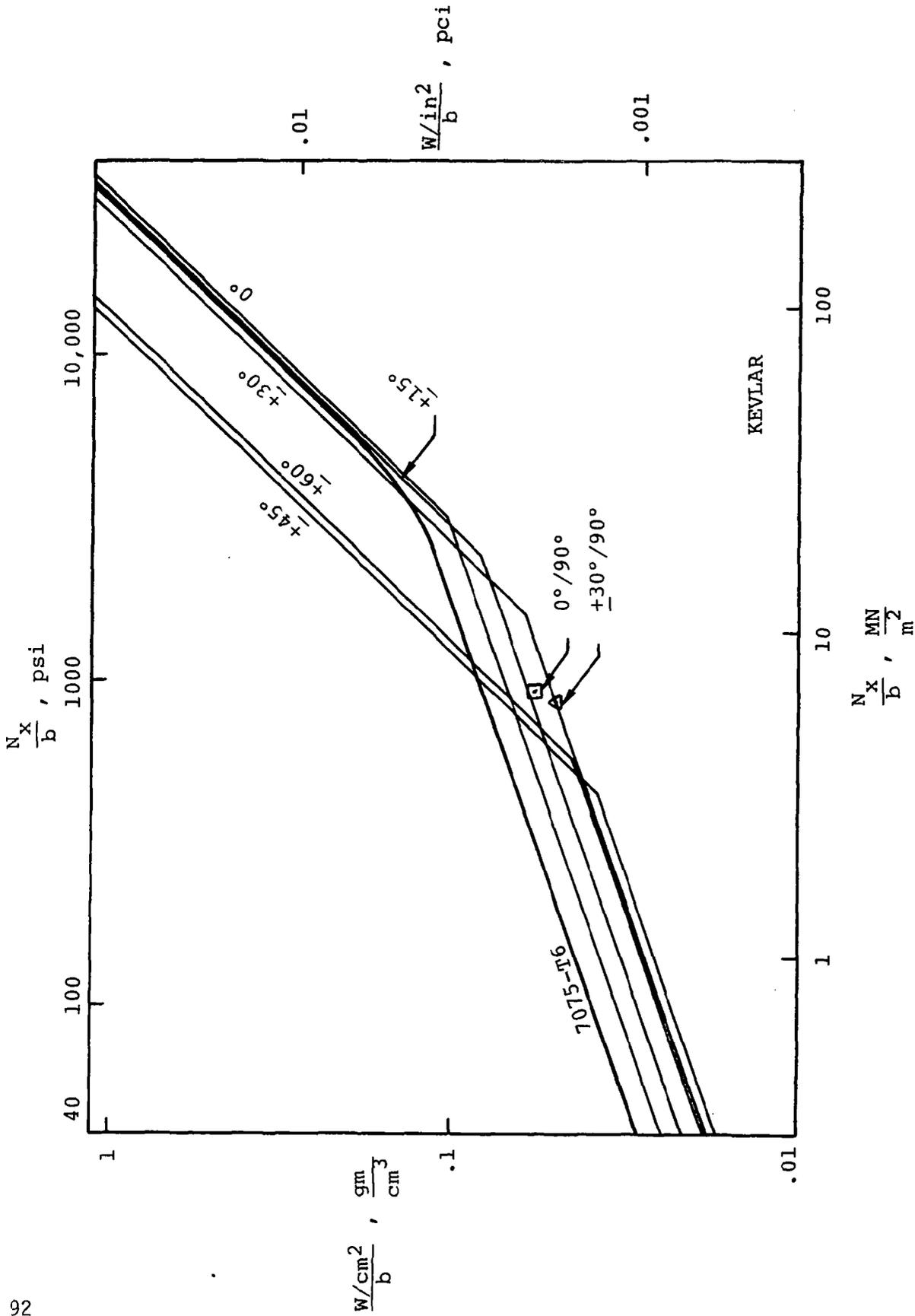
Figure B-6. Efficiencies of Various Reinforcement Configurations for Flat Plates in Compression; Kevlar/Epoxy Compared to 7075-T6 Aluminum Alloy.
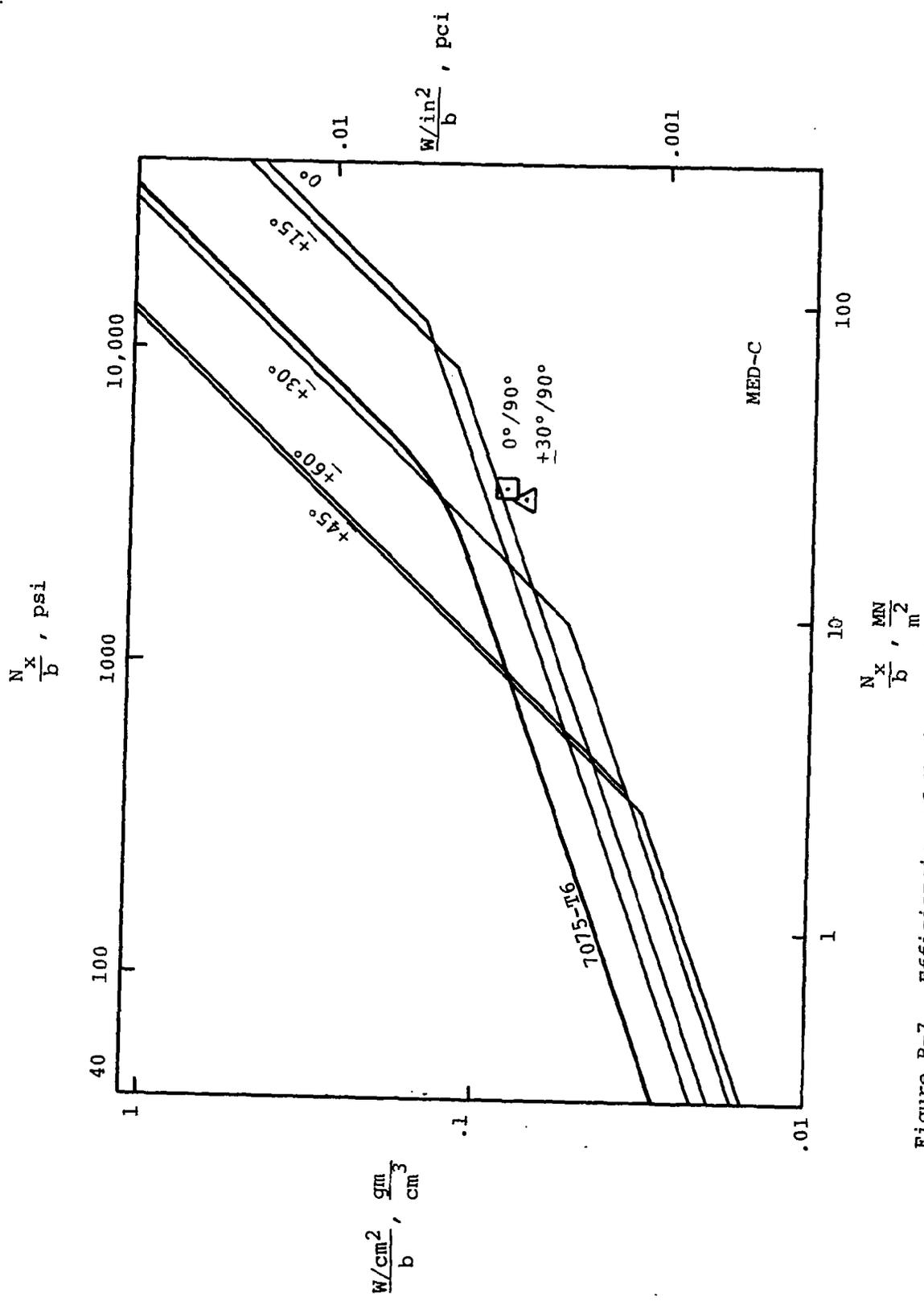
Figure B-7. Efficiencies of Various Reinforcement Configurations for Flat Plates in Compression; Medium-High Modulus Carbon Fibers in Epoxy Compared to 7075-T6 Aluminum Alloy.

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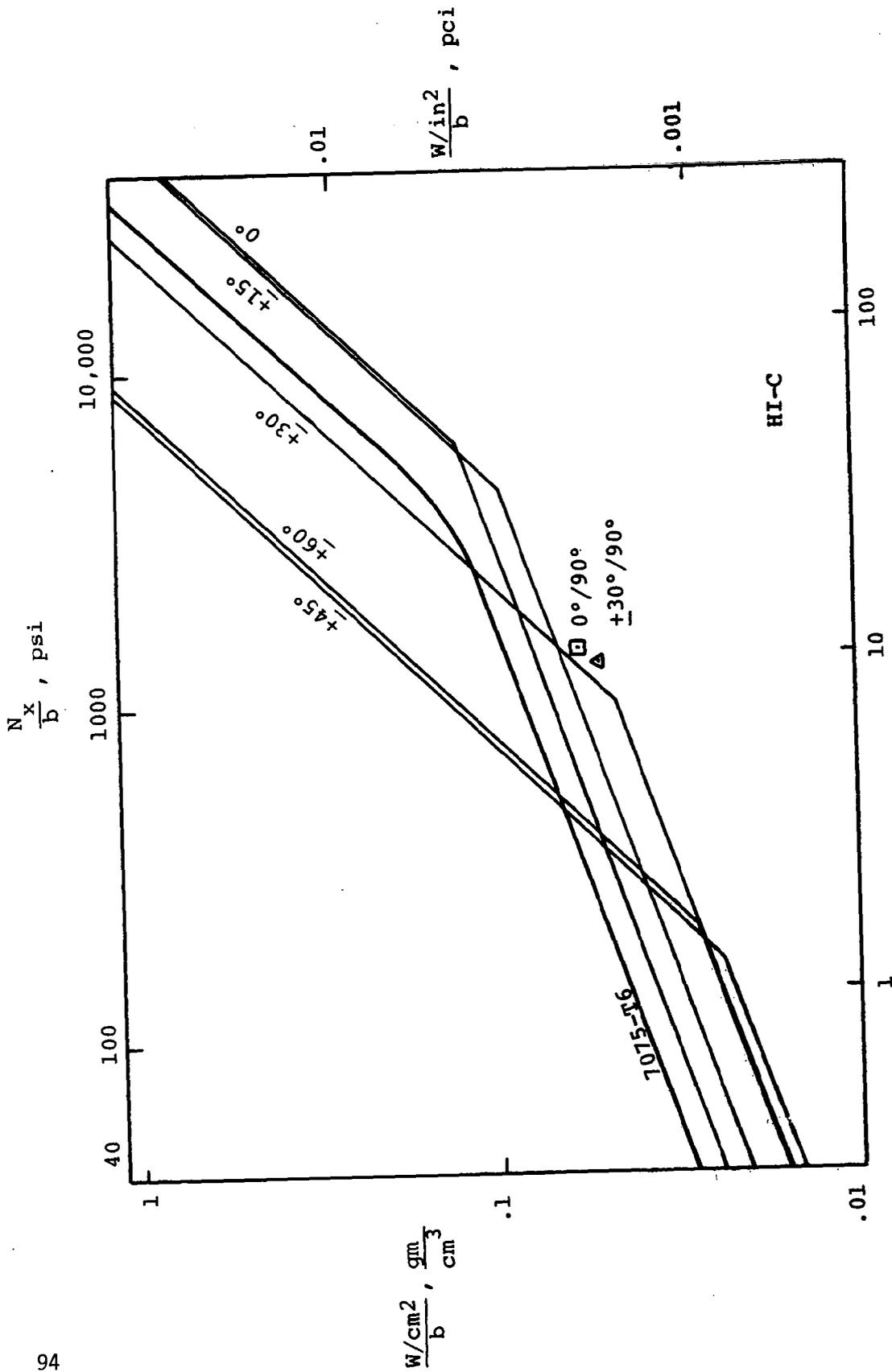
Figure B-8. Efficiencies of Various Reinforcement Configurations for Flat
Plates in Compression; High Modulus Carbon Fibers in Epoxy
Compared to 7075-T6 Aluminum Alloy.
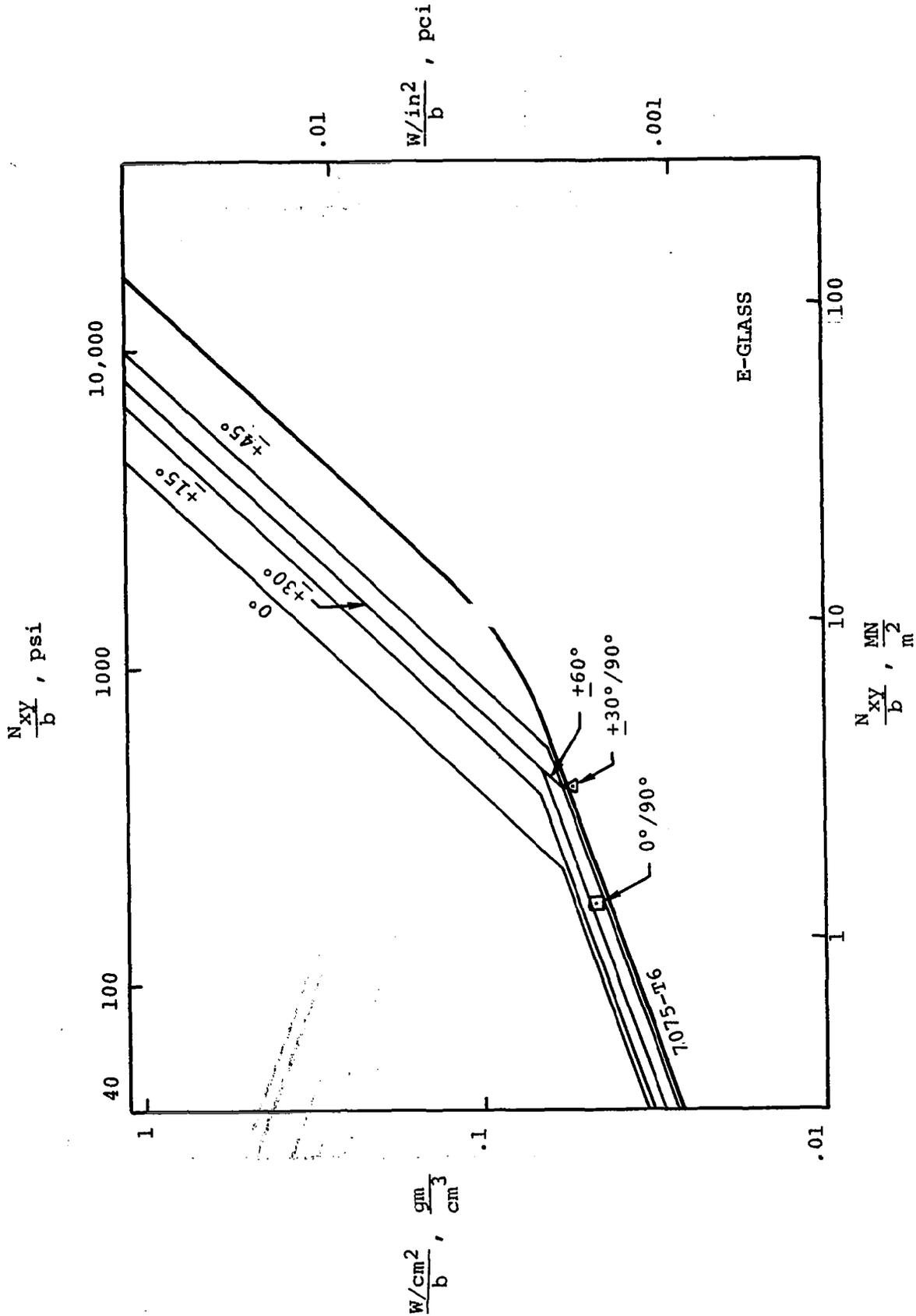
Figure B-9. Efficiencies of Various Reinforcement Configurations for Flat Plates in Shear; E-Glass/Epoxy Compared to 7075-T6 Aluminum Alloy.

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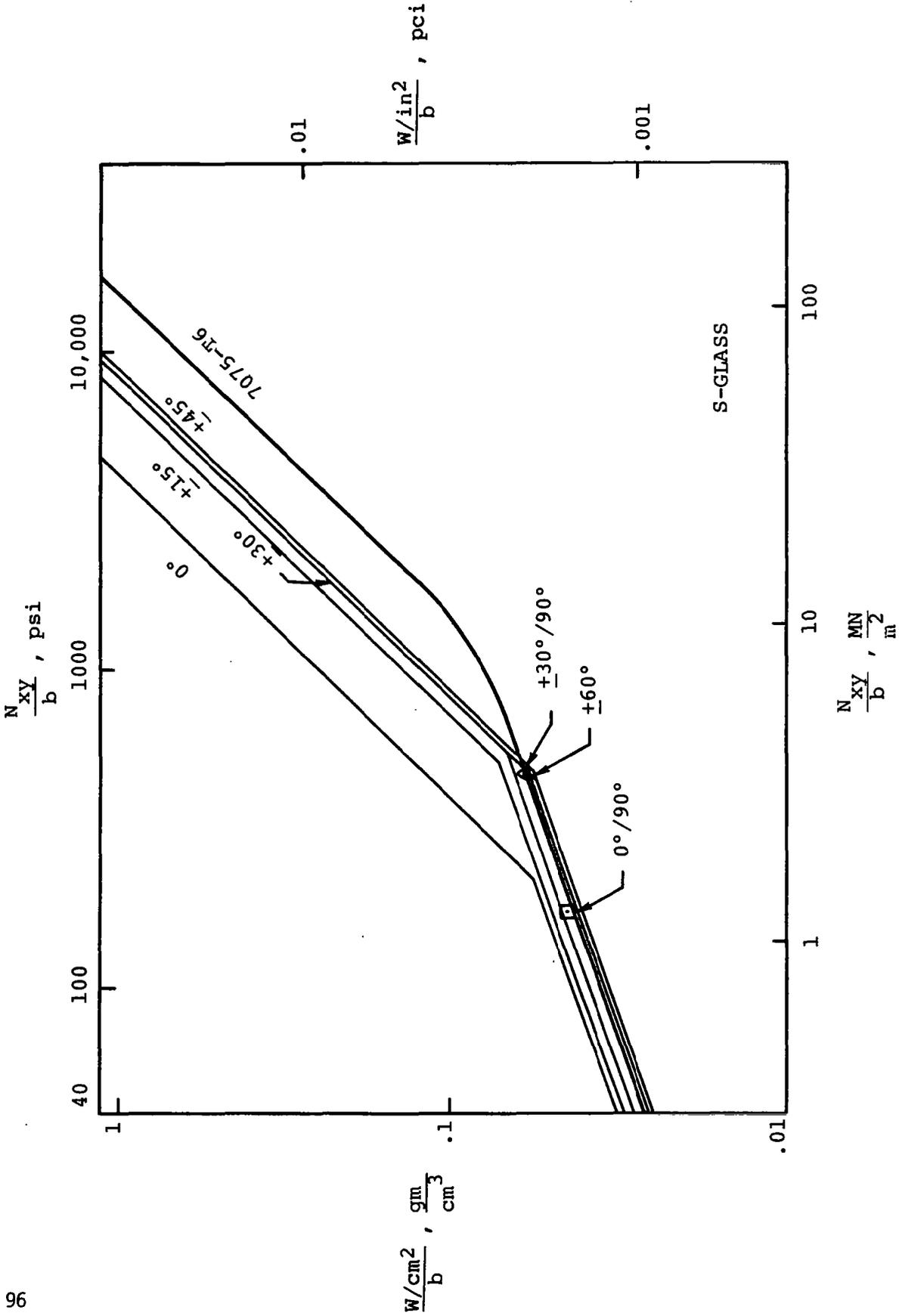
Figure B-10. Efficiencies of Various Reinforcement Configurations for Flat
Plates in Shear; S-Glass/Epoxy Compared to 7075-T6 Aluminum Alloy.
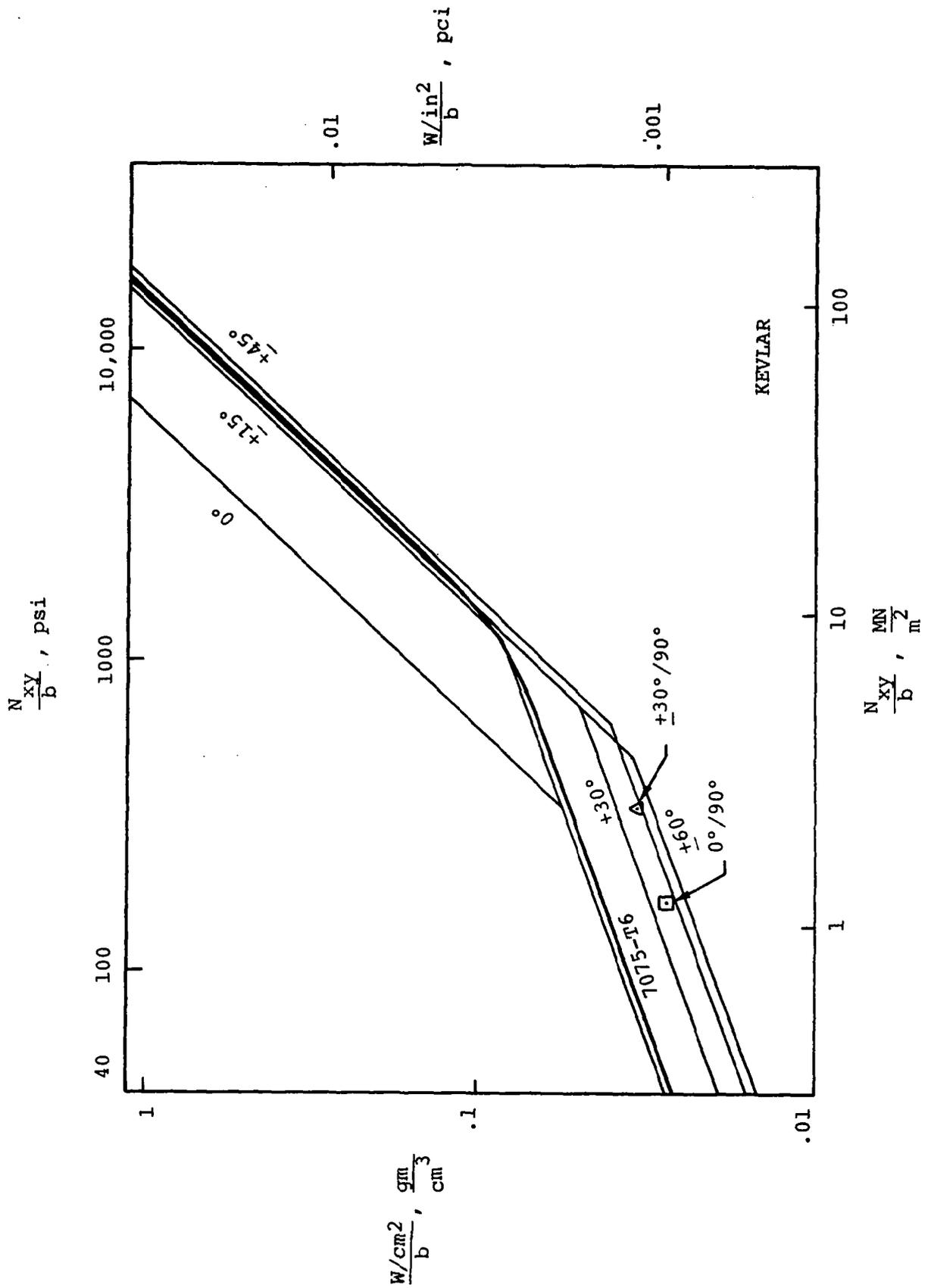
Figure B-11. Efficiencies of Various Reinforcement Configurations for Flat Plates in Shear; Kevlar/Epoxy Compared to 7075-T6 Aluminum Alloy.
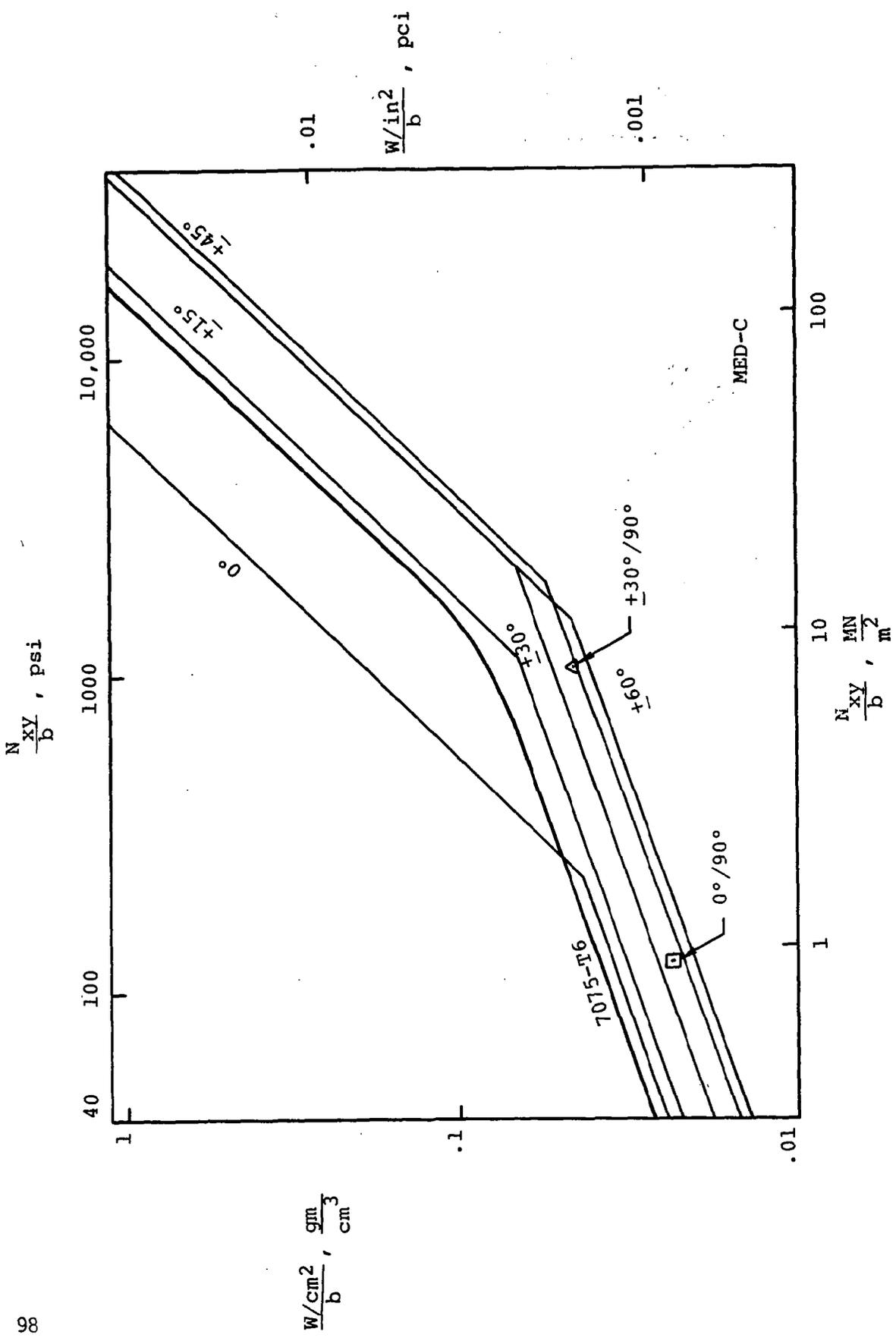
Figure B-12. Efficiencies of Various Reinforcement Configurations for Flat Plates in Shear; Medium-High Modulus Carbon Fibers in Epoxy Compared to 7075-T6 Aluminum Alloy.
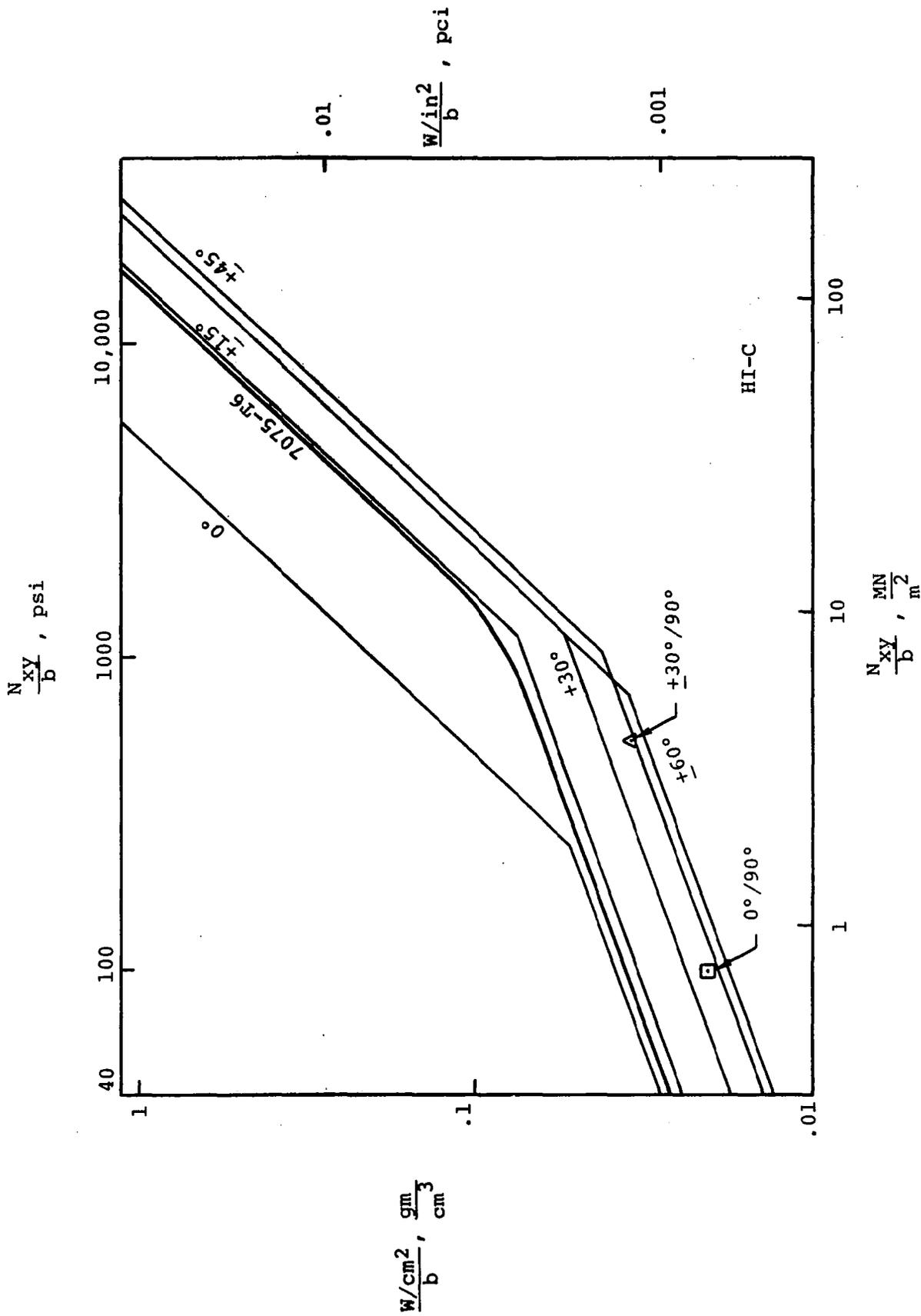
Figure B-13. Efficiencies of Various Reinforcement Configurations for Flat Plates in Shear; High Modulus Carbon Fibers in Epoxy Compared to 7075-T6 Aluminum Alloy.
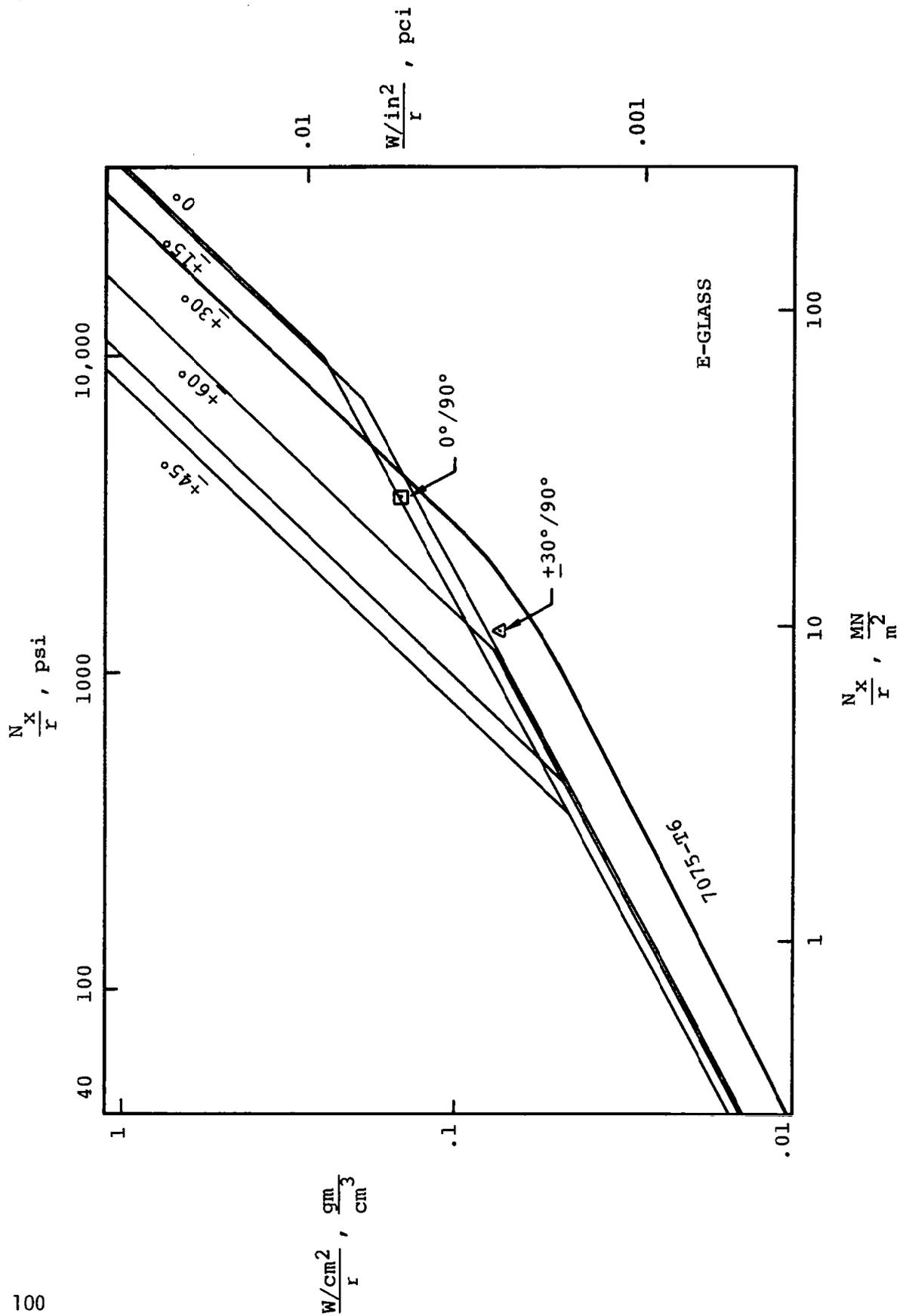
Figure B-14. Efficiencies of Various Reinforcement Configurations for Shells in Compression; E-Glass/Epoxy Compared to 7075-T6 Aluminum Alloy.
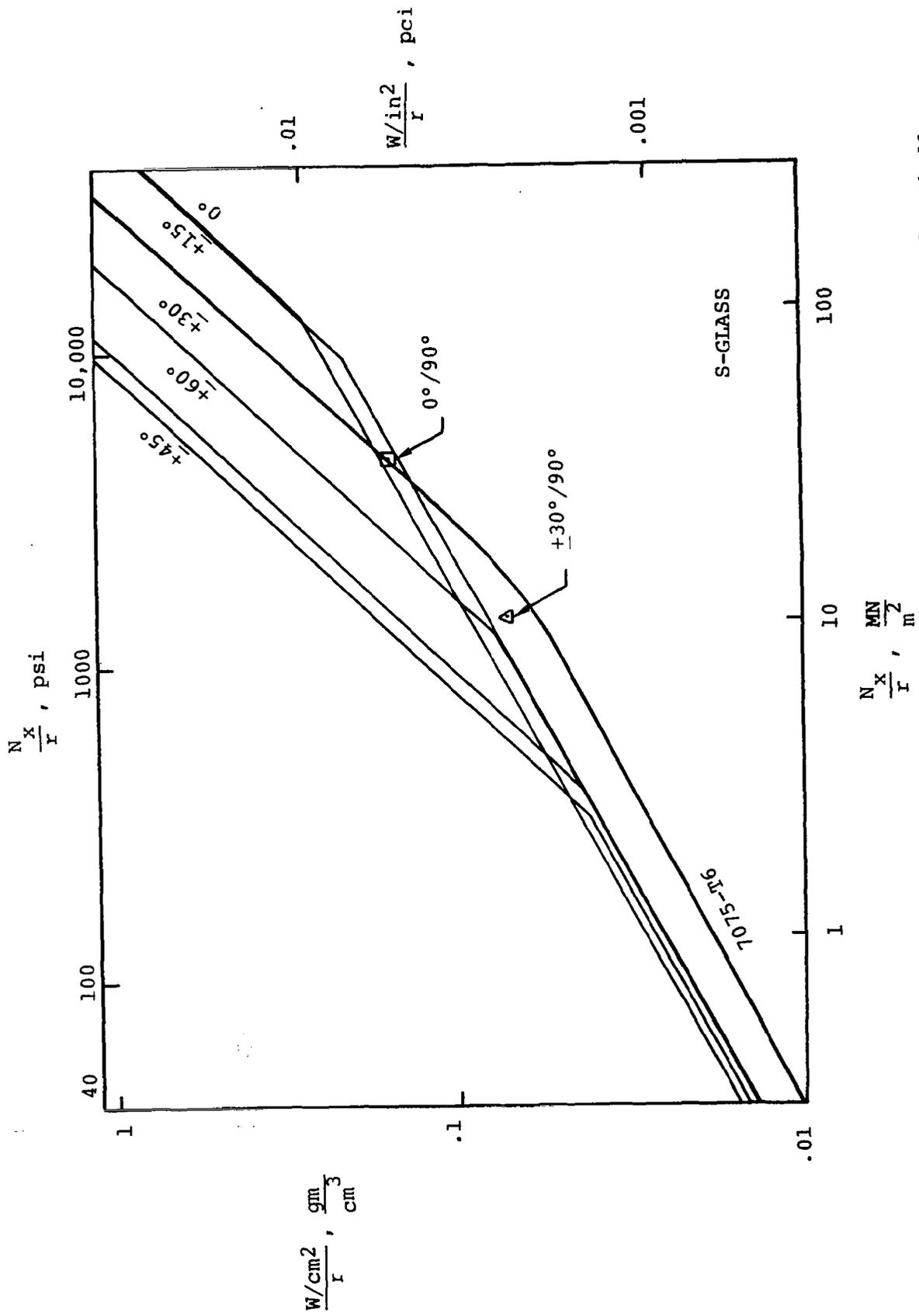
Figure B-15. Efficiencies of Various Reinforcement Configurations for Shells
in Compression; S-Glass/Epoxy Compared to 7075-T6 Aluminum Alloy.
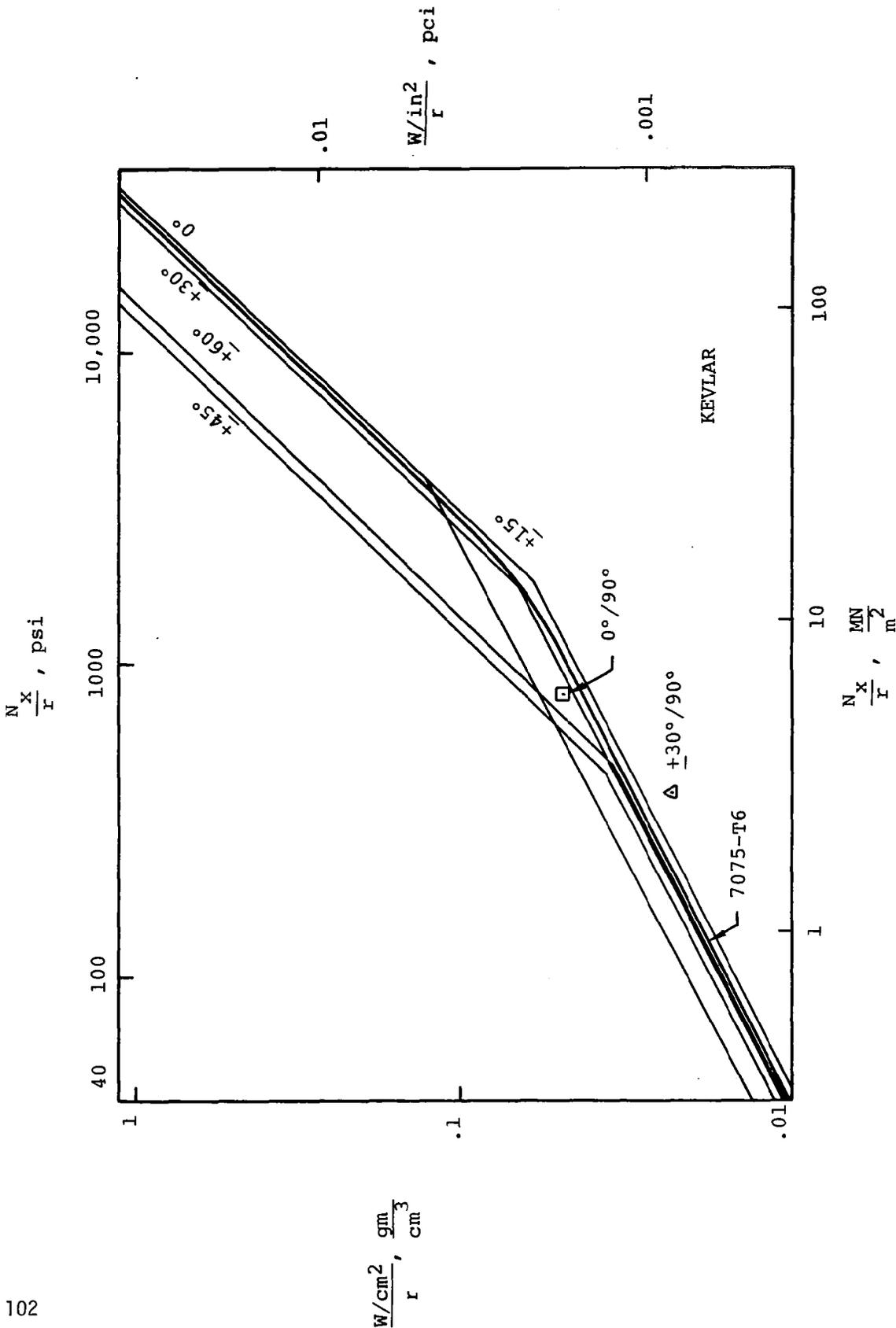
Figure B-16. Efficiencies of Various Reinforcement Configurations for Shells
in Compression; Kevlar/Epoxy Compared to 7075-T6 Aluminum Alloy.
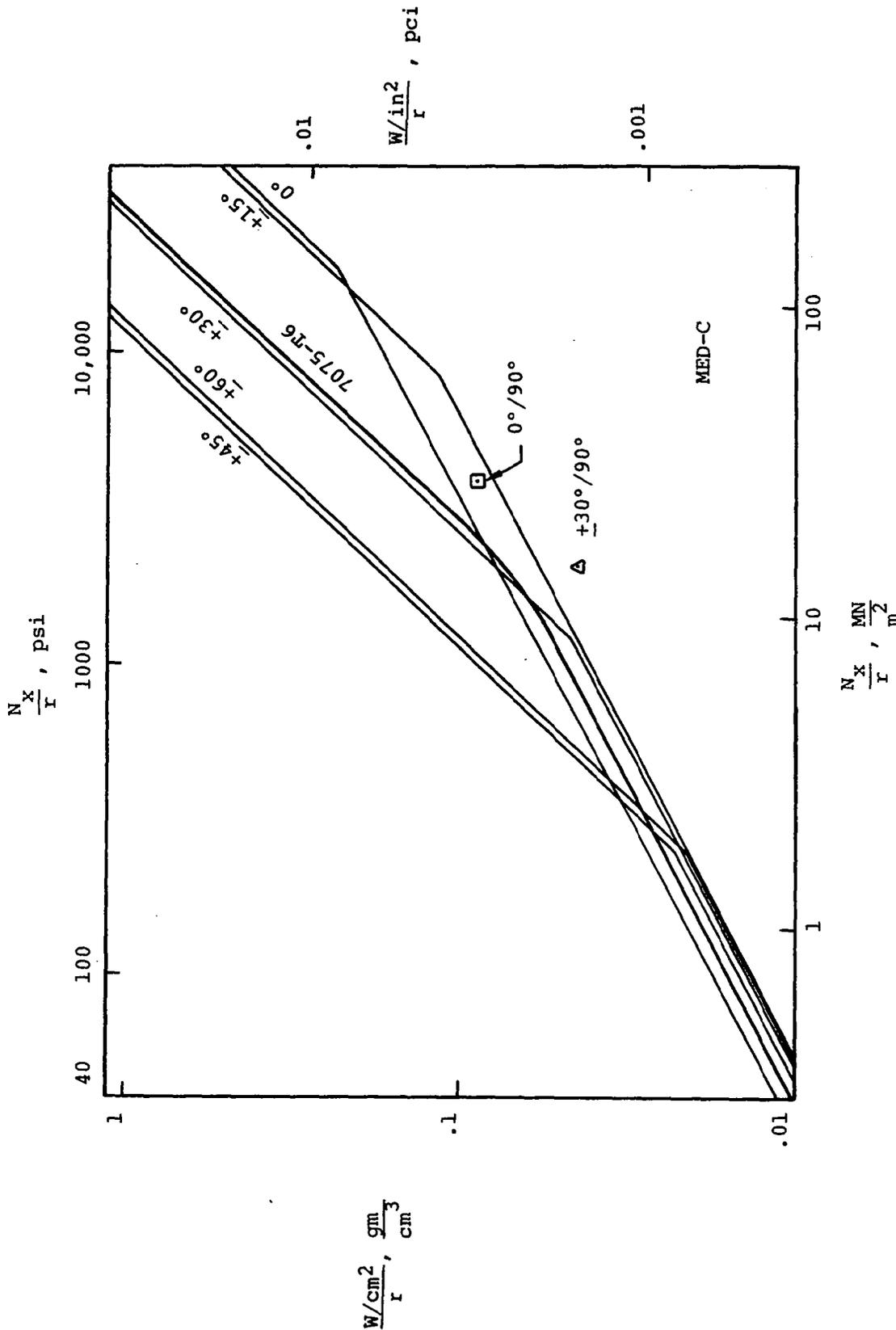
Figure B-17. Efficiencies of Various Reinforcement Configurations for Shells in Compression; Medium-High Modulus Carbon Fibers in Epoxy Compared to 7075-T6 Aluminum Alloy.

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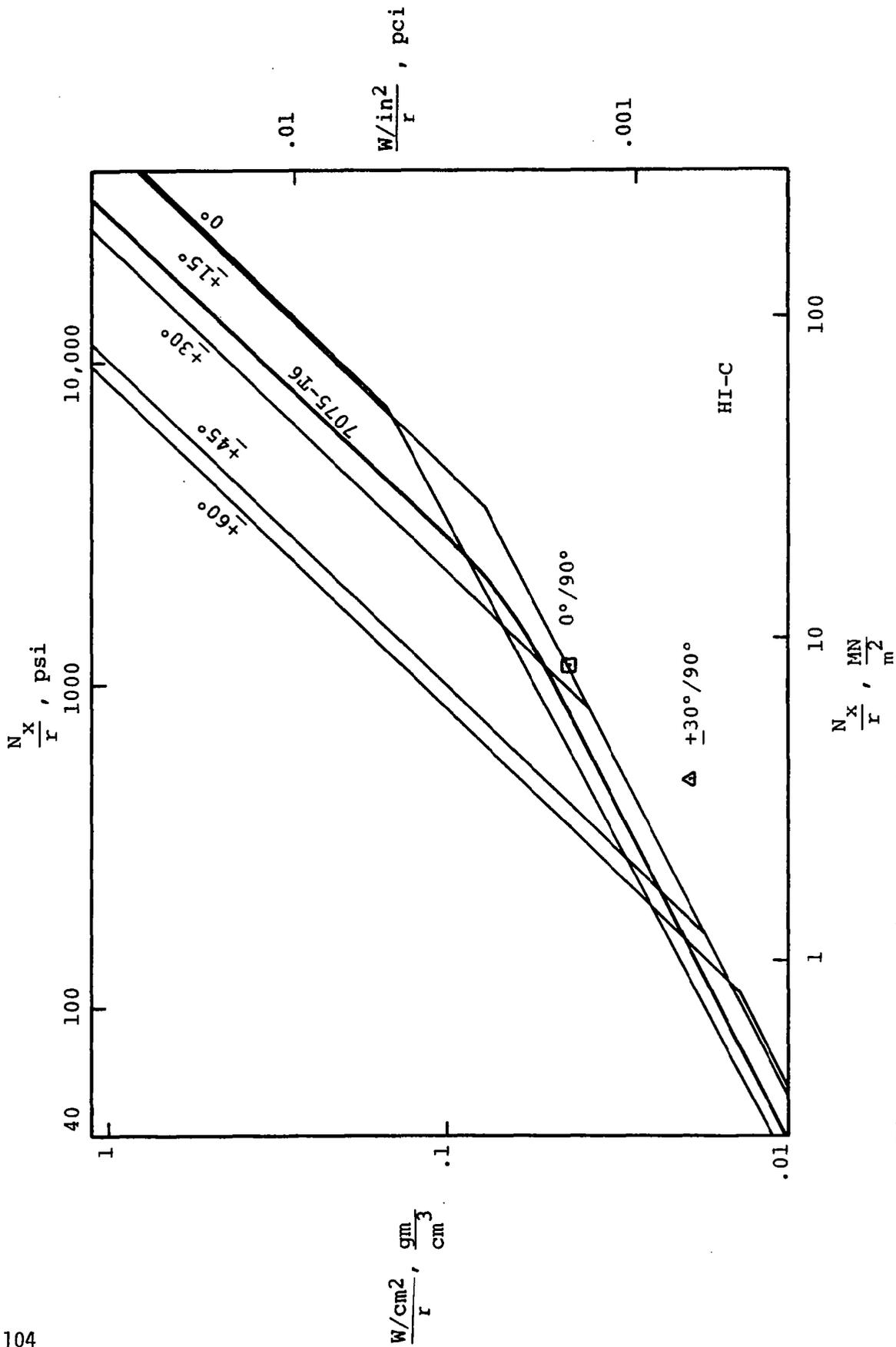
Figure B-18. Efficiencies of Various Reinforcement Configurations for Shells in Compression; High Modulus Carbon Fibers in Epoxy Compared to 7075-T6 Aluminum Alloy.
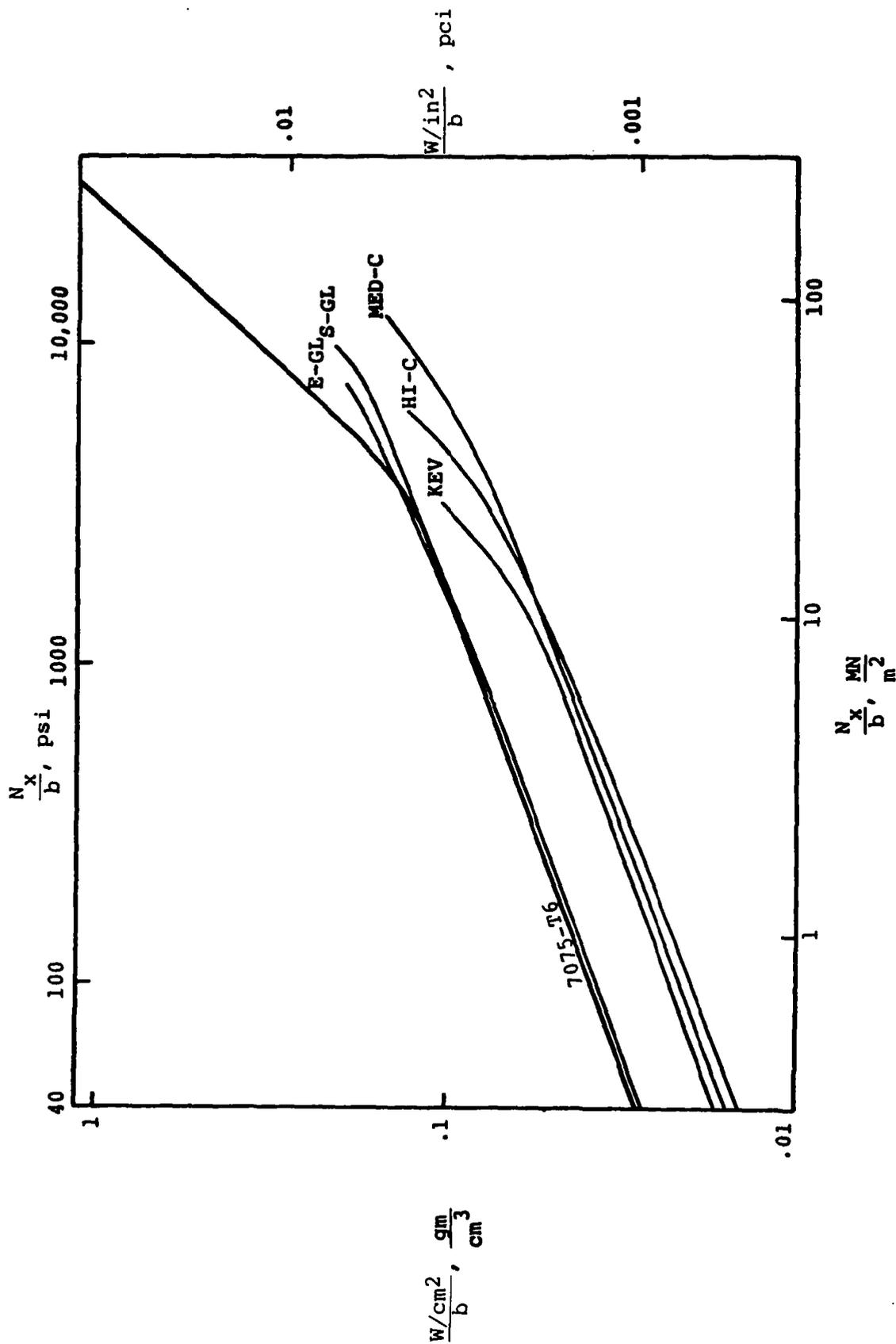
Figure B-19. Efficiencies of Candidate Materials for Flat Plates in Com-
pression Compared to 7075-T6 Aluminum Alloy.

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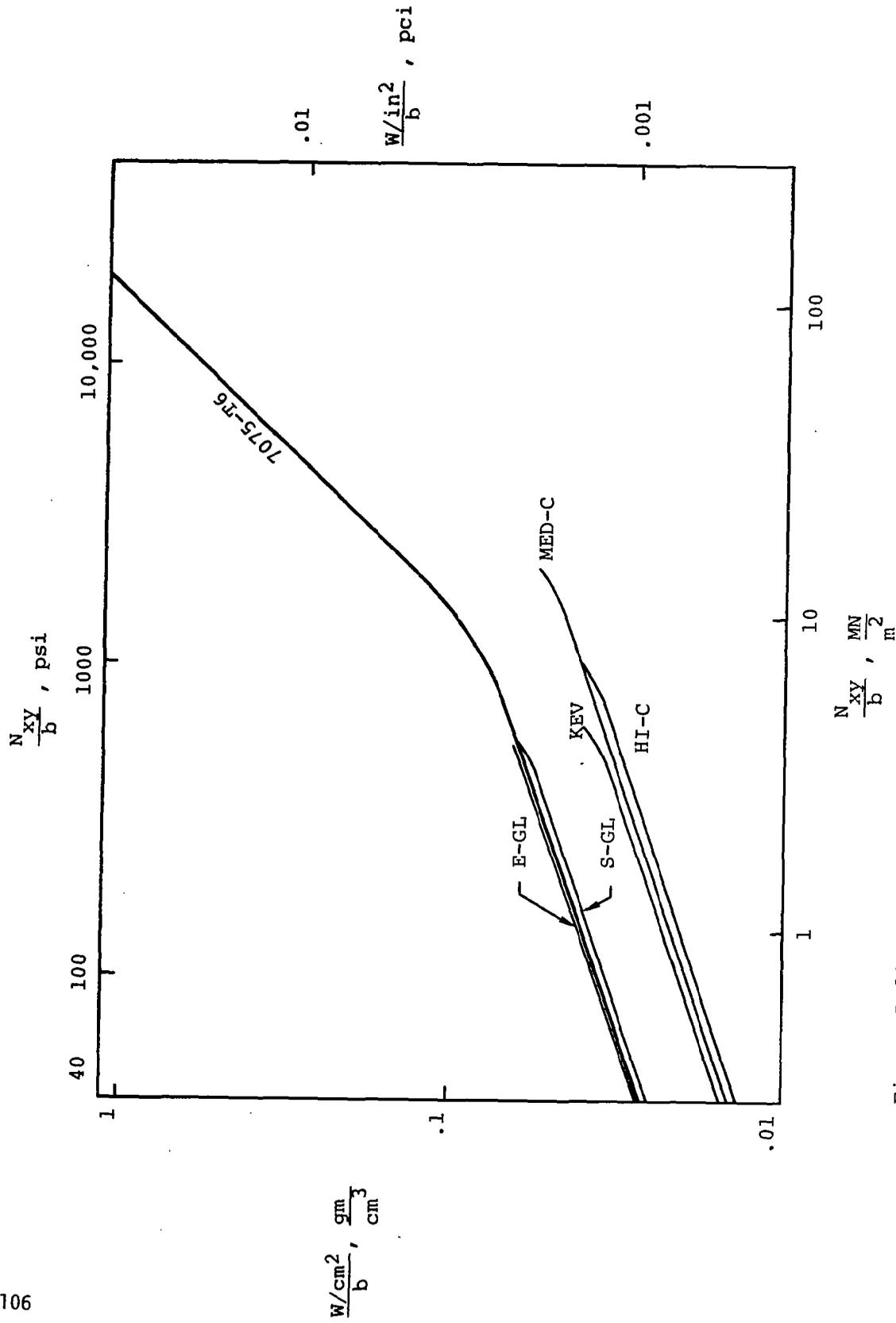
Figure B-20. Efficiencies of Candidate Materials for Flat Plates in Shear
Compared to 7075-T6 Aluminum Alloy.
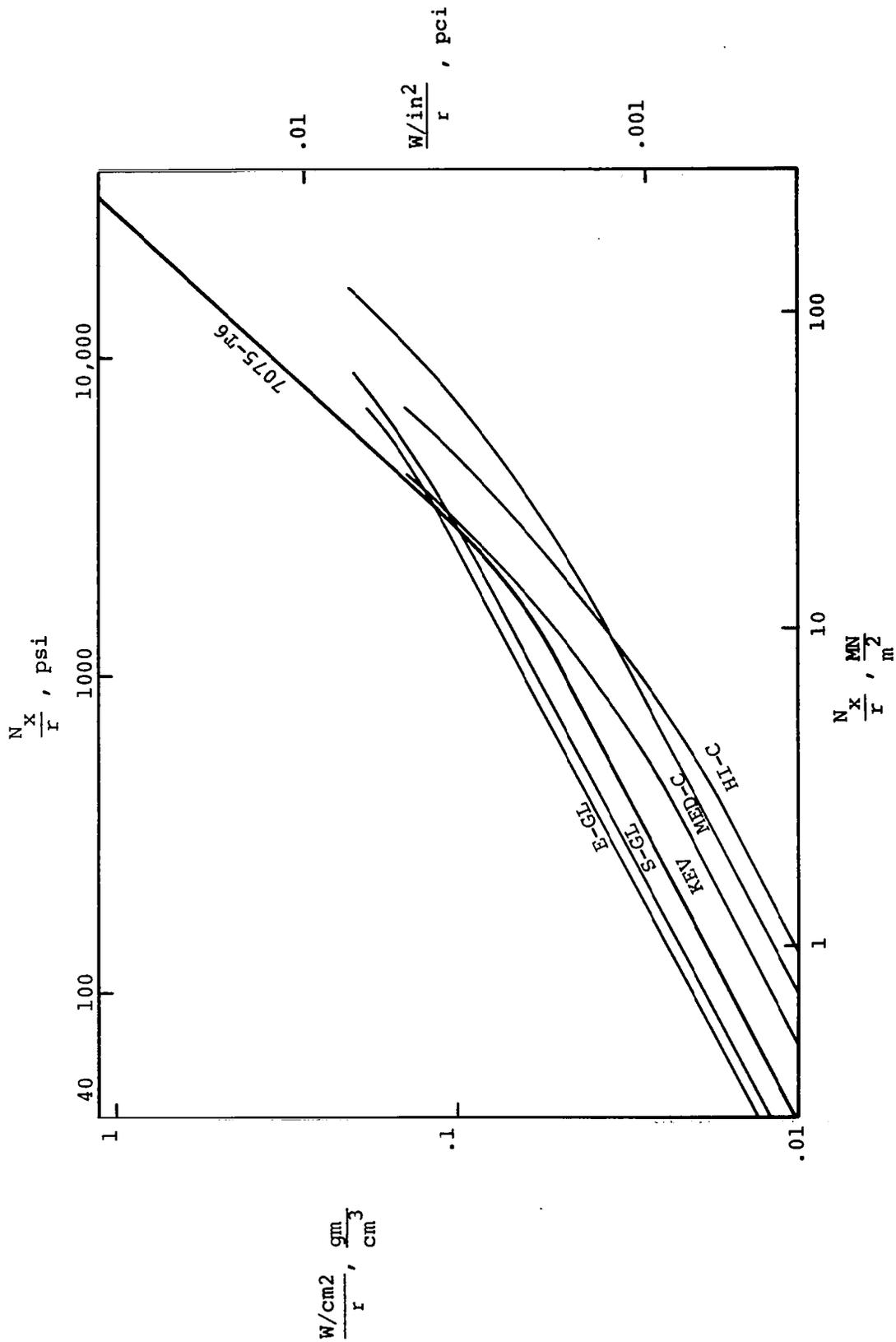
Figure B-21. Efficiencies of Candidate Materials for Shells in Compression Compared to 7075-T6 Aluminum Alloy. - Envelopes of All Reinforcement Configurations Except Quasi-Isotropic.
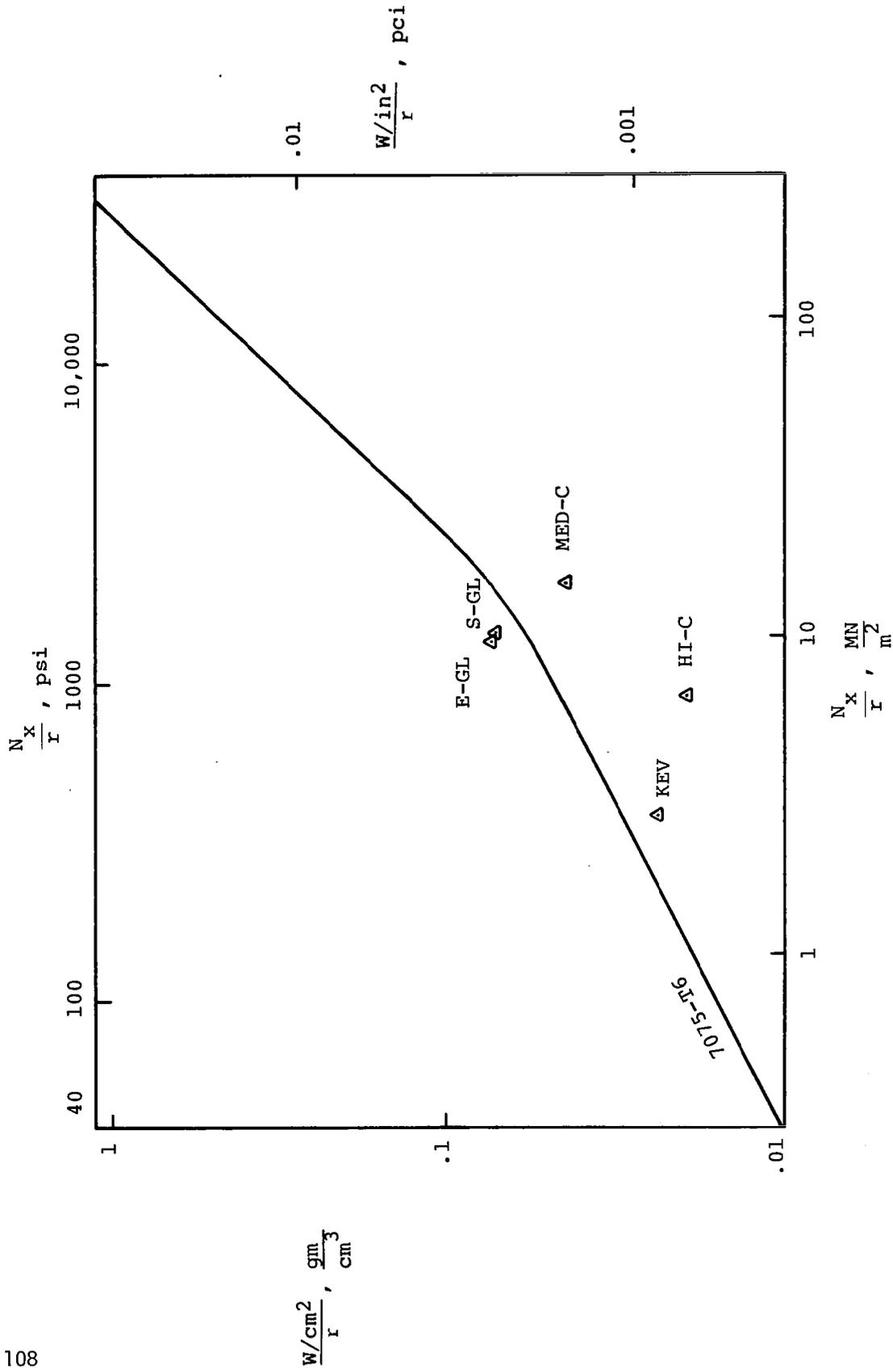
Figure B-22. Efficiencies of Candidate Materials as Shells in Compression Compared to 7075-T6 Aluminum Alloy ~ Quasi-Isotropic Rein-forcements.
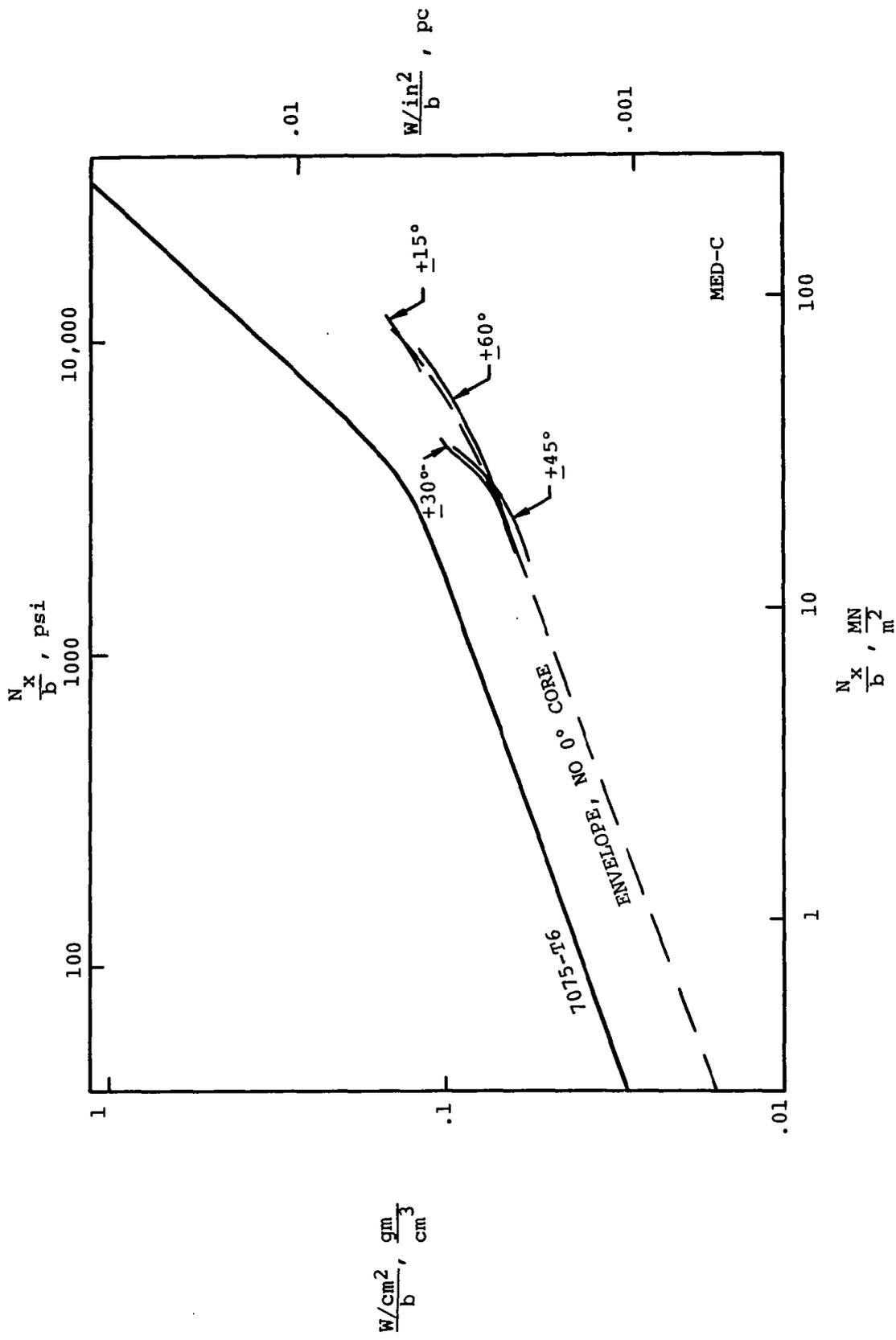
Figure B-23. Efficiencies of Hybrid Reinforcement Configurations, Having Angle-Ply Faces and a 0° Unidirectionally Reinforced Core, as Flat Plates in Compression.

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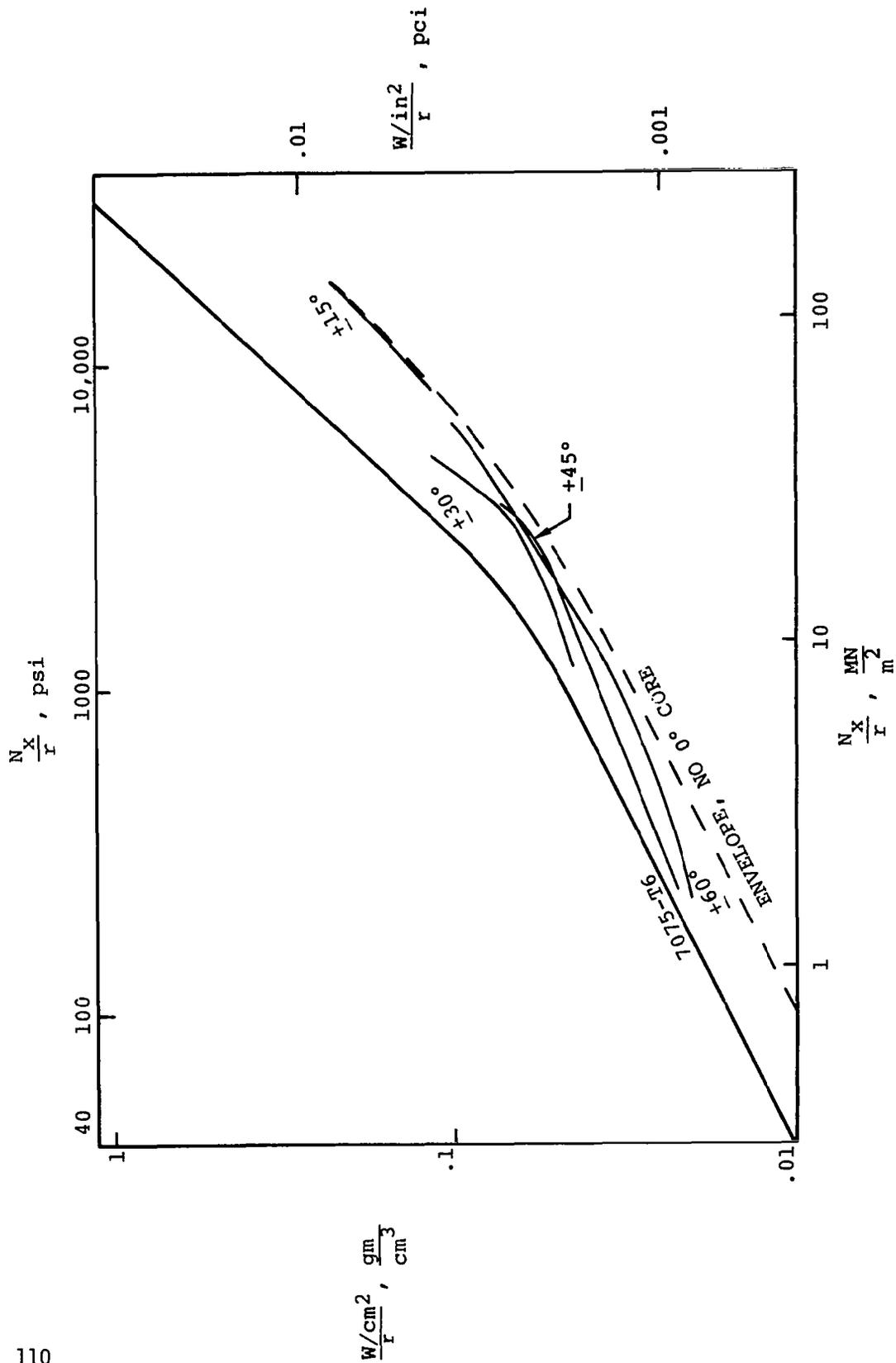
Figure B-24. Efficiencies of Hybrid Reinforcement Configurations Having
Angle-Ply Faces and a 0° Unidirectionally Reinforced Core,
as Shells in Compression.
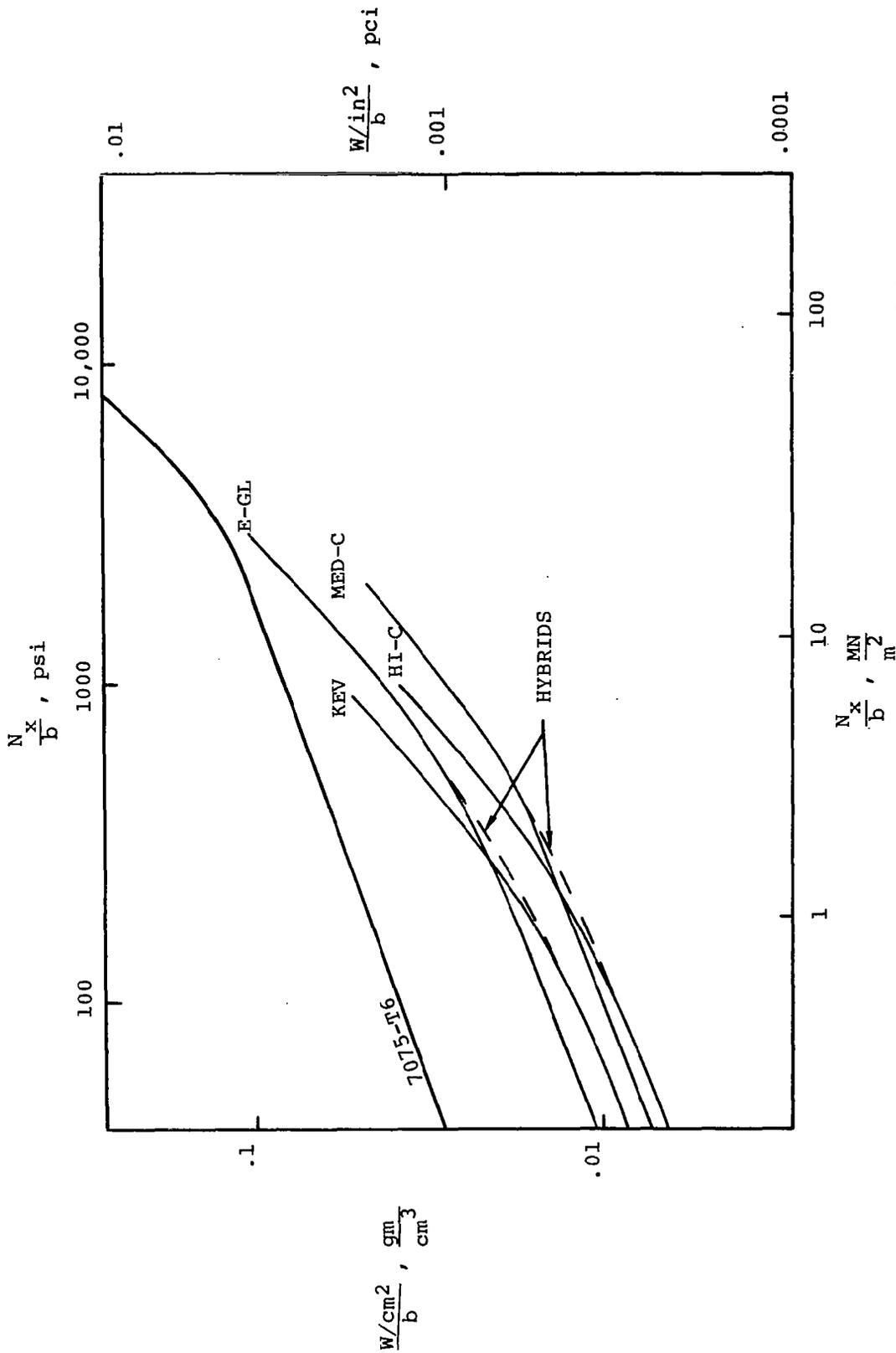
Figure B-25. Efficiencies of Hybrid Materials; Use of Higher Modulus Faces on Higher Strength 0° Cores; Kevlar on E-Glass; High Modulus Carbon on Medium-High Modulus Carbon.

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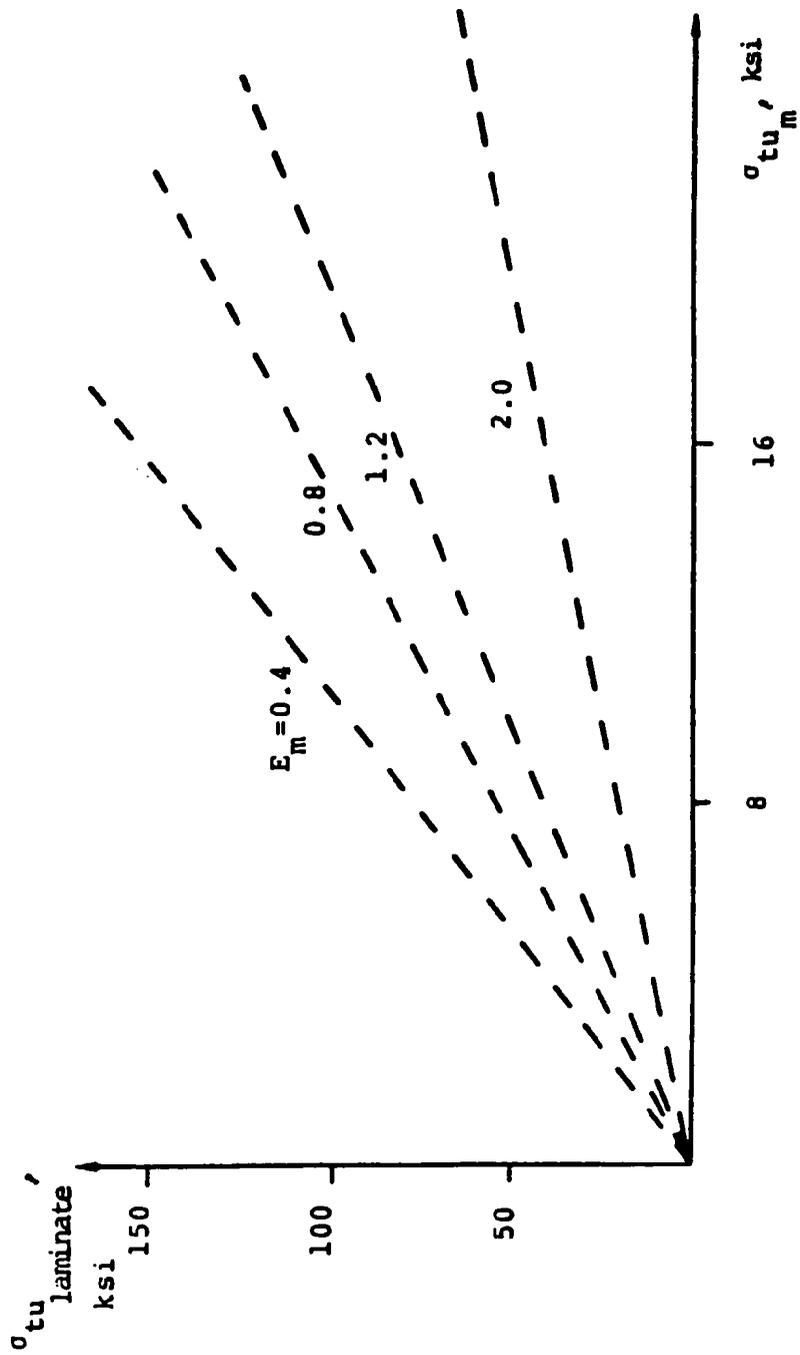
Figure B-26. Relationship of Laminate Ultimate Tensile Stress to Matrix
Strength and Stiffness, Matrix Failure Mode.

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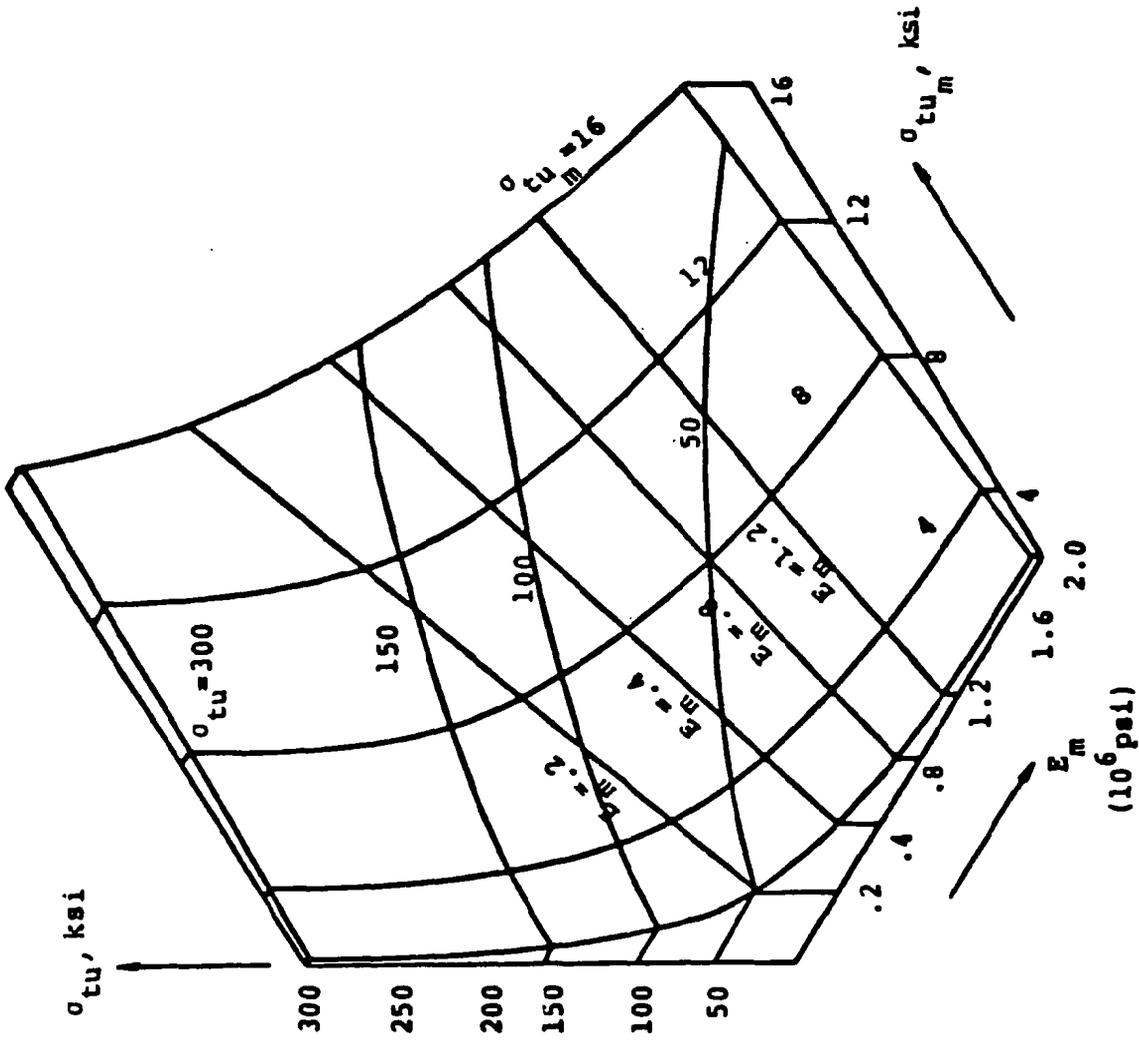
Figure B-27. Relationship of Ultimate Strength to Matrix Properties

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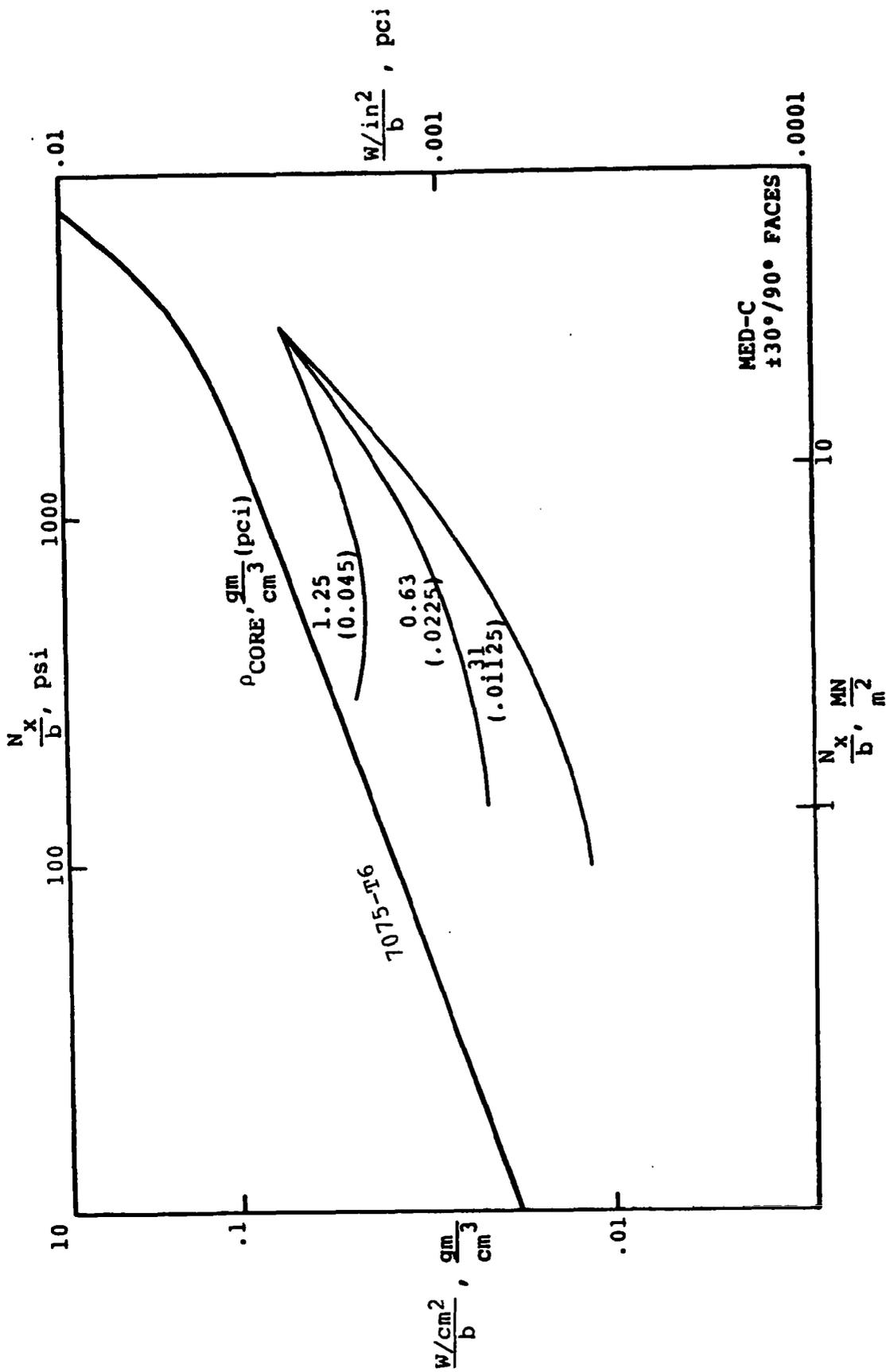
Figure B-28. Efficiencies of Composite Flat Plates Having Quasi-Isotropic Reinforcements in Their Face Plies and Cores of Varying Densities.
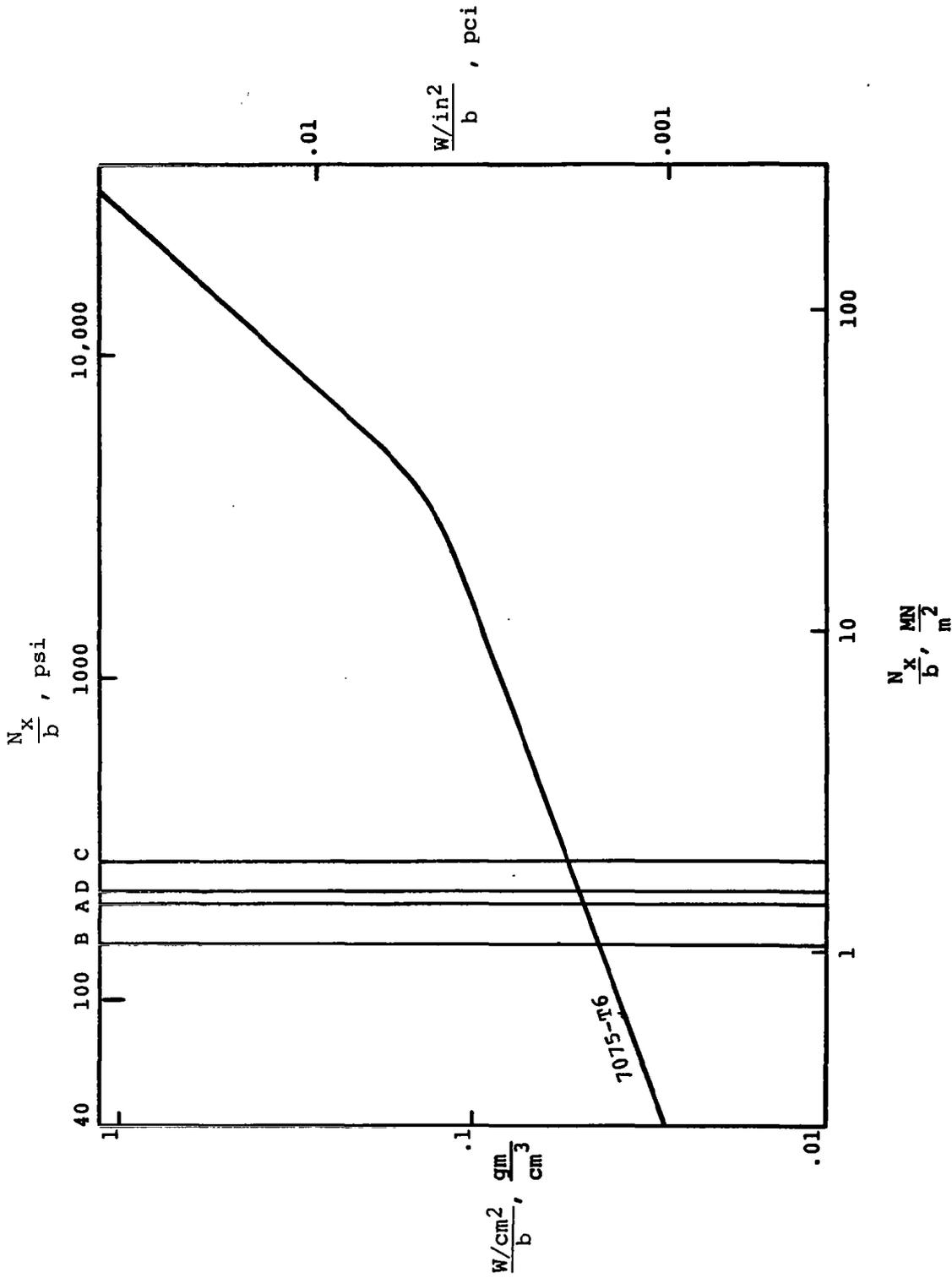
Figure B-29. Loading Intensities at the Wing Root for Four Hypothetical General Aviation Aircraft; A, a 1000 kg (2200 lb) Trainer, B a 2000 Kg (4400 lb), and C a 4000 kg (8800 lb) Single Engine Aircraft; D a 8000 kg (17,600 lb) Twin Engine Aircraft.
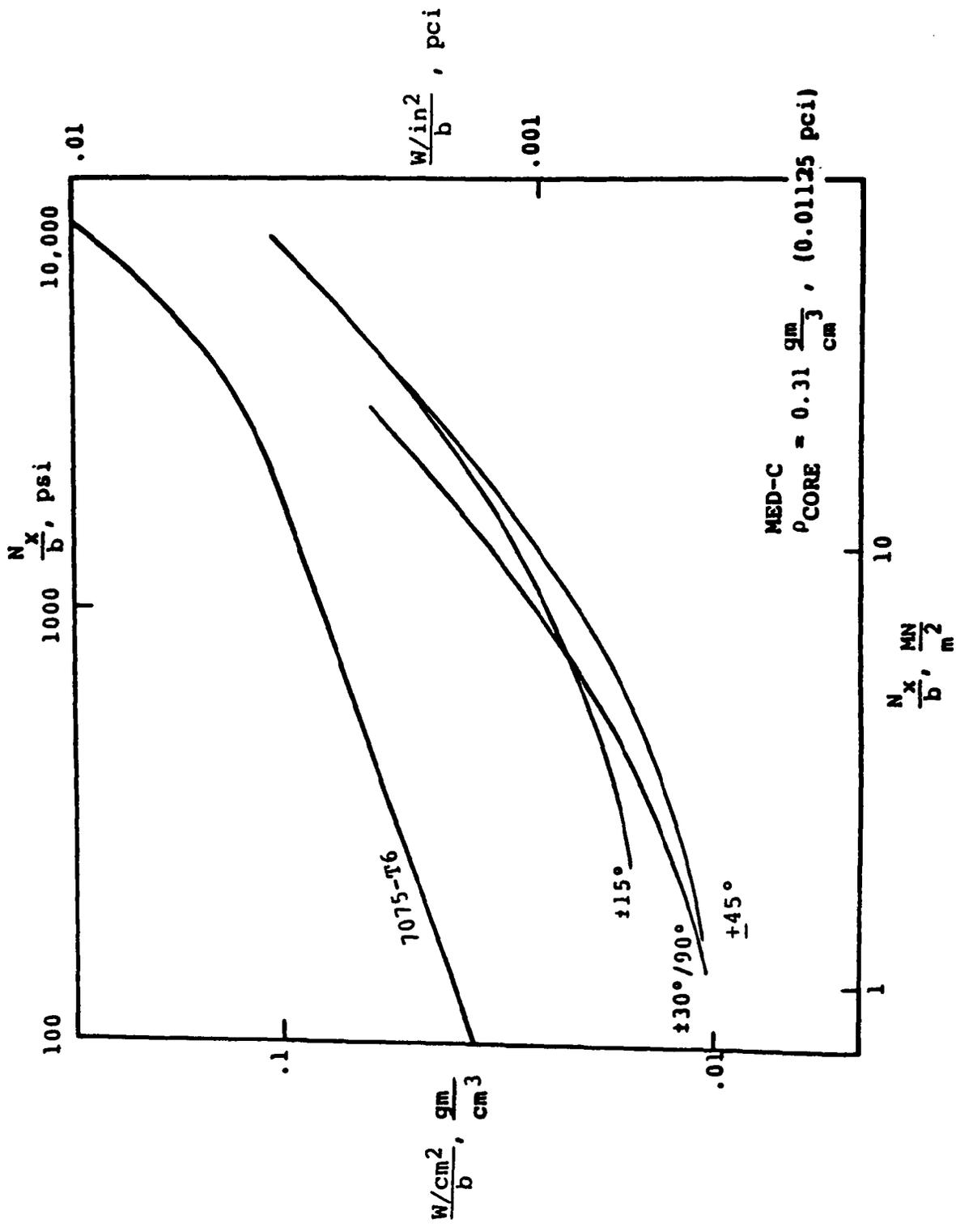
Figure B-30. Candidate Reinforcement Configurations and Materials for Research and Development for Flat Plate Compressive Applications.
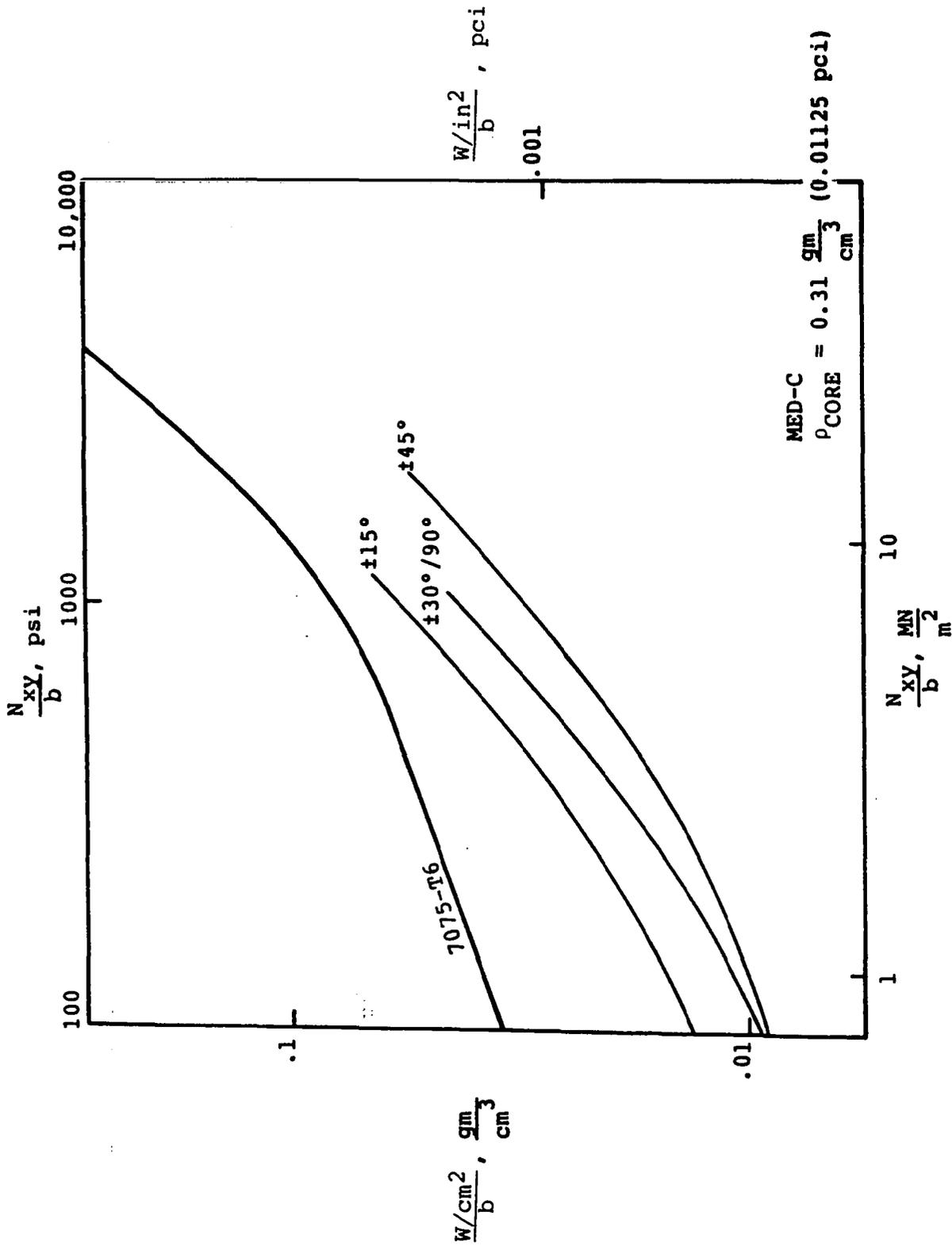
Figure B-31. Candidate Reinforcement Configurations and Materials for Research and Development for Flat Plate Shear Applications.
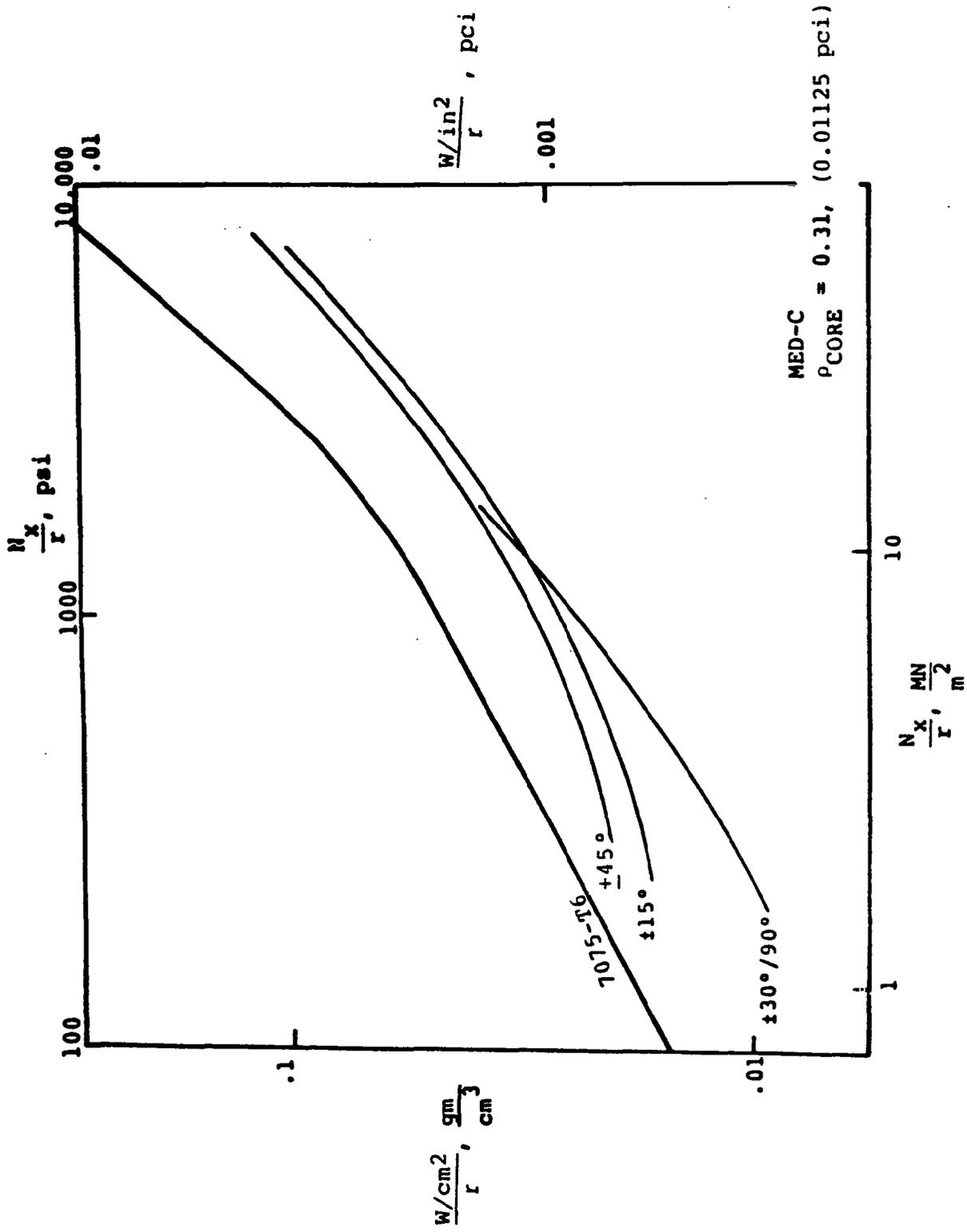
Figure B-32. Candidate Reinforcement Configurations and Materials for Research and Development for Shell Applications.
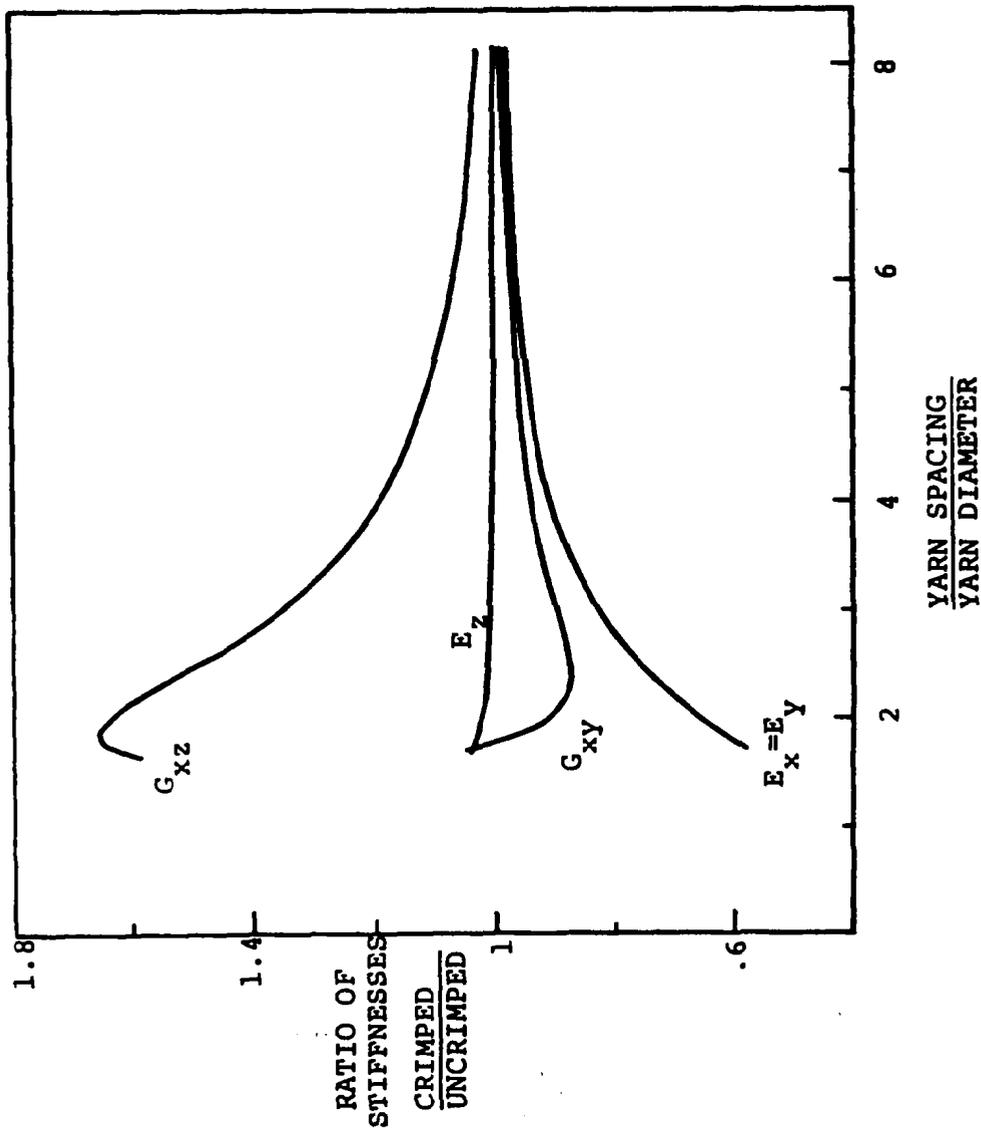
Figure B-33. Effect of Yarn Crimp Due to Weaving on Stiffnesses of Woven Fabric Reinforced Composites.
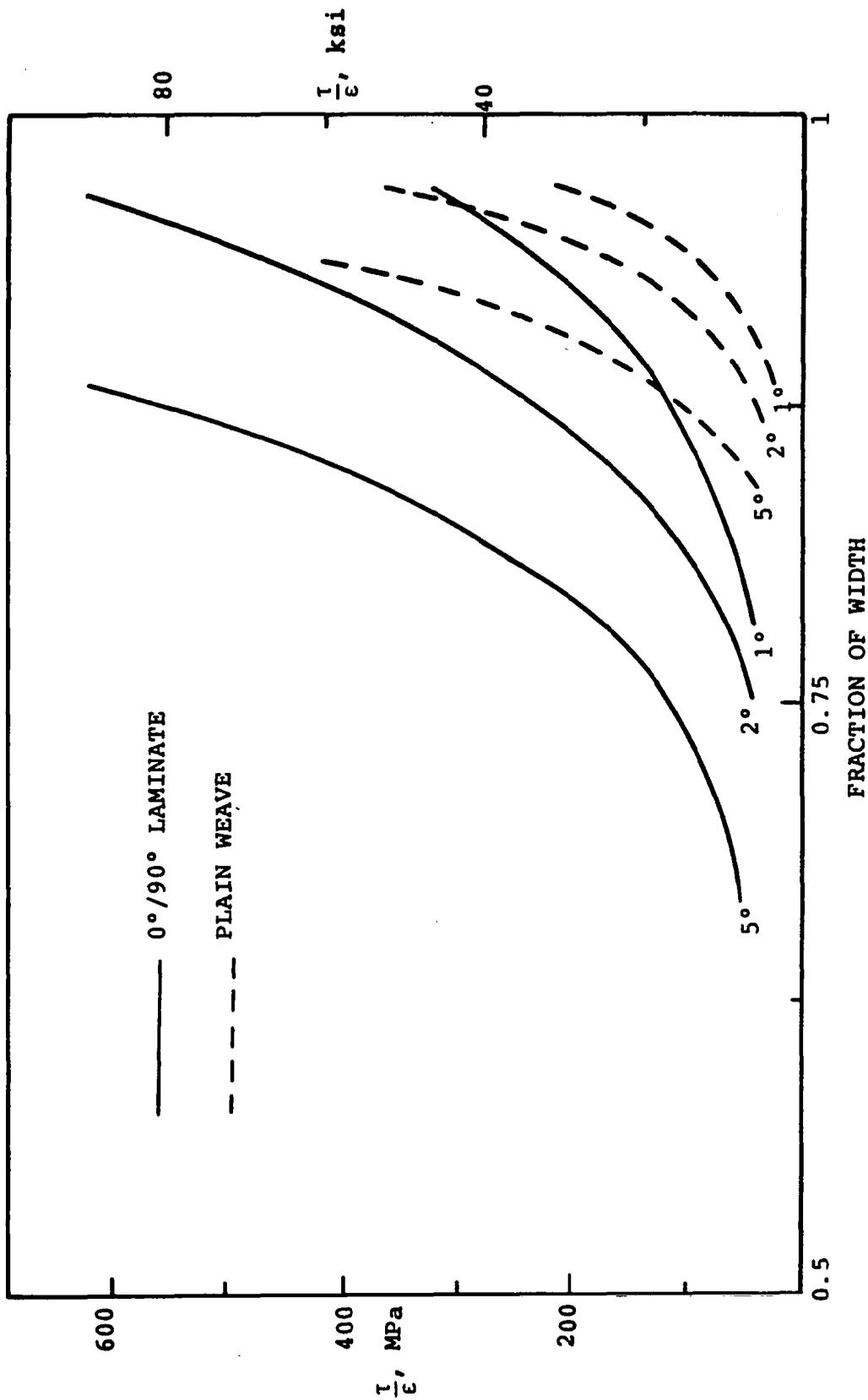
Figure B-34. Interlaminar Shear Stresses Calculated for Slight Angularity of Load Application to 0°/90° and Plain Weave Reinforced Composites.
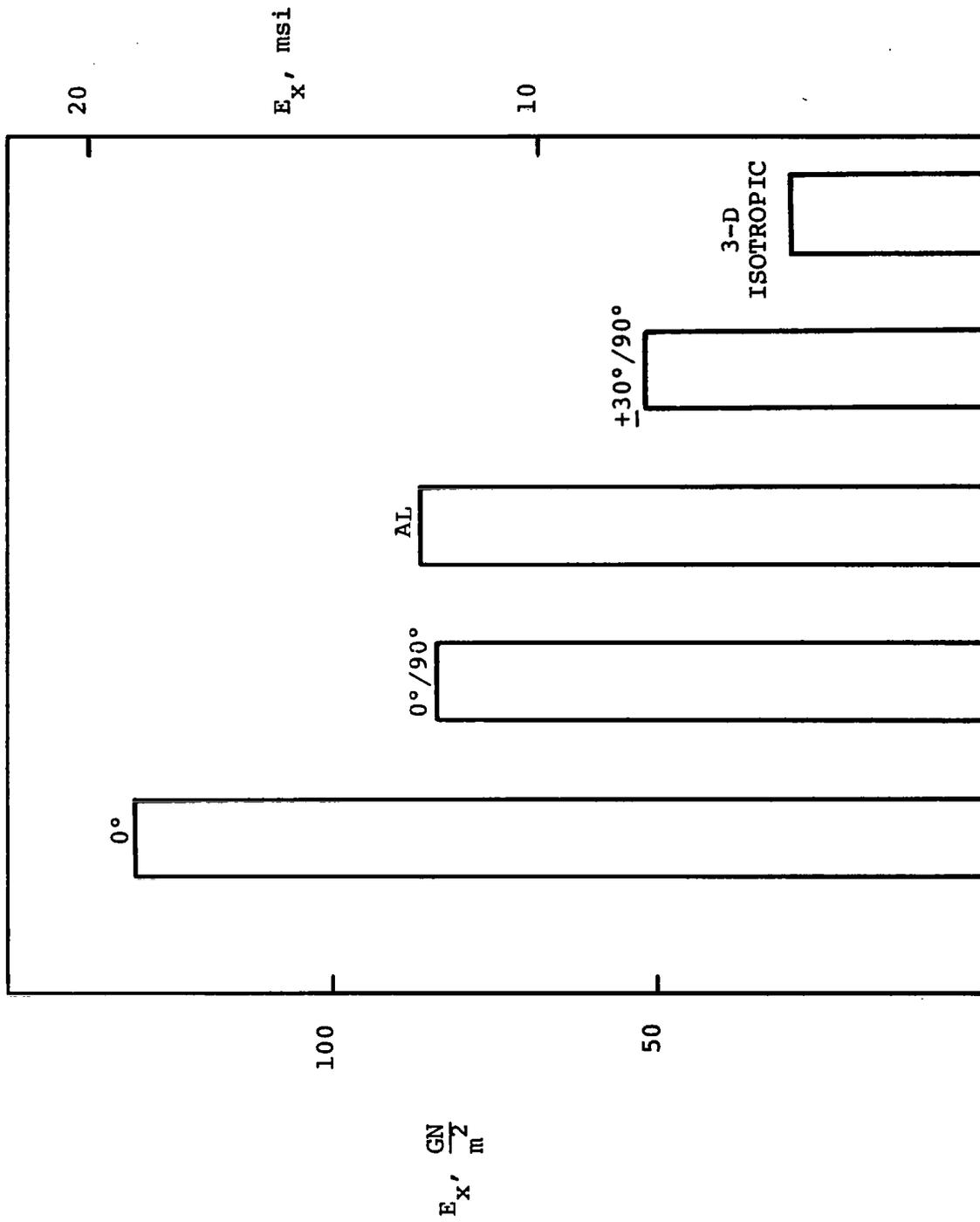
Figure B-35. Simple Relationships Between Numbers of Reinforcement Directions and Stiffnesses for Constant Volume Fraction Reinforcement.

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APPENDIX C - STUDIES OF STRUCTURAL COMPONENTS

STUDIES TO UTILIZE RESULTS FROM RESEARCHES ON ELEMENTS FOR THE
DEVELOPMENT OF ADVANCED COMPONENTS

The term "structural components" is used here to designate
assemblies of elements. In general, the components will be
considered to be subjected to simple loadings; their interac-
tions with other components will be treated under the heading
"structures", even as the interactions between elements within
the component are treated here for the component. The typical
example of a component is a stiffener section, - an assemblage
of plate or shell elements.

Stiffener Sections

Stiffener sections for composites may need to differ from
those for metals for several reasons, - because of differences
in methods of manufacture, differences in properties, differ-
ences in methods of attachment. In this section we will be
concerned with the latter two reasons. Manufacturing aspects
will be addressed in Appendix E.

Overall Configuration

A first task is the selection of a baseline family of over-
all configurations of stiffeners for study and development.
Exploratory research appears essential here if this selection
is to be done successfully. Consider, for example, the three
stiffener configurations shown in figure C-1. All three con-
figurations have been extensively studied in aluminum alloy
by the NACA, experimentally (refs. 7 - 9, and many others),
and theoretically (refs. 10 - 11, and many others). As a re-
sult of all the studies, characteristics and proportions of

the basic shapes have been thoroughly defined which will insure successful application — particularly, both local buckling and ultimate strength properties have been determined. No similar information base is available for the anisotropic composite materials.

Total repeat of all the extensive aluminum alloy studies is certainly not required to develop composite stiffeners, but there are critical questions to be resolved early on. Hence, the need for exploratory studies. These questions include:

(1) Can properties and proportions of composite Z-sections be defined, as they have been for the aluminum alloys, which will cause twisting instability of the section to be not critical, and hence, the Z-section to be both a simple and efficient component?

(2) Can the semi-empirical rules developed for the determination of the relationship between buckling stress and ultimate strength for sections made from a ductile material like aluminum (for example, ref. 12 be extended to materials like composites?

(3) Is the attachment flange design as crucial with bonded joints as it is for riveted joints (see ref. 13)? Are flanges extending both sides of the webs, as suggested pictorially in figure C-2, of merit, and worth the added cost of manufacture?

The exploratory studies should be directed toward the definition of a minimal number of baseline composite stiffener configurations to use for the generation of buckling and ultimate strength characteristics and criteria.

Effects of Use of Curved Elements

The exploratory studies of stiffener configurations should include evaluations of the potential for performance improvements associated with the use of curved instead of flat plate elements within the stiffener (see fig. C-3). Curving the webs of aluminum-alloy Y-section stiffeners (ref. 14) was found effective in raising the strength and efficiency of the section.

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The principle should be extensible to composite materials and other stiffener shapes.

In order to estimate the potential of the use of curved plate elements, an analysis was made on the basis that the buckling stress is the same for the curved element whose chord is the plate width as that of a full cylinder of the same thickness and curvature. The orthotropic buckling equations (B-3 and B-4) were employed, and the curvature and thickness were optimized for a given load on the plate to provide a minimun weight for buckling. The weight of the curved section was calculated to account for the length added due to curvature from the equation for the arc length ℓ

$$\ell = \frac{r}{90} \sin^{-1} \left(\frac{b}{2r}\right) \tag{C-1}$$

where

    r radius of curvature

    b flat plate width

(It can be shown that following this procedure for elastic buckling, $\ell = 1.175b$, or equivalently, that the angle subtended by the arc is constant for minimum weight at 1.935 radians, regardless of the buckling stress. Above the elastic limit, the radius of curvature increases, reaching infinity when the width/thickness ratio is equal to that for elastic flat plate buckling. The weight penalty for the use of a maximum subtended angle of only $\frac{\pi}{2}$ radians compared to 1.935 radians is negligible.).

Results of the calculations for 7075-T6, for the MED-C baseline composite in the $\pm 30°/90°$ configuration, and for both these materials on a medium-low density core are shown in figures C-4 to C-6. The curvature alone is shown to have a potential for up to 60% weight saving compared to flat aluminum plates (fig. C-4). The saving with the MED-C material compared to flat aluminum increases to as much as 72% (fig. C-5). With

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a medium low density core, the weight saving is over 80%. The potential for weight saving appears substantial. For lightly loaded sections, common in general aviation aircraft applications, the use of curved elements should be fully evaluated.

Multi-Web Beams

The study of multi-web beams as components ～i.e. idealized as regards loading conditions and neglecting tension cover requirements, - provides useful information without the complexity of complete wing structures. Such idealized beams can provide guidelines about efficiencies achieved with materials, configurations, and approaches. As will be shown, however, the consideration of the role of the tension cover can be of vital importance, particularly its influence on design for minimum part count. Herein, first overall evaluations are made of idealized multi-web beam components, following the methodology developed in reference 15. Then the influence of the tension cover properties are considered, and directions for research and development derived accordingly.

Overall Evaluations

To provide a yardstick for comparison, the efficiencies of 7075-T6 aluminum-alloy beams were first calculated as in reference 15. Results are shown in figure C-7 both for solid construction and for a sandwich construction utilizing the medium-low density core material of Appendix B. The weight saving associated with the medium-low density core material is substantial, - note the logarithmic scales. Actual percent weight savings are between 50% and 60% of the solid construction. Equally significant are the implications for part count reduction, as the web spacings can be approximately doubled with no weight increase by

the use of the medium-low density core sandwich.

Calculations, on the same basis as for the 7075-T6, for the MED-C material with $\pm 30°/90°$ faces on the medium-low density core material yield the results plotted in figure C-8. Except in the high loading range, this material is shown to offer weight saving potentials over those for the 7075-T6 sandwiches greater than the 50% - 60% of the aluminum sandwiches over solid aluminum.

Finally, for comparison, the weight-efficiencies of Z-stiffened panel covers of the $\pm 30°/90°$ MED-C material were calculated by the procedures of reference 18 and plotted together with the curves for the multi-web beams in figure C-9. Here the curve for the stiffened-panel construction represents strict weight optimization - utilizing the full material strength/weight characteristics of the MED-C material. It is accordingly the least weight construction, but it is also by far the highest part count, - almost without question prohibitively so. (Studies of relative costs of multi-web and skin-stringer construction are reported in Appendix E.) Morever, as will be shown in the following section, utilizing the full strength of compression cover material is often not possible. Stiffened panel construction was not investigated further.

The summary results shown in figure C-9 need to be put into proper perspective by relation to the load intensity ranges of concern. Maximum (wing root) values of $\frac{M}{bd^2}$ for the same four representative general aviation aircraft considered in figure B-29 were accordingly calculated. Results are plotted in figure C-10. Comparisons between figures C-9 and C-10 reveal that for the load values of interest, the medium-low density core MED-C material, is appropriate for an efficient single-cell (two-web) construction $\sim$ a minimal part count design. Unfortunately, these $\frac{M}{bd^2}$ values are maxima; over the rest of the wing (and tail surfaces) loadings are less, and multiple webs are

required for comparably efficient construction.

Design for Minimum Part Count

Design for minimum part count is primarily a matter of min-
imizing the amount of supporting structure required to prevent
buckling of the elements which are loaded in compression. The
problem is a complex one, and the studies in this area reported
here are aimed primarily at indicating problem areas and approach-
es requiring further research and development.

Preliminary calculations indicated early on that to treat
the compressive side of the beam structure independently of the
tension structure could be misleading. The stress in the compres-
sion cover is dependent not only on the applied moment but also
on the effective moment of inertia of the section and especially
on the location of the neutral axis of the section. Matching
materials strengths, buckling strengths, effective moments of
inertia, and neutral axis locations for efficient material utili-
zation with composite materials and their configuration-depen-
dent properties becomes sufficiently complex as to require a com-
puter program. The program BOZO was created for that purpose,
and its outputs will be reported primarily in Appendix D. Portions
of the results relating to component design for minimum part count
are reported here.

The One Cell Box Beam - Proper Balancing of Material Properties

The use of only two shear webs provides a model for mini-
mun part count, and results of calculations of weights required
for such a simple component provide directions for further study.
Such a box beam made of 7075-T6 aluminum alloy with a medium-low
density core yields weight/efficiency values as shown in figure C-
11, when now both tension and compression skin thicknesses have
been optimized for minimum weight. Corresponding results for

MED-C in a ±30°/90° reinforcement configuration are shown in figure C-12. Both of these plots represent simple cases for which the tension material is much the same as the compressive material, and the optimization is to trade off core weight against buckling stress.

Slight differences did exist, however, in the tensile properties used for these calculations. The tensile ultimate used for 7075-T6 was 483 $\frac{MN}{m^2}$ (70 ksi) and the compressive yield stress used was less than that, 414 $\frac{MN}{m^2}$ (60 ksi). For the MED-C the stresses were 414 $\frac{MN}{m^2}$ for tension and 473 $\frac{MN}{m^2}$ (68.6 ksi) in compression. This slight difference noticeably affected the high end of the MED-C curve where the stresses were near ultimate. The upsweep is the result of the neutral axis shift resulting from thicker tension flange required for the MED-C material.

While the effect just described was a small one, so were the input differences causing it. In order to bring out more fully the magnitude of similar effects that may be encountered, further calculations were made for hybrid beams having MED-C/±30°/90° compression covers with medium-low density cores and unidirectional Kevlar tension covers. (Properties used for the Kevlar are given in table B-1.)

A simplistic view would suggest increased efficiency for such a hybrid beam due to the high tensile strength/density ratio of the Kevlar. Actual results show weight efficiencies nearly identical to those for the all-MED-C beams. The reason is that optimum proportions never could utilize the high strength potentials of the Kevlar (for example, the highest Kevlar stress at the lower end point in the $\frac{d}{b}$ = 0.16 curve was 0.5 σ_{tu}).

If box beams are to be designed to take proper advantage of the wide ranges of properties accessible with advanced composites to provide minimum part count the complex interactions introduced by dissymetries need to be fully evaluated.

The One or Two Cell Box Beam - Use of Curved Plate Elements

The use of curved-plate elements at the tops of the shear webs, as shown in figure C-13, should provide additional fixity to delay buckling of the compression cover of the box. Quantitative values of such restraint are not available, but the configuration deserves evaluation.

Needed first are guidelines for proportions, particularly of the curved portions of the shear webs to maximize the restraint provided to the skin between webs. If the restraint coefficients are promising, complete characterizations of the design approach is desirable.

DEVELOPMENT OF DIRECTIONS FOR RESEARCH AND DEVELOPMENT ON STRUCTURAL COMPONENTS

The studies of structural components reported in this Appendix have generated the following directions for research and development -

(1)   Definition of a generic family of stiffener shapes and multi-web beam configurations for advanced composites for characterization and exploitation.

(2)   Definitions of modes of buckling and failure, and determination of ultimate strength characteristics of chosen designs, with particular attention to twisting vs. buckling modes for sections such as Zees, and to buckling vs. crushing modes for webs of multi-web beams.

(3)   Determination of the influence of attachment flange design on strengths of stiffened and multi-web constructions, and generation of guidelines for attachments to generate the potentials of the constructions.

(4)   Evaluations of the influence of fabrication methodol-

ogy, including laminate stacking sequence, on the re-
sulting component characteristics, with emphasis on
minimum gage constructions.

(5) Develop design approaches and constructions for min-
imum part count and low loading intensities.
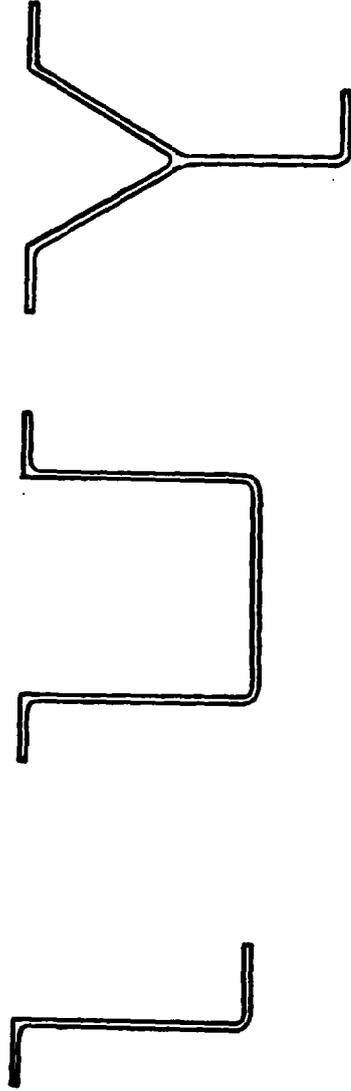
Figure C-1. Generic Stiffener Configurations for Which Performance Characteristics Have Been Thoroughly Defined for Aluminum Alloy Construction.

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Overhang Added to Delay Separation (Peeling) Type Failure
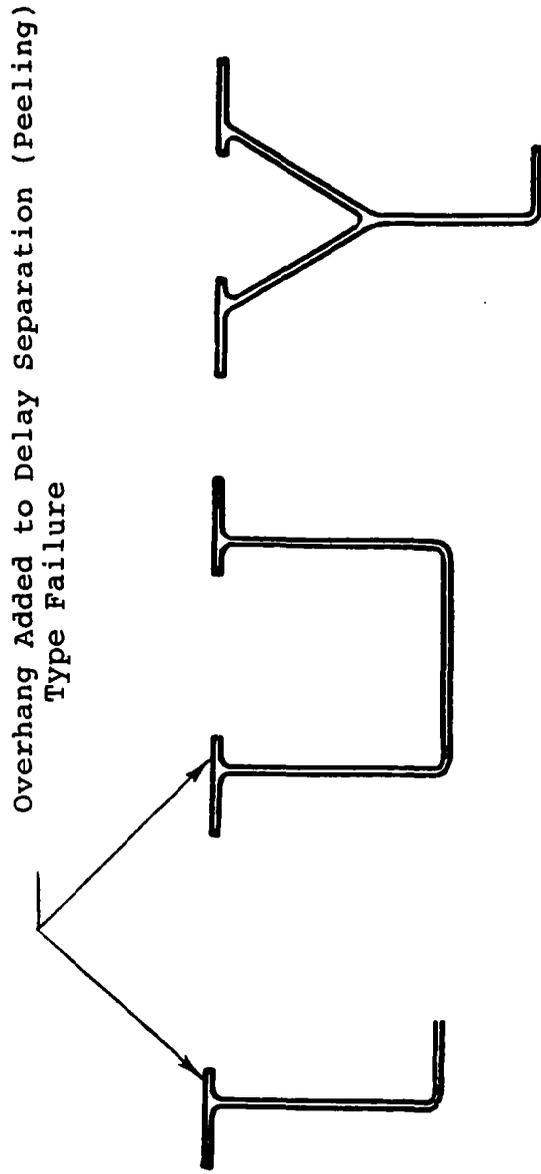
Figure C-2. Pictorial Representation of the Type of Attachment Flange Design Variation from that Shown in Fig. C-1 Needing Evaluation for Ultimate Strength Enhancement.
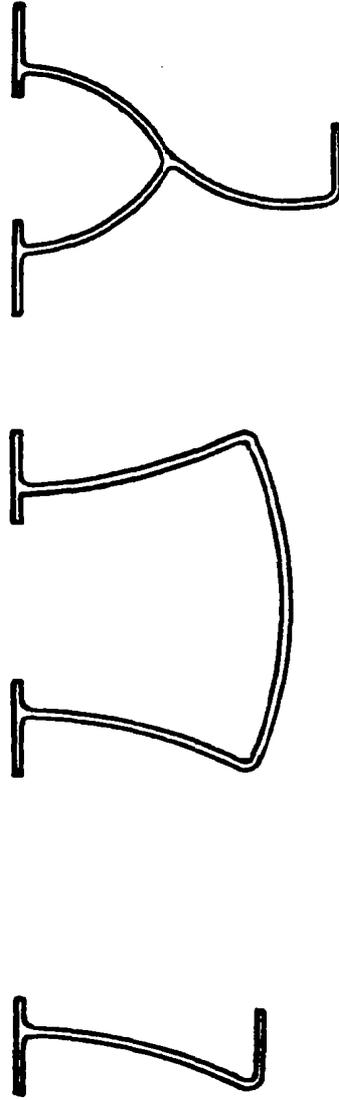
Figure C-3. Pictorial Representation of Use of Curvature to Increase Stability of Flat Plate Elements in Stiffener Sections.

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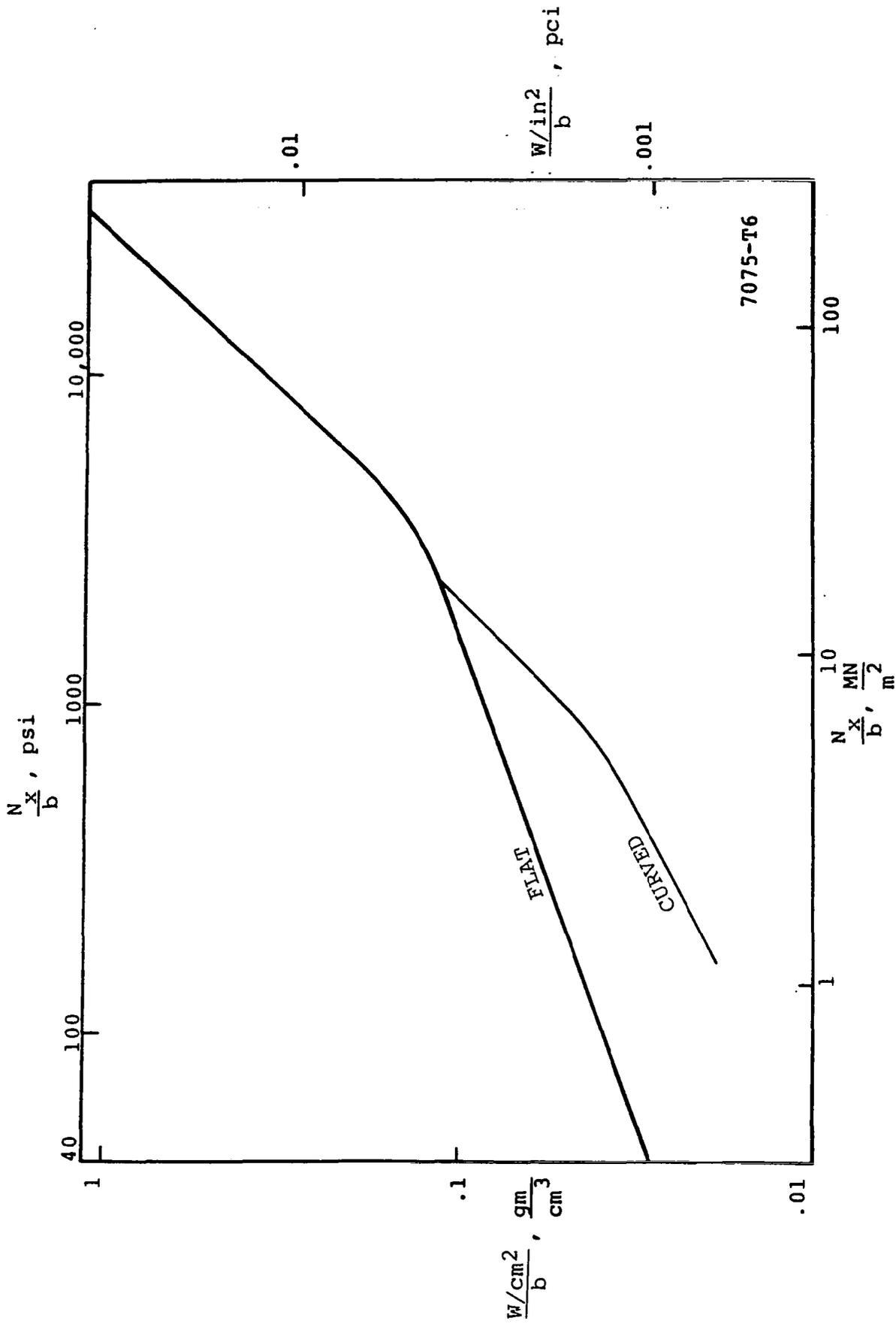
Figure C-4. Evaluation of Efficiency of Curved vs. Flat Plate 7075-T6 Aluminum Alloy Elements in Compression.
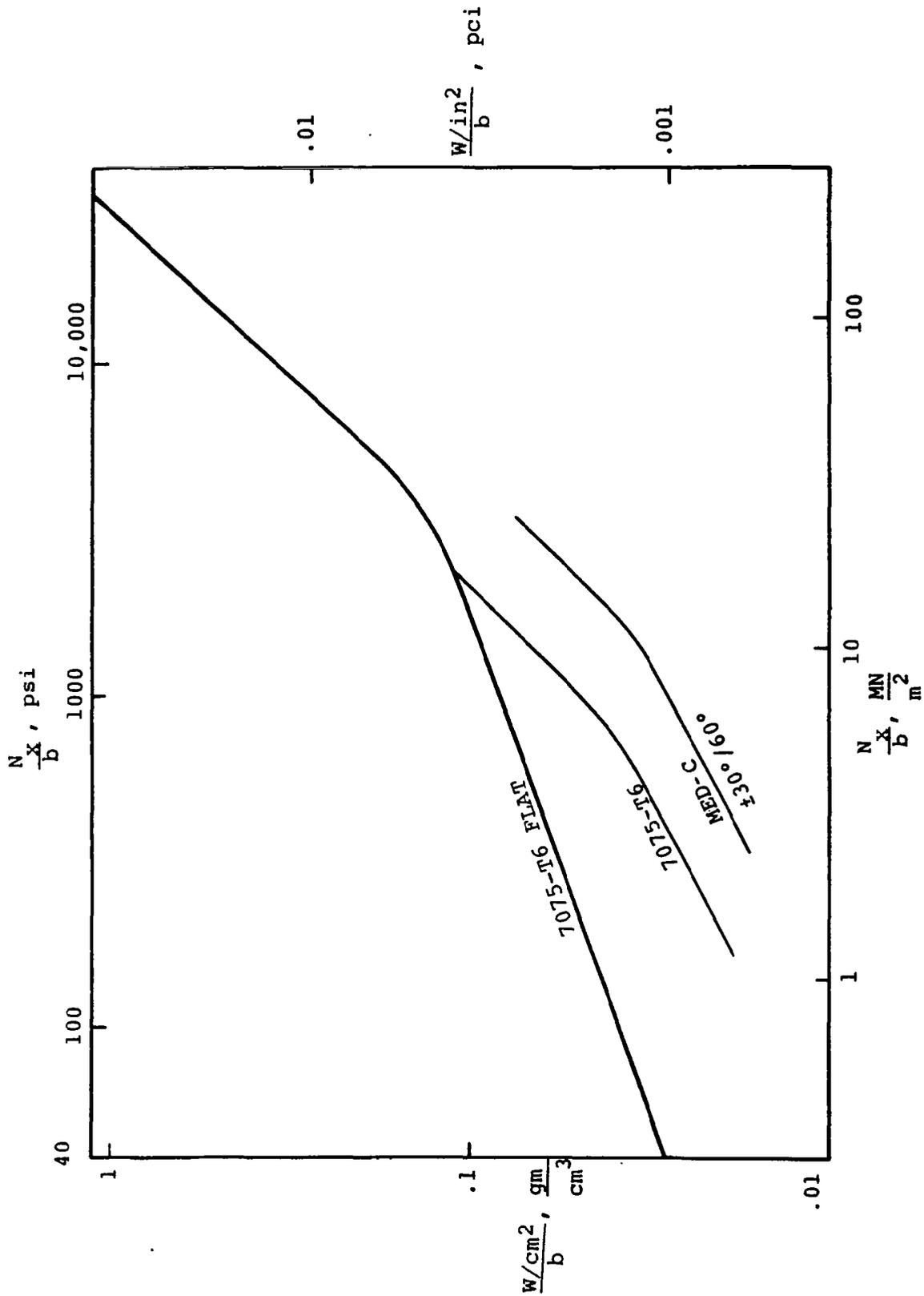
Figure C-5. Evaluation of Efficiency of Curved Plate Elements of Medium-High Modulus Carbon Fiber/Epoxy Compared to 7075-T6 Aluminum Alloy Flat Plates.

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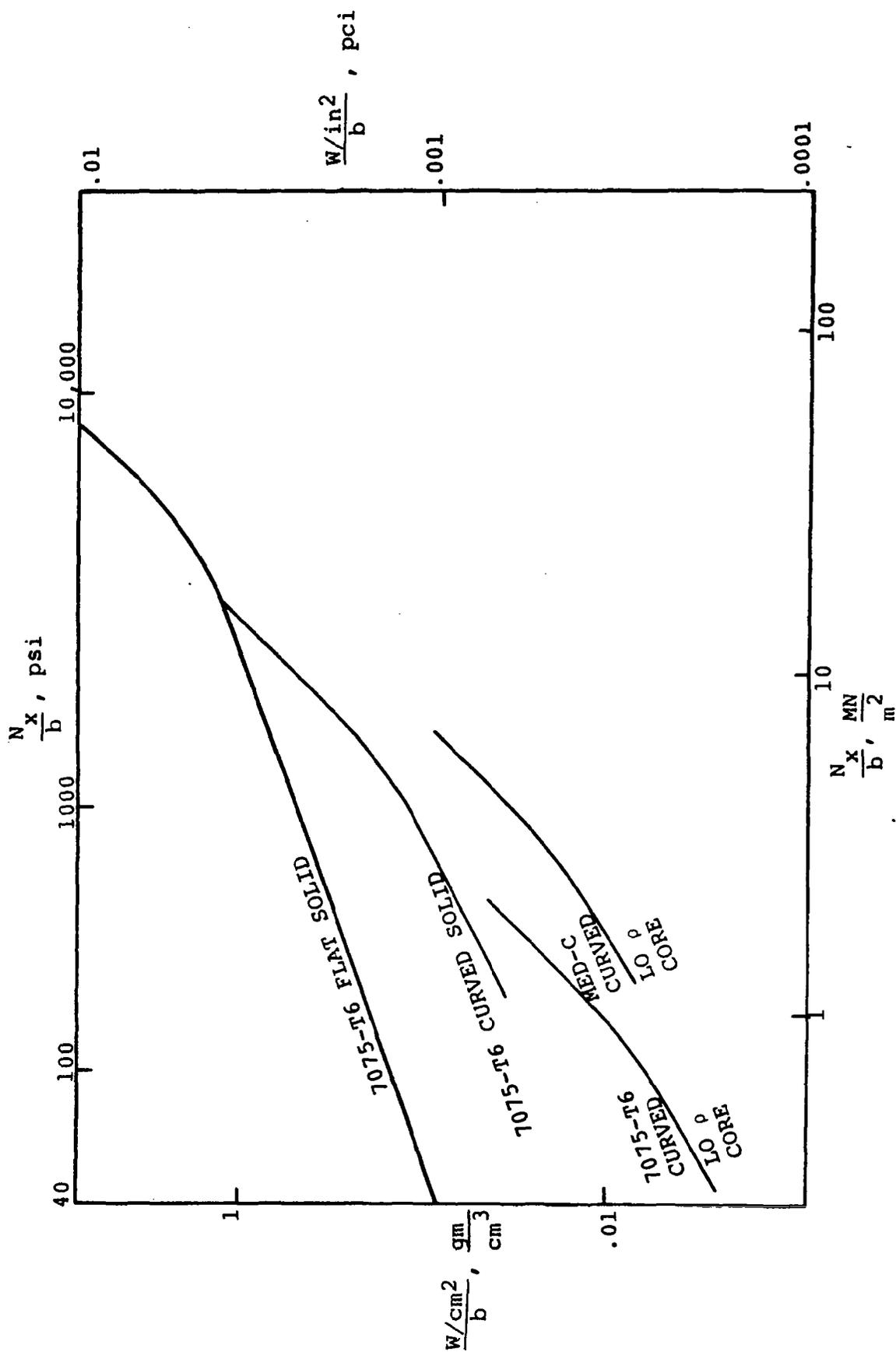
Figure C-6. Evaluation of Efficiency of Curved Plate Elements of Medium-High Modulus Carbon Fiber/Epoxy and 7075-T6 Aluminum Alloy Incorporating Medium-Low Density Cores.
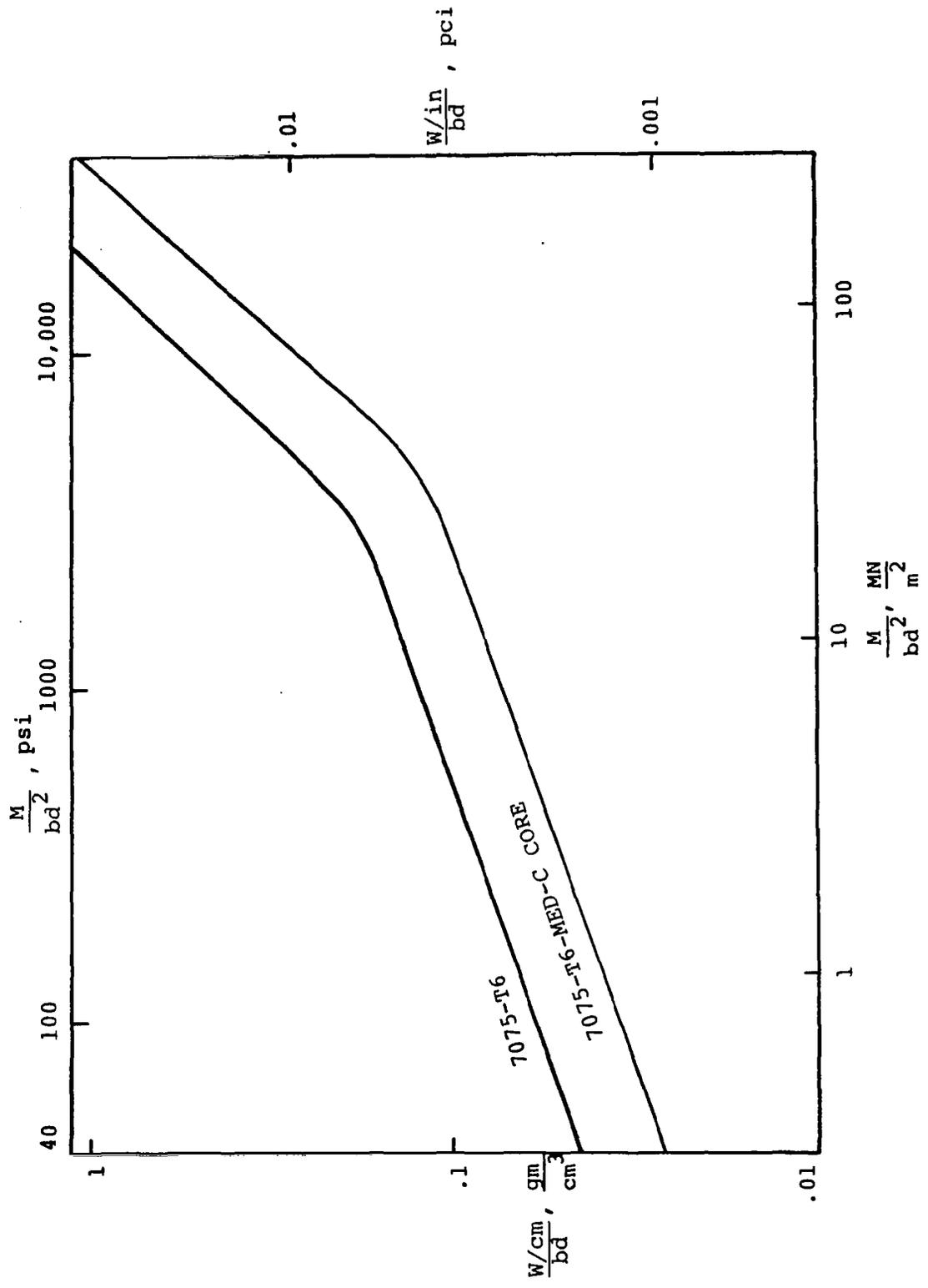
Figure C-7. Efficiencies of Compression Covers (Including Web Weights) of 7075-T6 Aluminum Alloy Box Beams in Bending.

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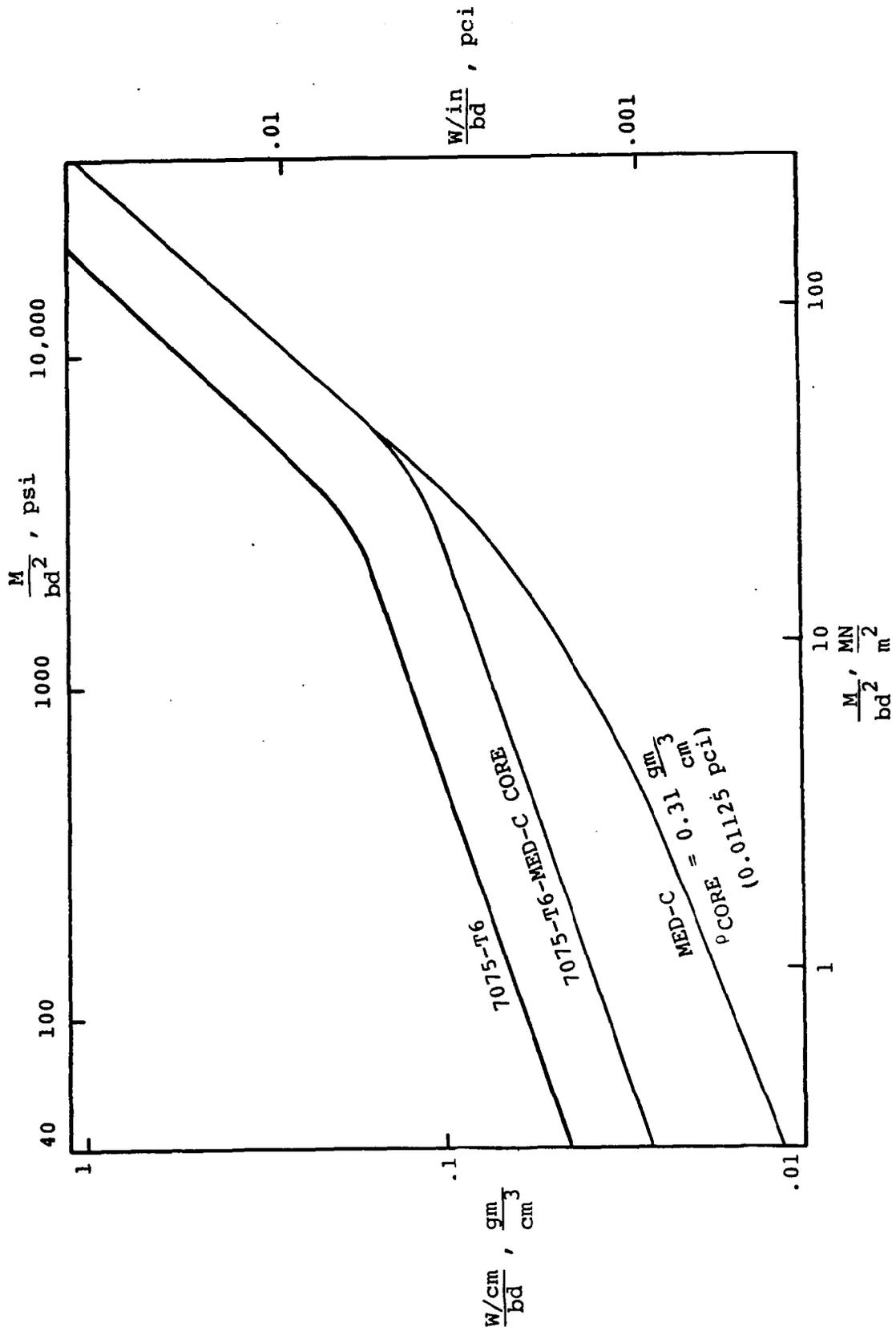
Figure C-8. Comparison of Efficiencies of Compression Covers (Including Web Weights) of 7075-T6 Aluminum Alloy and +30°/90° Quasi-Isotropic, Medium-High Modulus Carbon Fiber/Epoxy Box Beams in Bending.
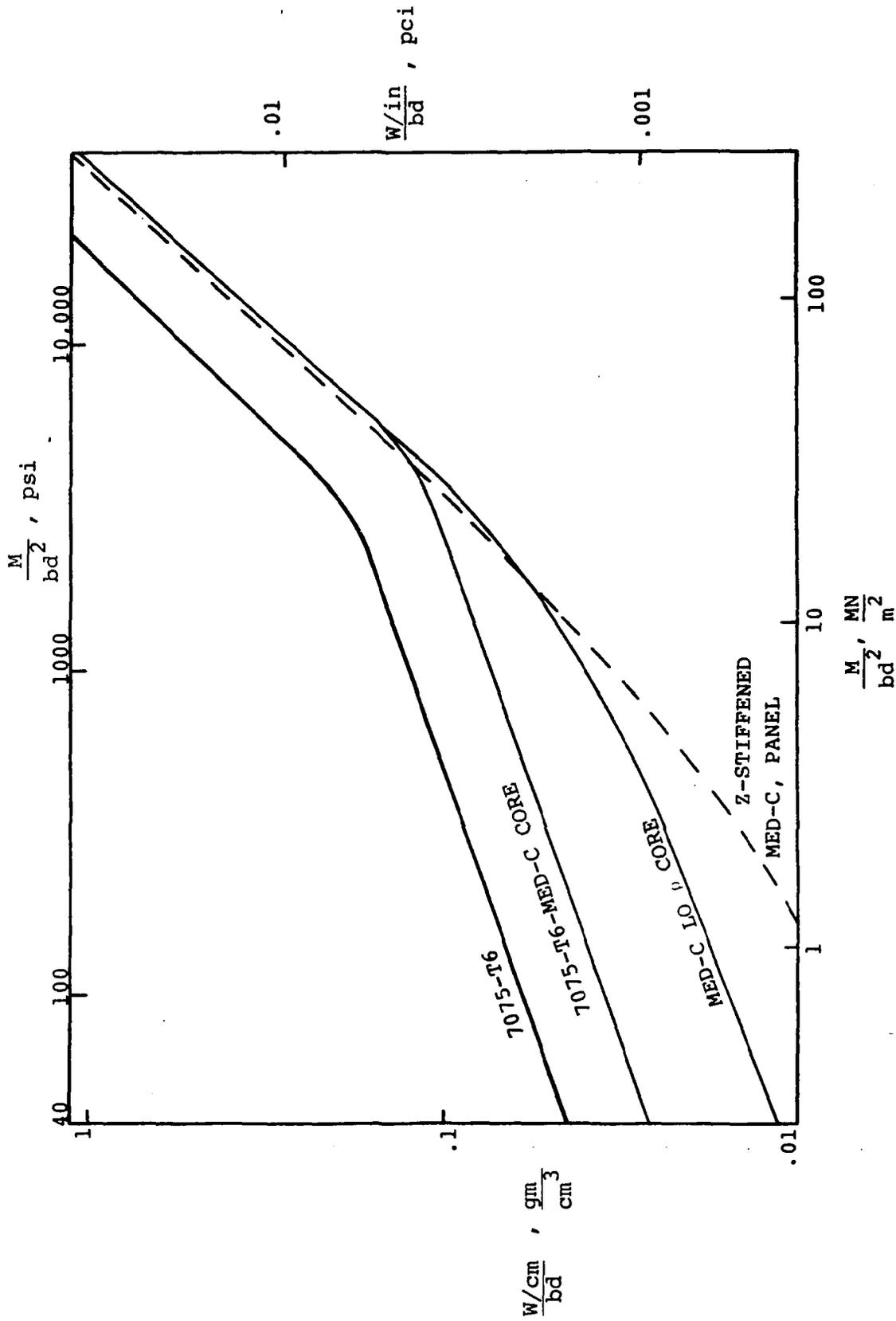
Figure C-9. Comparison of Efficiencies of Multi-Web (Including Web Weights) and Optimized Z-Stiffened Panel (Including Rib Weights) Compression Covers of $\pm 30°/90°$ Medium-High Modulus Carbon Fiber/Epoxy for Box Beams in Bending.
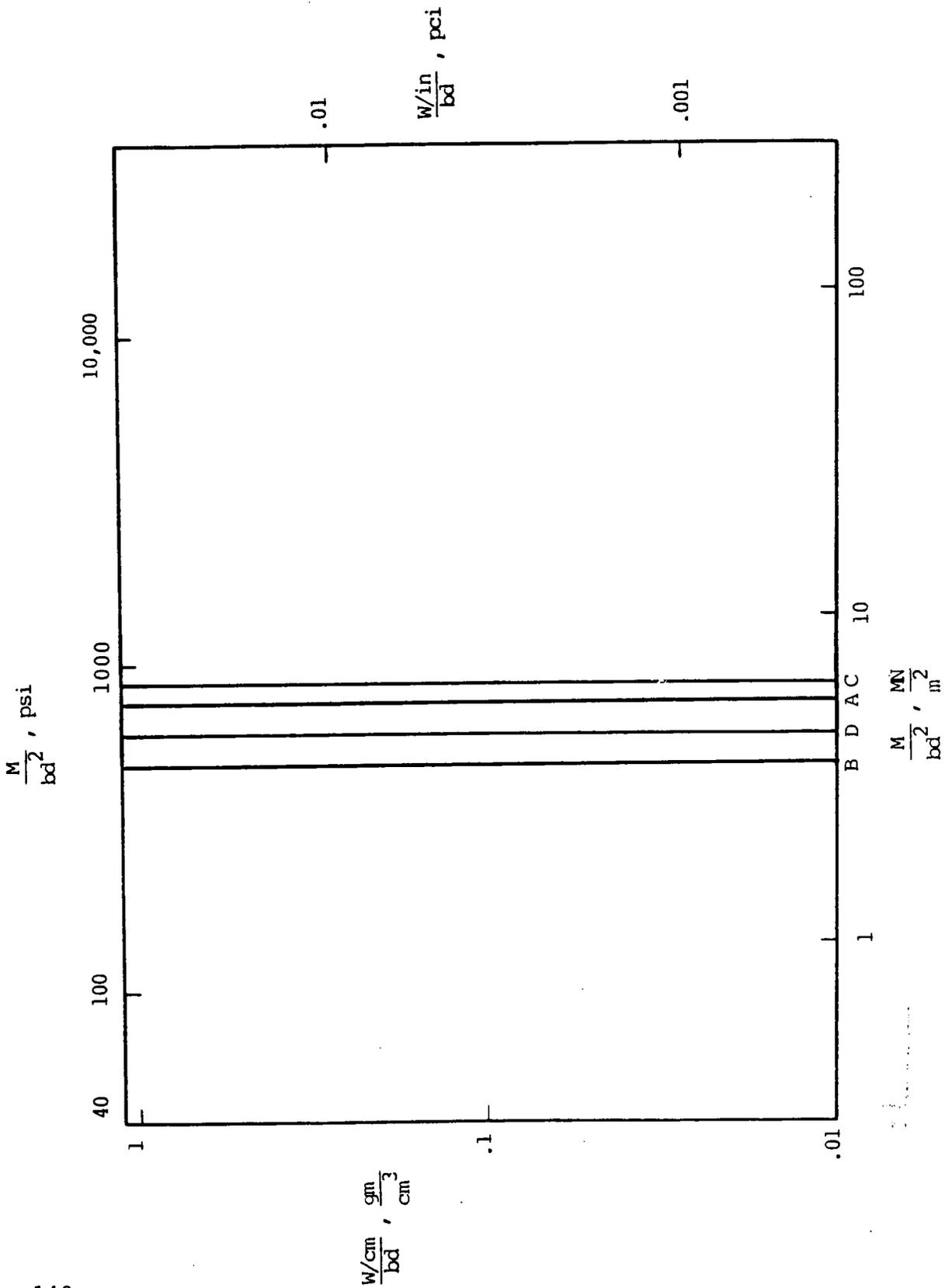
Figure C-10. Bending Loading Intensities at the Wing Root for the same General Aviation Aircraft Considered in Figure B-29.
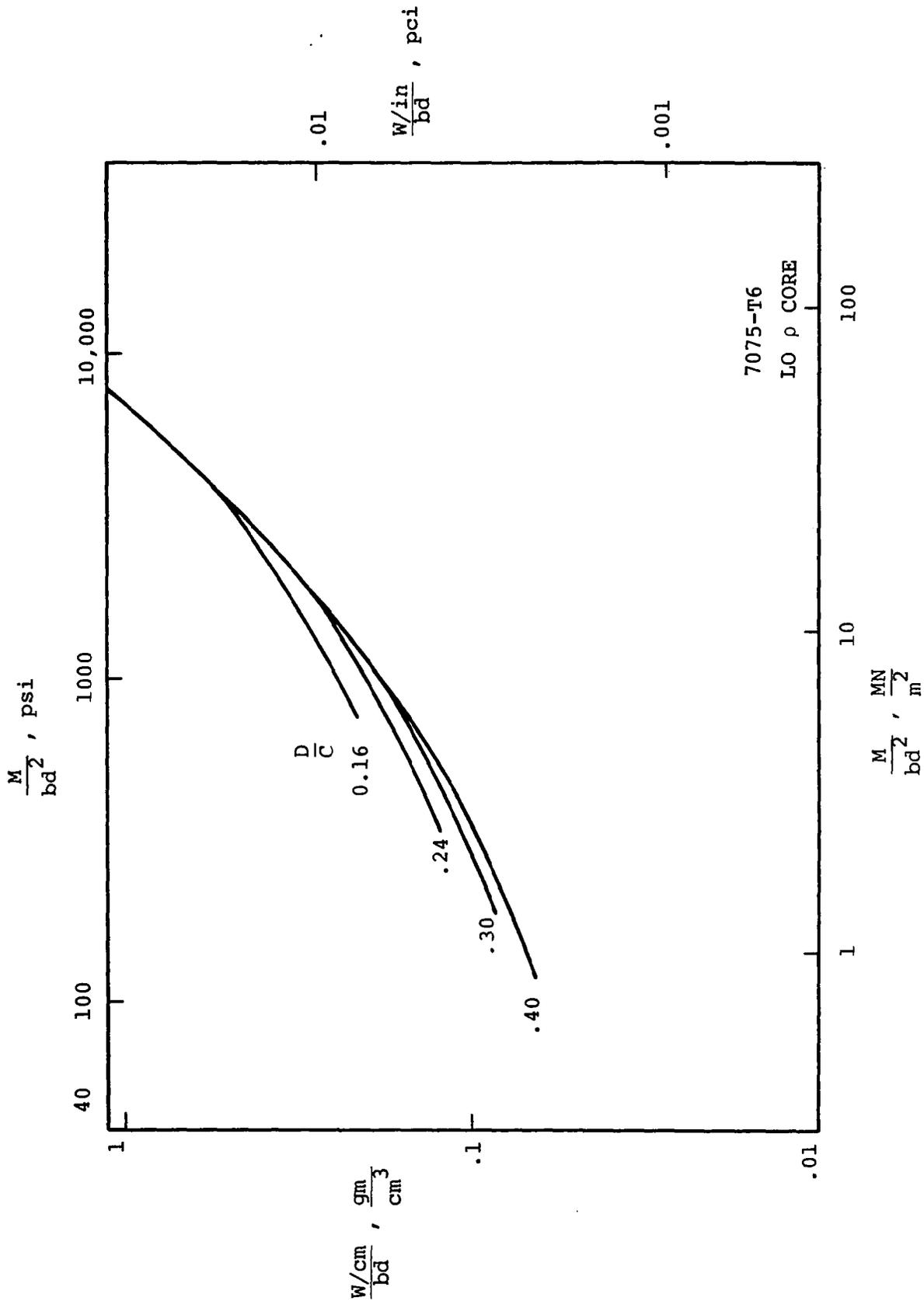
Figure C-11. Efficiencies of Multi-Web Beams in Bending: 7075-T6 Aluminum Alloy, With A Medium-Low Density Core for the Compression Cover.
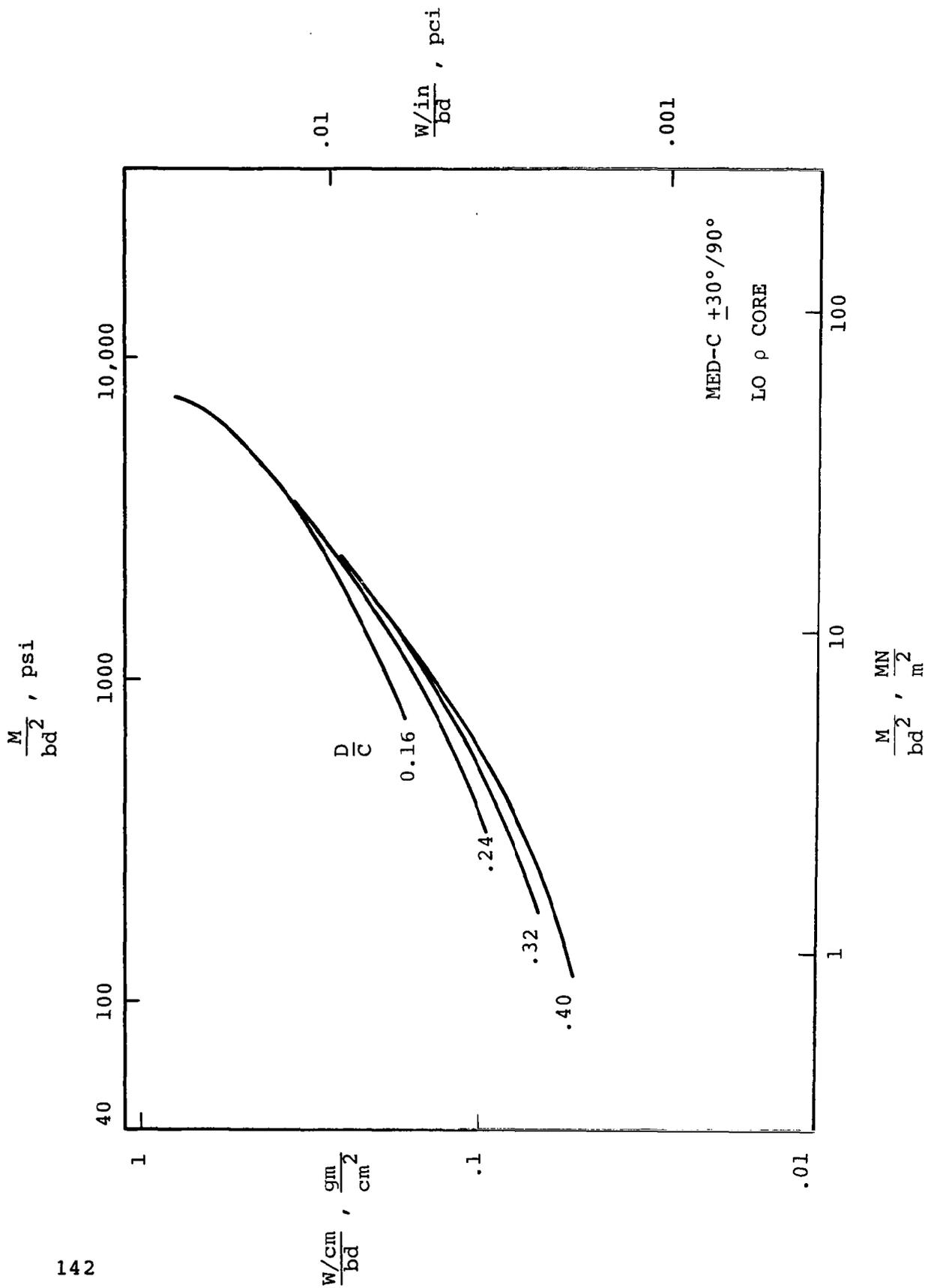
Figure C-12. Efficiencies of Multi-Web Beams in Bending: ±30°/90° MED-C Epoxy, With a Medium-Low Density Core for the Compression Cover.

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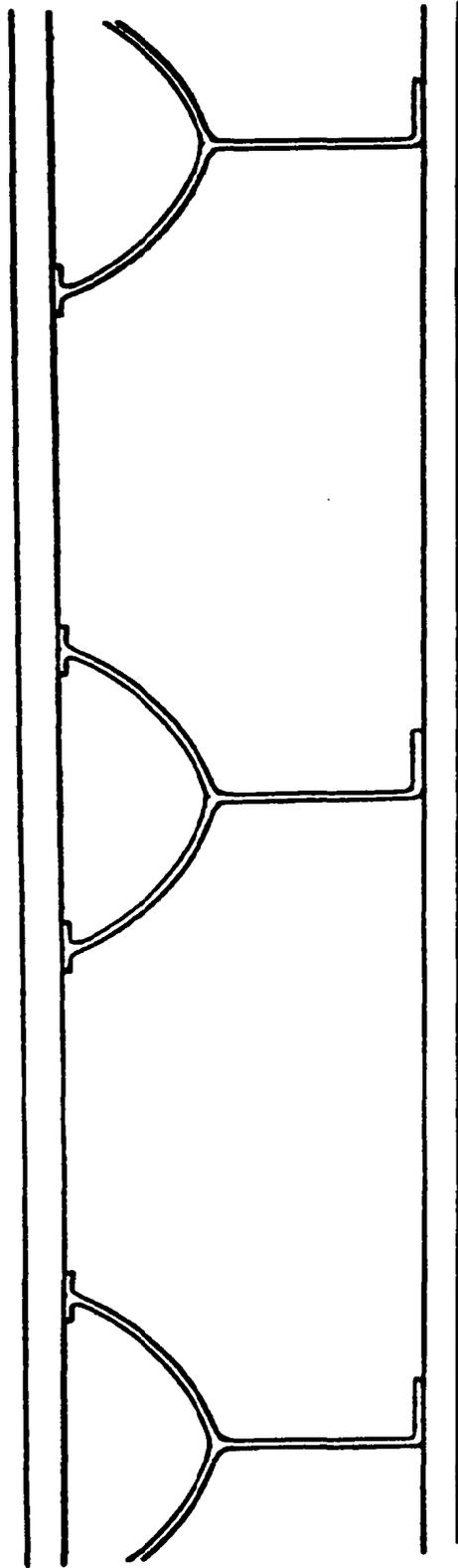
Figure C-13. Use of Curved-Plate Elements to Provide Maximum Fixity to the Elements of the Compression Cover of a Two-Cell Box Beam.

APPENDIX D - STUDIES OF COMPOSITE, GENERAL AVIATION AIRCRAFT STRUCTURES

LIST OF SYMBOLS (not already defined in Appendix B)

$\mathcal{R}$	aspect ratio
C	chord
D	depth
G	shear modulus
I	moment of inertia
J	polar moment of inertia
M	moment
$\mathcal{R}$	taper ratio

Just as the multi-web beam component becomes more complex considered as a whole than when treated without consideration of the influence of the tension cover, so does the actual wing box produce significant complications by comparison to the multi-web beam component. Chief among these for general aviation aircraft are the low loading intensities and resulting minimum gage problems. Another complication is the stiffness requirement. The influence of such factors on directions for research were evaluated by a series of parametric and trial designs, as described in the following sections of this Appendix.

Studies of Wing Structures

Methodology

A computer program, herein designated BOZO, was developed to calculate minimum-weight box beams having sandwich covers (core density = $0.08 \frac{gm}{cm^3}$) (0.0029 pci), the configurations shown

schematically in figure D-1. The program has several options: (1) it calculates minimum-weight boxes to carry a given bending moment by optimizing skin and sandwich thicknesses for input material properties; (2) it calculates the bending and twisting stiffnesses EI and GJ of the box beam as optimized; (3) it repeats this process for a series of spanwise stations on a wing, relating the bending moment to aircraft gross weight, span, load factor, etc. to achieve an overall wing design; (4) it summarizes characteristics of the overall design, such as tip deflections and percent of wing area for which minimum gage constraints rather than buckling or strength govern the design.

For this report only those BOZO results most pertinent to the questions of research direction will be considered: these include:

(1)   Development of a generalized method of presentation to facilitate comparisons among various design approaches, and use of such comparisons to guide research planning.

(2)   Detailed design studies of idealized general aviation type aircraft as a guide to problem areas and pay-offs for the use of composites.

(3)   Assessment of the potential of extensions of programs more sophisticated than BOZO to aid in the application of composites to advance the overall state of the art of general aviation aircraft design.

Generalized Results

The type of generalized results obtainable from a program such as BOZO are illustrated in figures D-2 and D-3. Plotted against the same non-dimensional parameters ($\frac{W/m}{CD}$ and $\frac{M}{CD^2}$) used in Appendix C for the multi-web beam studies are weight/efficiency curves for constant cross-section structural boxes of varying aspect ratio $\frac{D}{C}$ (the solid curves on the figures). Shown, as is to be expected, is that there is a weight penalty for reducing wing thickness (de-

creasing $\frac{D}{C}$). More importantly, however, the magnitude of the penalty is quantified, is shown to be greater for 0° reinforcements (fig. D-2 compared to D-3), and greater percentagewise at low loads than high loads (greater separations between the curves to the left on the figures).

Even more importantly, related parameters can be cross-plotted on these generalized plots. By way of illustration, the stiffness parameters EI and GJ have been superposed on the weight/efficiency curves as shown. Thus, the relative stiffness characteristics of comparative designs as well as their weight/efficiencies are clearly evident. Here, in the load range $\frac{M}{CD^2} = 2$ to 4×10^6 $\frac{N}{m^2}$, for example, it is shown that the 0°/±45° reinforcement configuration not only is the lighter in weight but also has nearly ten times the torsional stiffness with little loss in bending stiffness.

Other factors, such as wing taper ratio and aspect ratio can also be cross-plotted on such generalized representations. The task is a large one to cover the many possible combinations, perhaps well suited to computer graphics, as a valuable tool for design.

Design Examples

The generalized example cited above noted a range of values of $\frac{M}{CD^2}$ (comparable to that identified as appropriate for general aviation aircraft in fig. C-10) for which the 0°/±45° reinforcement configuration was found lighter in weight than the 0°. Further inspection of figures D-2 and D-3, however, also shows that at higher values of $\frac{M}{CD^2}$ the 0° configuration is more efficient. Because higher values of $\frac{M}{CD^2}$ are accessible by, for example, the use of a two-cell rather than one-cell beam, - see figure D-1, valid comparisons of various materials and reinforcement are best made on an overall wing design basis. This was done using BOZO for the generic family of general aviation aircraft (identified

A, B, C, D in figures B-29 and C-10) and the results are reported and discussed in this section.

The characteristics of the example designs are presented in Tables D-1 to D-4. While the gross weights varied from 1000 kg to 8000 kg, the values of the bending moment parameters at the wing root $\frac{M_R}{C_R D_R^2}$, influenced by other factor such as aspect ratio, taper ratio, wing loading, etc. stayed in the narrow range between 3.37 X $10^6 \frac{N}{m^2}$(489 psi) and 6.06 X $10^6 \frac{N}{m^2}$(879 ksi).

Results of calculations are presented only for the one - and two-cell box beams pictured in figure D-1, in order to focus on minimum - part-count construction. Results are given for the following quantities:

Total wing box weight.

Tip deflection at maximum load.

Values of the efficiency parameters $\frac{M}{CD^2}$ and $\frac{W}{CD}$ at the root section.

Values of the bending and twisting stiffness parameters $\frac{EI}{CD^3}$ and $\frac{GJ}{C^2 D^2}$.

Values of the percent of the compression surface for which the thickness is a minimum gage (0.38 mm)(0.015 inch).

Values of the percent of the compression cover which utilizes the material to its yield stress.

Wing Box Weights

Wing box weights were a maximum for the 7075-T6 and a minimum for the MED-C material for all four aircraft. Maximum weight savings (approximately 45%) were for the two cell beams and 0° reinforcement configurations. E-Glass averaged 19% lighter than 7075-T6.

The hybrid beams of MED-C and Kevlar gave mixed results. For the one-cell construction their weight was nearly identical to that for the $0^O/\pm 45^O$ MED-C. For the two cell construction, however, the hybrids were the lightest of all considered, parti-

cularly for the heavier aircraft.

Stiffnesses

The bending stiffnesses and corresponding tip deflections for the 7075-T6 and MED-C constructions were virtually the same for all four aircraft. The hybrid MED-C/KEV was about 50% less stiff. The E-Glass deflections were enough greater to be of concern, - running about 20% of the semi-span.

The torsional stiffnesses of the 0° E-Glass wings (measured by the values of $\frac{GJ}{CD^2}$ at the root) were an order of magnitude less that those for the 7075-T6. The MED-C $0^\circ/\pm45^\circ$ configuration had slightly higher torsional stiffnesses than the 7075-T6, but the 0° MED-C ran about 25° below the E-Glass values, and the 0° MED-C/KEV was lower yet $\sim$ running about 1/20 to 1/25 of the torsional stiffness of the aluminum-alloy construction.

Minimum Gage Construction

As is to be expected, the percentage of wing skin material that is of minimum gage thickness varies with the gross weight of the aircraft, as does the related percent of material worked to the yield stress. The significant result here is the magnitude of the percent of the material that is of minimum gage. It was approximately 60% of the wing skin for the small aircraft $\sim$ somewhat less for the $0^\circ/\pm45^\circ$ MED-C, somewhat more for the hybrid 0° MED-C/KEV. Clearly a need exists for a minimum gage construction of integrity such as a one-ply woven-fabric reinforced composite.

DEVELOPMENT OF DIRECTIONS FOR RESEARCH ON GENERAL AVIATION AIRCRAFT STRUCTURES

The studies of general aviation aircraft structures reported in this Appendix have generated the following directions for re-

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search and development -

 (1) Definition of the need for carrying forward a general-
ized approach to the design of aerodynamic surfaces
(wings and tail) to unify the design process to meet
loads and stiffness requirements. Development should
integrate completely with aerodynamic and aeroelasticity
specifications to provide the designer with the capa-
bility of properly trading off one against another.
A computer program probably strongly dependent on com-
puter graphics appears indicated.

 (2) Development of guidelines for solutions to the problems
of fabrication of actual structures as opposed to lab-
oratory idealizations. To be considered are effects
of taper, dimensional tolerances and surface smoothness
requirements. Demonstration articles to verify com-
posite applicability in wind tunnel or flight are indi-
cated.

 (3) Considerable emphasis on the minimum gage problem. Stud-
ies relating to the medium-low density core material,
and reinforcing fabrics to provide biaxial and isotro-
pic properties in one ply are indicated.

MATERIAL	NO. OF CELLS	WING WEIGHT, kg (lb)	$\dfrac{\text{TIP DEFL}}{\text{SEMI-SPAN}}$	$\dfrac{M_R}{C_R D_R^2}$ $\frac{N}{m^2}$ (psi)	$\dfrac{W_R}{(\text{SPAN})C_R D_R}$ $\frac{kg}{m^3}$ (pci)	$\dfrac{EI_R}{C_R D_R^3}$ $\frac{N}{m^2}$ (psi)	$\dfrac{GJ_R}{C_R D_R^2}$ $\frac{N}{m^2}$ (psi)	% MIN GAUGE	% FULLY STRESSED
7075-T6	1	64.3 (141.7)	0.0981	5.25X10⁶ (762)	87.7 (0.00317)	3.66X10⁸ (53,100)	1.12X10⁸ (16,300)	60	40
	2	58.1 (128.1)	0.0968	5.25X10⁶ (762)	81.8 (0.00295)	3.66X10⁸ (53,100)	1.04X10⁸ (15,100)	60	40
MED-C	1	48.6 (107.1)	0.0915	5.25X10⁶ (762)	58.0 (0.00209)	3.35X10⁸ (48,600)	9.36X10⁶ (1360)	60	20
	2	38.3 (84.4)	0.0948	5.25X10⁶ (762)	40.8 (0.00147)	3.35X10⁸ (48,600)	8.44X10⁶ (1230)	60	40
E-GL	1	59.8 (131.8)	0.227	5.25X10⁶ (762)	75.6 (0.00273)	1.48X10⁸ (21,500)	1.22X10⁷ (1770)	60	40
	2	58.0 (127.8)	0.227	5.25X10⁶ (762)	49.7 (0.00179)	1.48X10⁸ (21,500)	1.11X10⁷ (1610)	60	40

Table D-1. Calculated Characteristics of 7075-T6 Aluminum-Alloy, and 0° E-Glass and MED-C Wings for a 1000 kg Aircraft.

$\mathcal{R}$ = 7.0

$\mathcal{R}$ = 1.0

THICKNESS = 12%

LOADFACTOR = 6.0

MATERIAL	NO. OF CELLS	WING WEIGHT, kg (lb)	TIP DEFL / SEMI-SPAN	$\frac{M_R}{C_R D_R}$ $\frac{N}{m^2}$ (psi)	$\frac{w_R}{(SPAN)C_R D_R}$ $\frac{kg}{m^3}$ (pci)	$\frac{EI_R}{C_R D_R^3}$ $\frac{N}{m^2}$ (psi)	$\frac{GJ_R}{C_R D_R^2}$ $\frac{N}{m^2}$ (psi)	% MIN GAUGE	% FULLY STRESSED
7075-T6	1	97.3 (214.4)	0.0837	3.37X10^6 (489)	59.6 (0.00215)	2.34X10^8 (34,000)	7.93X10^7 (11,500)	40	60
	2	87.2 (192.2)	0.0831	3.37X10^6 (489)	55.1 (0.00199)	2.34X10^8 (34,000)	7.34X10^7 (10,700)	40	60
MED-C	1	80.0 (176.3)	0.0738	3.37X10^6 (489)	45.0 (0.00162)	2.32X10^8 (33,700)	6.93X10^6 (1010)	60	0
	2	58.1 (128.1)	0.0795	3.37X10^6 (489)	31.1 (0.00112)	2.15X10^8 (31,200)	5.93X10^6 (861)	60	20
E-GL	1	96.2 (212.0)	0.194	3.37X10^6 (489)	55.4 (0.00200)	9.52X10^7 (13,800)	8.62X10^6 (1250)	60	40
	2	76.6 (168.8)	0.195	3.37X10^6 (489)	41.7 (0.00151)	9.52X10^7 (13,800)	7.79X10^6 (1130)	60	40

Table D-2. Calculated Characteristics of 7075-T6 Aluminum-Alloy, and 0° E-Glass and MED-C Wings for a 2000 kg Aircraft.

$\cancel{R}$ = 7.5

R = 0.75

THICKNESS = 0.14

LOADFACTOR = 5.0

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MATERIAL	NO. OF CELLS	WING WEIGHT, kg (lb)	TIP DEFL / SEMI-SPAN	$\dfrac{M_R}{C_R D_R^2}$ $\dfrac{N}{m^2}$ (psi)	$\dfrac{W_R}{(SPAN)C_R D_R}$ $\dfrac{kg}{m^3}$ (pci)	$\dfrac{EI_R}{C_R D_R^3}$ $\dfrac{N}{m^2}$ (psi)	$\dfrac{GJ_R}{C_R^2 D_R^2}$ $\dfrac{N}{m^2}$ (psi)	% MIN GAUGE	% FULLY STRESSED
7075-T6	1	142.8 (314.7)	0.0869	6.06×10^6 (880)	98.0 (0.00354)	4.22×10^8 (61,200)	1.43×10^8 (20,800)	40	60
	2	133.3 (293.8)	0.0865	6.06×10^6 (880)	94.2 (0.00340)	4.22×10^8 (61,200)	1.32×10^8 (19,200)	40	60
MED-C	1	99.2 (218.6)	0.0873	6.06×10^6 (880)	57.1 (0.00206)	3.86×10^8 (56,000)	1.18×10^7 (1730)	40	20
	2	73.6 (162.2)	0.0902	6.06×10^6 (880)	42.3 (0.00153)	3.86×10^8 (56,000)	1.07×10^7 (1550)	40	60
E-GL	1	126.4 (278.6)	0.209	6.06×10^6 (880)	77.9 (0.00281)	1.71×10^8 (24,800)	1.55×10^7 (2250)	40	60
	2	100.9 (222.4)	0.209	6.06×10^6 (880)	62.9 (0.00227)	1.71×10^8 (24,800)	1.40×10^7 (2030)	40	60

Table D-3. Calculated Characteristics of 7075-T6 Aluminum-Alloy, and 0° E-Glass and MED-C Wings for a 4000 kg Aircraft.

$\mathcal{R} = 7.5$

$\mathcal{T}R = 0.75$

THICKNESS = 0.14

LOADFACTOR = 4.5

MATERIAL	NO. OF CELLS	WING WEIGHT, kg (lb)	TIP DEFL / SEMI-SPAN	$\dfrac{M_R}{C_R D_R^2}$ $\frac{N}{m^2}$ (psi)	$\dfrac{W_R}{(\text{SPAN})C_R D_R}$ $\frac{kg}{m^3}$ (pci)	$\dfrac{EI_R}{C_R D_R^3}$ $\frac{N}{m^2}$ (psi)	$\dfrac{GJ_R}{C_R^2 D_R^2}$ $\frac{N}{m^2}$ (psi)	% MIN GAUGE	% FULLY STRESSED
7075-T6	1	196.2 (432.4)	0.0740	4.28X10^6 (621)	71.8 (0.00259)	2.98X10^8 (43,300)	1.09X10^8 (15,800)	20	80
	2	184.2 (406.0)	0.0739	4.28X10^6 (621)	68.8 (0.00248)	2.98X10^8 (43,300)	1.01X10^8 (14,700)	20	80
MED-C	1	141.2 (311.2)	0.0734	4.28X10^6 (621)	46.3 (0.00167)	2.78X10^8 (40,300)	9.14X10^6 (1330)	40	0
	2	101.0 (222.6)	0.0776	4.28X10^6 (621)	35.6 (0.00219)	2.73X10^8 (39,600)	8.11X10^6 (1180)	40	60
E-GL	1	176.2 (388.3)	0.181	4.28X10^6 (621)	59.9 (0.00216)	1.21X10^8 (17,600)	1.18X10^7 (1710)	20	60
	2	138.6 (305.5)	0.180	4.28X10^6 (621)	47.8 (0.00173)	1.21X10^8 (17,600)	1.07X10^7 (1550)	40	60

Table D-4. Calculated Characteristics of 7075-T6 Aluminum-Alloy, and 0° E-Glass and MED-C Wings for an 8000 kg Aircraft.

$\mathbb{R}$ = 8.0

$\mathbb{R}$ = 0.5

THICKNESS = 16%

LOADFACTOR = 3.8

AIRCRAFT WEIGHT, kg (lb)	NO. OF CELLS	WING WEIGHT, kg (lb)	$\dfrac{\text{TIP DEFL}}{\text{SEMI-SPAN}}$	$\dfrac{M_R}{C_R D_R^2}$ $\dfrac{N}{m^2}$ (psi)	$\dfrac{W_R}{(\text{SPAN})C_R D_R}$ $\dfrac{kg}{m^3}$ (pci)	$\dfrac{EI_R}{C_R D_R^3}$ $\dfrac{N}{m^2}$ (psi)	$\dfrac{GJ_R}{C_R D_R^2}$ $\dfrac{N}{m^2}$ (psi)	% MIN GAUGE	% FULLY STRESSED
1000	1	49.34 (108.7)	0.0960	5.25×10^6 (762)	66.71 (0.00241)	3.82×10^8 (55,400)	1.33×10^8 (19,300)	40	60
	2	41.73 (31.9)	0.0954	5.25×10^6 (762)	60.21 (0.00217)	3.82×10^9 (55,400)	1.23×10^8 (17,900)	40	60
2000	1	74.8 (164.9)	0.0819	3.37×10^6 (489)	46.34 (0.00167)	2.45×10^8 (35,600)	9.41×10^8 (137,000)	20	60
	2	64.06 (141.3)	0.0817	3.37×10^6 (489)	41.09 (0.00148)	2.45×10^8 (35,600)	8.71×10^8 (126,000)	40	60
4000	1	107.5 (236.9)	0.0836	6.06×10^6 (880)	73.40 (0.00265)	4.41×10^8 (64,000)	1.69×10^8 (24,500)	20	80
	2	97.55 (215.1)	0.0835	6.06×10^6 (880)	68.60 (0.00248)	4.41×10^8 (64,000)	1.57×10^8 (22,800)	20	80
8000	1	150.8 (332.4)	0.0713	4.28×10^6 (620)	54.45 (0.00197)	3.11×10^8 (45,100)	1.29×10^8 (18,700)	20	80
	2	135.9 (299.5)	0.0712	4.28×10^6 (620)	50.49 (0.00182)	3.11×10^8 (45,100)	1.20×10^8 (17,400)	20	80

Table D-5. Calculated Characteristics of $0^\circ/\pm 45^\circ$ MED-C Wings for Four Generic General Aviation Aircraft.

AIRCRAFT WEIGHT, kg (lb)	NO. OF CELLS	WING WEIGHT, kg (lb)	TIP DEFL / SEMI-SPAN	$\dfrac{M_R}{c_R D_R^2}$ $\frac{N}{m^2}$ (psi)	$\dfrac{W_R}{(SPAN)c_R D_R}$ $\frac{kg}{m^3}$ (pci)	$\dfrac{EI_R}{c_R D_R^3}$ $\frac{N}{m^2}$ (psi)	$\dfrac{GJ_R}{c_R D_R^2}$ $\frac{N}{m^2}$ (psi)	% MIN GAUGE	% FULLY STRESSED
1000	1	50.4 (111.1)	.145	5.25×10^6 (762)	60.9 (0.00220)	2.14×10^8 (31,100)	4.67×10^6 (678)	80	20
	2	38.8 (85.5)	.144	5.25×10^6 (762)	41.8 (0.00151)	2.14×10^8 (31,100)	4.43×10^6 (643)	80	20
2000	1	83.3 (183.6)	.122	3.37×10^6 (489)	47.0 (0.00170)	1.37×10^8 (19,900)	3.36×10^6 (488)	80	20
	2	58.9 (129.8)	.123	3.37×10^6 (489)	31.7 (0.00114)	1.37×10^8 (19,900)	3.10×10^6 (450)	80	20
4000	1	103.4 (227.9)	.140	6.06×10^6 (880)	59.9 (0.00216)	2.47×10^8 (35,800)	6.04×10^6 (877)	40	40
	2	75.0 (165.3)	.140	6.06×10^6 (880)	43.3 (0.00156)	2.47×10^8 (35,800)	5.58×10^6 (810)	40	60
8000	1	146.4 (322.7)	.120	4.28×10^6 (621)	47.9 (0.00173)	1.75×10^8 (25,400)	4.65×10^6 (675)	40	40
	2	102.8 (226.6)	.120	4.28×10^6 (621)	34.3 (0.00124)	1.75×10^8 (25,400)	4.23×10^6 (614)	40	60

Table D-6.   Calculated Characteristics of Hybrid, $0^\circ/\pm 45^\circ$ MED-C Compression Cover, 0° Kevlar Tension Cover Wings for Four Generic General Aviation Aircraft.
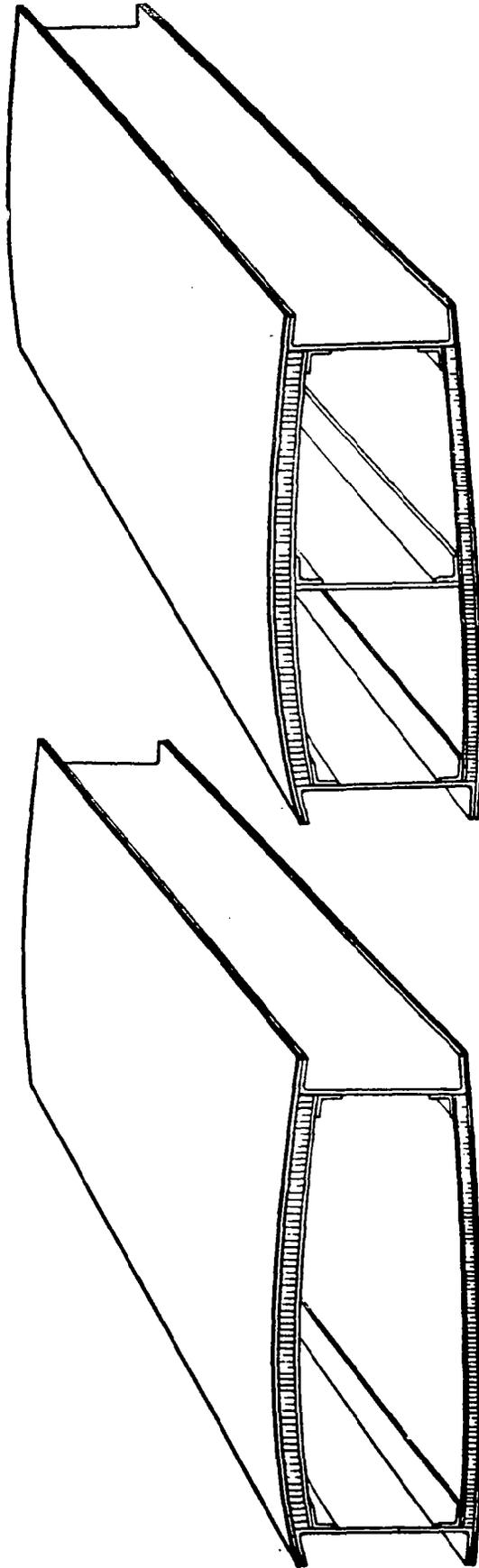
Figure D-1. Idealized One – and Two–Cell Box Beams Used as Model for BOZO Code Analysis.
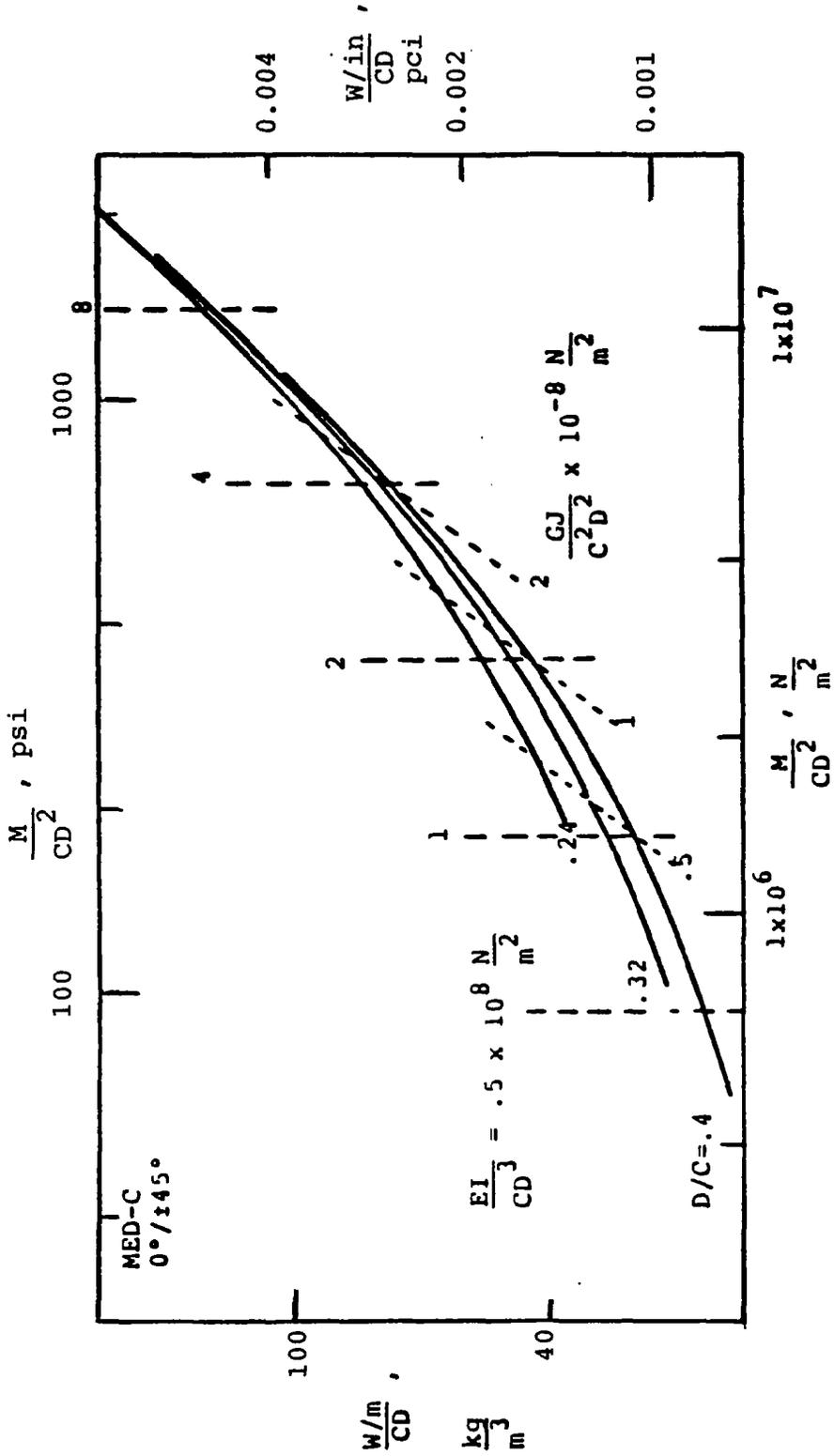
Figure D-2. Generalized Presentation of Results of Weight/Efficiency Analysis, Showing Crossing Curves for Bending Stiffnesses EI and Twisting Stiffnesses GJ.
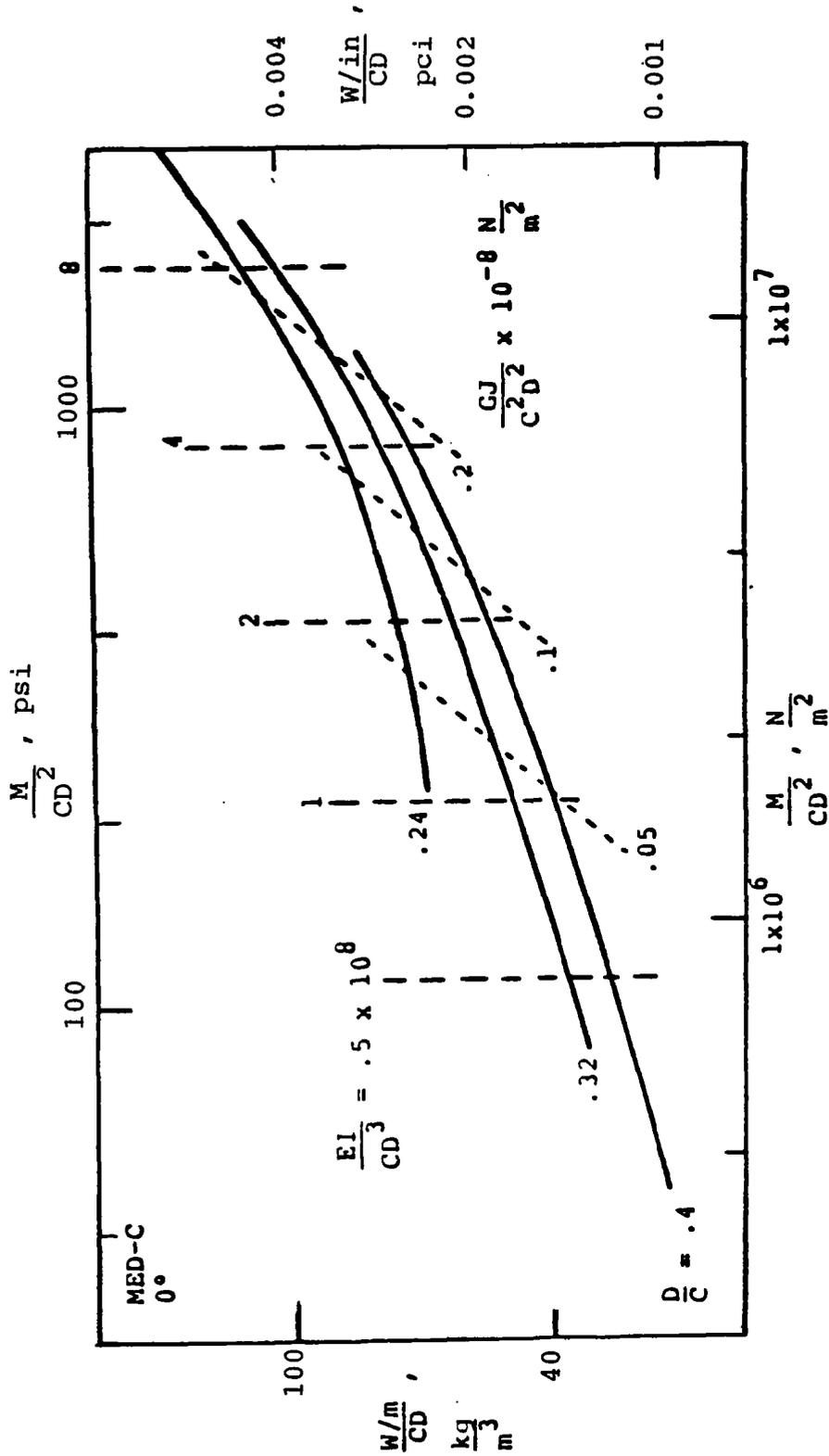
Figure D-3. Generalized Presentation of Results of Weight/Efficiency Analysis of Box Beams in Bending Showing Effect on Twisting Stiffness GJ of Added ±45° Reinforcements Compared to 0° in Figure D-2.

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APPENDIX E - MANUFACTURING TECHNOLOGY

This appendix presents background material, as well as recommendations in the area of manufacturing technology. Several outside consultants were used to assist in the preparation of this material. The services of Mr. M. L. Salvador, Composite Consultant, Mr. G. Lubin, Consultant, and Mr. R. Kollmansberger, of Composites Technology Consulting, are acknowledged and appreciated. The helpful discussions with Dr. B. Jones, of Compositek Engineering are also appreciated.

Manufacturing Development Programs #1 - #4 derive from the studies described in this appendix.

BOX BEAM MANUFACTURING ANALYSIS

In order to provide a framework for the evaluation of the operations required in the manufacture of composite structures, a typical component was defined, and costs for its manufacture developed. The component selected was a simple box beam embodying three types of construction, as shown in figure E-1. Costs of manufacture of this component in the three types of materials represented by the candidate composites of these studies were then developed as a summation of the costs of the following, detailed operations.

Manufacturing Operations

The first step in the analysis was the definition of the manufacturing operations required, as follows.

Core Material

- Purchase core in thickness required.

- Stabilize core stock prior to detail fabrication -
  oven cure @250° X 1 hr.
- Bandsaw core to detail shape.
- Pot core for fastener installation. Oven cure @250°
  X 1.5 hrs. (mechanical assembly method only).

Layup Operations

- Cut materials to shape using aluminum cutting templates
  and stanley knife.
    - A materials nesting program is critical to effective materials usage. Typical materials efficiency ranges between 50% and 60%. This can be increased to approximately 75% with an effective nesting program and an automated cutting system.
- Locate plies onto mold form using locating templates
  (metal or mylar as applicable).
- Compact (debulk) plies. Vacuum compaction is required
  for graphite materials. Roller compaction is generally
  acceptable for fiber glass and Kevlar.

For Skin Sandwich

- Locate nomex core onto layup using locating templates
  and compact into place.
- Complete layup and compact as above.

For Skin Laminate

- Cut and apply film adhesive for spars, stringers and/or
  ribs as applicable.
- Position spars, stringers and/or ribs as applicable.
- Rough bag to compact parts into place.

All Layups

- Install thermocouples to monitor cure temperature.
- Apply bleeder/breather materials.

Cure Operations

- Apply nylon bagging material with prestite tape. A reusable form fitted rubber bag may be cost-effective for volume production.
- Apply vacuum and check leak rate.
- Load autoclave and plug into autoclave vacuum source. Check leak rate.
- Apply temperature and pressure, vent vacuum at 20psi.
- Hold at 350° F as required by specification.
- Reduce temperature.
- Unload autoclave, strip vacuum bag and remove part.
- Clean tool and apply release agent for next use.
     For the purpose of estimates, it is assumed that one autoclave cure cycle yields one shipset of parts, for example:
  - 350°F X 45psi yields sandwich skins required to produce one box beam assembly.
  - 350°F X 80psi yields solid laminated parts to produce one box beam assembly.

Trim Operations

- Load tool.
- Trim periphery with hand router.
- Dress up routed edge (hand sand with 180 grit paper).

Assembly Operations - Secondary Bond Method

- Load lower skin assembly into tool.

- Cut and apply film adhesive as required.
- Apply upper skin assembly.
- Install thermocouples to monitor cure temperature.
- Clamp and/or vacuum bag as applicable.
- Load oven.
- Apply temperature to 350°F as required by specification and hold.
- Reduce temperature as required by specification.
- Remove tool from oven, remove clamps and/or strip vacuum bag and remove assembly.
- Clean-up assembly for next operation.
- Clean tool for next use.

Assembly Operations-Mechanical Assembly

- Load tool with upper and lower skin assemblies.
- Drill and countersink (spacematic) for .187 DIA. Hilok fasteners.
- Install Hilok fasteners.
- Drill and countersink (spacematic) for blind fasteners.
- Install blind fasteners.
- Remove assembly from tool.

Finishing Operations

- Scuff sand surface to be finished (180 grit paper), and solvent wipe.
- Apply static conditioner. Room temperature cure and sand smooth.
- Apply pinhole filler and/or surfacer as required. Room temperature cure.
- Sand smooth (180 grit paper).
- Apply primer - oven cure per specification.
- Scuff sand and apply finish coat. Oven cure per specification.

Manufacturing Cost Analyses

First up-to-date raw material costs were obtained by M. Salvador These are given in table E-1.

The manufacturing costs were developed for the various steps given above for each of the three configurations and materials. Results are given in tables E-2 to E-4. These values were arrived at by the use of the procedures and values given in tables E-5 to E-7 for carbon composites. Tables E-5 to E-7 provide a procedure for costing the layup effort required to produce carbon reinforced skins, spars, and stringers.

The basic layup cost addresses the cost to prepare tools, setup, apply breather/bleeder materials, and rough bag. The A and B values address the cost to cut, apply, and compact materials. To obtain the costs for Kevlar and E-Glass apply the following factors.

STRINGER

        Kevlar   - Use 73.6% of A Value
        Fiberglass   - Use 60.4% of A Value

SPAR

        Kevlar   - Use of 77.7% of A & B Values
        Fiberglass   - Use 72.0% of A & B Values

SKIN

Use tables E-5 to E-7 "as is" for graphite, Kevlar and fiberglass construction.

Results for Cost Analyses

As expected, the single cell, sandwich type box beam required

the least number of man-hours to manufacture in all three materials. Among the materials, the E-Glass was the most economical both in raw material cost and fabrication costs. The Kevlar required the greatest number of man hours for fabrication, primarily due to costs of cutting.

THERMOPLASTICS FOR EASE OF MANUFACTURE

The use of thermoplastic laminates is generally considered to offer aircraft designers materials which are comparable in strength to thermosetting laminates, but much more economical to fabricate into simple forms. The main advantages, besides labor cost are: (1) ability to use thicker reinforcements, (2) eliminating interlaminar failures, (3) use of inexpensive tools, and (4) better part uniformity.

Parts recommended for use of thermoplastics in aircraft include: beams, ribs and stringers for wing and tail surfaces and simple curvature skins. Use of current thermoplastics in parts in contact with aircraft fuel and hydraulic fluids is not recommended.

Resins for Thermoplastics

Depending upon the use temperature, there are two types of thermoplastic resins suitable for use in laminates. For lower temperatures in the range from 150 - 180°F, acrylic resins are the most promising. Their main advantage is their ability to impregnate the fibers completely when a monomer/polymer mixture is used. This mixture is prepared by dissolving ground polymer powder in a liquid monomer from which the inhibitor has been removed and benzoyl peroxide catalyst has been added. This forms a syrupy mixture which is used to impregnate the reinforcements using a knife blade arrangement or a simple rolling mill. The impregnated reinforcements are usually allowed to stand for a day to cure the resin partially to a semi-solid stage

which is then cured in a continuous or stationary press. High
strength and stiffness properties are realized because of com-
plete wetting of the fibers. Typical data for graphite/acrylic
181 style fabrics are as follows:

	σ_{tu}, $\frac{MN}{m^2}$ (ksi)	E_x, $\frac{GN}{m^2}$ (msi)
Room temperature	440 (64)	52 (7.5)
77°C (170°F)	280 (40)	38 (5.5)

For higher temperatures, several resins including phenoxy,
polycarbonate and polyether sulfone (PES) have been used. PES
has better properties than the others and has been used more
extensively. It can withstand temperatures of up to the 330 –
350°F range. The average tensile strength at room temperature is
360 $\frac{MN}{m^2}$ (52 ksi) with a modulus of 48 $\frac{GN}{m^2}$ (7 msi). The preparation
of polyether sulfone laminates is more complicated and
costly than for acrylics. The resin is first dissolved in a sol-
vent and is then used to impregnate the dry graphite reinforce-
ments. The next step is to remove the solvent and laminate
the reinforcements between two plies of PES film. The mater-
ial cost of this thermoplastic laminate is about 10 times that
of acrylic. In all cases, the properties of thermoplastic lam-
inates depend upon the type and thickness of the reinforcements
used.

Reinforcements for Thermoplastics

E-Glass, S-Glass, Kevlar, and graphite can all be used in
thermosetting laminates. All of these fibers require the ident-
ical impregnating operations and can be used individually or in
hybrid form. While unidirectional cloths can be handled, it has
been found that woven fabrics are preferable since they are not
subject to yarn separation during the forming process.

Thermoforming

The forming operations for both high and low temperature materials are essentially similar. Except for acrylics, the forming temperature range is much greater than for PES, allowing more time between heating and forming operations without decreasing the quality of the laminate. The PES plastic range is so narrow that the heating time must be precisely controlled to obtain satisfactory parts.

Thermoforming can be performed by three different methods.

Continuous Forming

Continuous forming is used for large scale production and the thermoplastic is supplied in the form of a continuous tape. This tape is fed from a spool into the forming machine which consists of one or more heating and forming stages, and the final shape is extruded continuously and cut into desired lengths. At present, such machines must be designed for each specific shape since there is no possibility of using interchangeable dies for this operation.

Matched Die Forming

For short lengths of any contour, matched die forming can be used. The sheet is usually preheated in an oven, placed into a press and formed between two matched dies, and cooled. Good reproducibility is obtained and the mold cost is reasonable.

Vacuum Forming

For large shapes with little curvature, vacuum forming can be used. Mold and processing costs are the lowest but reproduci-

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bility between parts is not as good as for matched tools.

Part Trimming

The machining of the formed parts is similar but simpler than for molded parts - there is no resin flash and in many cases, only simple cutting or sawing operations are required.

Surface Protection

All current thermoplastic parts are soluble in wash solvents or other organic liquids used in aircraft and must be protected with an impervious coating Polyurethane and epoxy paints, specially formulated solvents must be used. Any damage to the paint layer must be repaired as soon as detected.

Implications for General Aviation Aircraft Applications

Clearly thermoplastics offer the potentials for cost-effecness desirable for general aviation aircraft applications. Areas for research and development are also clear: (1) the need for resin formulations not requiring surface protection from solvents, and (2) the adequate demonstration that strength performance competitive with epoxy is achievable.

MANUFACTURING TECHNOLOGY DEVELOPMENT

This study considers both short and long term programs directed toward the goal of establishing composite manufacturing costs (in manhours per kilogram of structure) below those possible with metal structure. The short term efforts are directed at resolving specific problems currently facing the general aviation industry. The long term efforts should develop manufacturing technologies that will establish and maintain U.S.A. gener-

al aviation industry supremacy in the marketplace. New and
innovative techniques that change the state of the art must
be developed, - especially those related to automation. Tech-
nology gains are needed in both lower cost operations and in-
creased confidence/reliability in the manufactured product.
The latter would manifest itself in such ways as reduced quali-
ty control cost based on proven manufacturing reliability and
less Material Review Board (MRB) activity that raises indirect
costs and delays production part flow through the shop.

General Aviation Manufacturing Technology Requirements

The composite structures manufacturing technology require-
ments are perhaps best put into perspective by comparisons with
the military and transport industries, inasmuch as most of the
manufacturing technology developments to date have been generated
in those industries. Many of the parameters governing general
aviation are different. Production rates are generally higher
than either military or transport aircraft plants as evidenced
by a major general aviation factory producing over 6,000 aircraft
of all types in one year. Total production in the U.S.A has
reached over 13,000 general aviation aircraft per year while
total military aircraft production is usually less than 1,000
per year and transport category aircraft companies range from
300 to 600 aircraft per year. The advantage of general aviation's
higher production rates is that it makes automation and unique
tooling/fabrication concepts pay off better than if used by
either military or transport category aircraft.

Composites Manufacturing High Cost Centers

Manufacturing technology plans should be evolved from the
entire manufacturing operation. This includes reviewing direct
fabrication tasks plus the support operations such as quality

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control, tooling, process planning, and computer system inter-
action with the entire system. Direct fabrication tasks include
raw material acceptance, material cutting and storage, part layup
and cure, trimming and machining, subassembly, bonding, systems
installation, and final assembly. Analyzing the direct and sup-
port operations, the following high cost centers or choke points
are identified. A choke point is a portion of the operation that
tends to backlog parts at a critical portion of the manufacturing
flow whether or not it is a high cost operation in itself.

a. Tooling - This includes tool surface preparation on
    a routine basis, repair of leaky tools, and the overall
    coordination especially during program start-up.

b. Hand Layup and Bagging - A very labor intensive oper-
    ation that requires automation to reduce cycle time
    and cost. Tape laying machines and robots are already
    being used in this operation. However, the bagging
    and unbagging operation is still time consuming.
    Where a tape laying machine or robot cannot do the
    layup work because of contour an automated layup
    assist system is required.

c. Autoclave Cure Time - Autoclave cycles normally require
    six (6) to ten (10) hours depending on tool and part
    load, autoclave efficiency, and loading scheme. More
    rapid cure prepregs that are not exotherm restrained
    should be developed or methods of rapidly curing
    current composite and adhesive prepregs are needed.

d. Non Destructive Inspection (NDI) - Ultrasonics A-Scan
    or C-Scan is the industry standard today. Some radio-
    graphy is also used. NDI methods are under develop-
    ment that are a magnitude of time faster than ultra-
    sonics. Typical of these new methods are thermography
    and acoustographics. Their development must be pushed.
    In addition, ultrasonic C-Scan NDI equipment is very
    expensive at $250,000 to $1,300,000 for single units

depending on capability desired. C-Scan units can scan at 5 to 60 ft.^{2}/hour, but at general aviation production rates of over 2,000 parts per day this would require an enormous NDI facility.

e.   Surface Preparation (Bonding and Painting) - Manual methods of surface preparation are very time consuming. Peel ply alone does not provide as much bond strength as does a grit blasted or sanded surface. In addition, peel ply provides a "rough" appearing external surface if used as a surface preparation prior to painting. Improvement could be an improved peel ply or a portable energy source (i.e., laser) that would deoxidize the surface with no damaging penetration of the laminate.

f.   Assembly Bonding - Current assembly bonding methods are time consuming using film adhesive. Paste type adhesives do not have a sufficient data base for structural consideration. Film adhesives require elaborate in-situ bond tools or removal to oven or autoclave, and in addition have out time problems. A rapid curing technique performed on the assembly tool with minimal tooling complexity is required. Expanding the data base of paste adhesives would also be desirable.

g.   Redundant Processing - This is more insidious a cost factor than the other items because the rerouting of parts for additional processing ties up tools, requires additional NDI, expands the overhead cost base, and is less efficient. The basic solution is more cocure of assemblies, but this requires more confidence in sophisticated tooling techniques. This is particularly true in fuselage fabrication and assembly operations.

h.   Engineering Tolerances - Engineers generally have the attitude of "let's make it the best we can" just to

be on the safe side. This sometimes does not consider
what the production shop can most efficiently produce
and results in many rejections that, simply put, "tie
the shop in knots". Engineering tolerances on drawings
and specifications must be reviewed by operations per-
sonnel to assure their producibility in the production
environment.

i.   Material Review Board (MRB) - This high cost center
      is the result of many other items but should be
      mentioned as a potential choke point. It is slow
      problem resolution that results in much of the delay
      and backlog of work. While parts await MRB decisions,
      substitute parts must be made because the production
      line must keep moving.

j.   Paperwork System - How mundane! However, the normal
      shop paperwork system creates as much of a problem
      as the actual parts. Paperwork coordination between
      engineering requirements, process planning, and qual-
      ity control functions is usually difficult. A CADCAM
      system should be established to avert the mountain
      of paperwork and provide better coordination of re-
      quirements.

Manufacturing Technology Programs

The following manufacturing technology programs have been
selected in part on the basis of the foregoing "choke point"
analysis, in part on the basis of findings of other studies
herein, and in part to avoid conflict with, but rather to supple-
ment, ongoing DOD programs, such as: Northrup contract, "Manu-
facturing Technology for Composites Assembly; Grumman, "Inte-
grated Laminating Center Development"; General Dynamics/Fort
Worth, "Tape Laying Machine Development." Most of these pro-
grams direct effort at prepreg layup, whereas here we shall

be more concerned with the entire process.

Thermoplastic Matrix Prepreg Fabrication

The fabrication capabilities and limitations for the newer
generation thermoplastic matrix prepregs, such as polyether
ether ketone (PEEK) must be defined. These thermoplastics offer
advantages of toughness, moisture resistance, repairability,
solvent resistance, good mechanical properties, and quickness
of consolidation/forming operations. Their limitations are
primarily in the processing area where the boardy like prepreg
must be consolidated at 650° - 750°F with pressures of 100 -
300 psi for only a few minutes. Forming operations are performed
in the same temperature range which correlates better with metal
forming operations than epoxy matrix composite parts.

Research is needed to define the processing parameters
for thermoplastic composites relative to the equipment required,
tooling concepts, automation possibilities for prepreg placement,
and time-temperature-pressure ranges. This research should con-
centrate on aircraft parts that are prime candidates for thermo-
plastic matrix composites such as empennage structure, control
surfaces, landing gear doors, stringers, and fuselage skins.

An interesting concept is the use of an automated tape laying
machine for part consolidation to contour directly on the tool.
This would required some type of heated and pressurized roller
for tape application and, if necessary, a preheating system
to soften the thermoplastic matrix prior to reaching the tape
consolidating head. The additional advantage of the tape lay-
ing machine is that the unidirectional tape material is the
easiest material form to produce. Another concept to investi-
gate is hot forming of preheated prepreg in "cold" tools using
a bladder type molding operation against a solid tool surface.

Other operations needing evaluation for thermoplastics are
machining, fastener installation, adhesive bonding with surface

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preparation, NDI techniques, and other assembly operations.
These should be done after the basic fabricating parameters
have been evaluated under a variety of shop conditions with
shop equipment. Another avenue that should be investigated
is how to use the industry's existing investment in autoclaves
for processing thermoplastic composites. This could be done
using the autoclave as a pressure vessel and supplementary heat-
ing source in addition to an integrally heated tool.

Rapid Processing Resins

The standard cure cycles for both prepregs and adhesives
(epoxy based) are long; consuming as much as 8 - 10 hours for
a complete cycle depending on part and tool mass. These times,
although not labor intensive, do take the part out of the normal
production flow for excessive periods of time, - plus autoclave
operation itself is costly. Methods of rapidly curing the re-
sins, possibly in as little as 10 - 30 minutes, would be a tre-
mendous advantage in keeping the normal autoclave cure opera-
tions from becoming a choke point.

Two methods should be investigated for potential solution
of this problem. First, is the formulation of rapidly curing
resin systems that are suitable for prepregs and resist exotherm-
ing as epoxies do. Perhaps acrylate resin based formulations
would be a suitable starting point. These should be resins
that would require only 15 to 30 minutes at temperature (pre-
ferrably lower than 350°F to preserve tool life) to complete the
cure. Any longer than that and the R & D expenditure would
hardly be worth the effort. A second approach would be to find
more rapid curing methods such as microwave, Xenon flash, or
similar high energy inputs which could be used to cure prepregs
in several minutes rather than the one to two hours required for
standard prepregs in the oven/autoclave. This type of rapid cure
would be particularly useful in subassembly and assembly bonding

operations wherein moving the assembly and its tool to the oven or autoclave is very cumbersome and fraught with the danger of misaligning some of the details.

Filament Winding

Filament wound structure is suitable for general aviation aircraft because of size of assemblies and the relative simplicity of structural configurations. Military aircraft are generally a more complex design with buried engines, non-circular fuselages, large numbers of access panels, and higher structural loadings resulting in more complex designs less suitable for filament winding. Helicopters are the only other aircraft suitable for a major filament winding application. Helicopter tailcones have already been filament wound successfully over the past 5 to 7 years.

Filament winding techniques recently developed offer more cost and manufacturing advantages than previous filament winding methods. These techniques basically involve winding a structure on an inflatable, reinforced mandrel that is later used to exert cure pressure against the structural laminate as it is forced against a split shell OML hard tool that controls the exterior surface. The inflatable internal mandrel contains depressions for skin reinforcements such as Z-Sections. Thus, one process cycle would wind and cure for example, a completely reinforced fuselage skin. Skin ply buildups could also be accommodated. Cutouts would be added later per templates.

The economics of filament winding are such that a fuselage as described above should be 35% to 40% of the cost of a conventional composite fuselage. The cost of the inflatable internal mandrel tool is higher than standard tools, but the recurring cost savings far outweigh the high tooling cost.

Filament winding should be investigated for empennage and control surfaces structures wherein the substructure and skins

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are cocured at the same time. This is particularly applicable to truss-web substructure which would be wound on mandrels, and then the outer skins would be over-wound on the substructure mandrels and windings. External mold line control is established by a clam-shell type of tool.

Filament winding has the advantage of going directly from the raw fiber and resin directly to the final part/assembly form. There are no intermediate steps of (1) making the prepreg, (2) storing and dispensing the prepreg, (3) cutting prepreg plies and dispensing them, and (4) layup of the prepreg on the tool. Additional economies are generated by the composite material laydown rate using an automated, programmable machine.

Braiding Machine Development

Throughout these studies, the merits of woven-fabric rein-forcements have been cited (recurrently) both for performance and for ease of manufacture. Weaving of advanced filaments such as high modulus carbon filaments, however, into intricate configurations presents problems, both of tightness of weave and of possible damage to the filaments.

A fabric forming process which through the years has not been fully exploited offers promise of alleviating those pro-blems. This is the braiding approach instead of the convention-al shuttle-loom weaving approach. Braiding has been used almost exclusively for small products because large products require very large machines.

Apart from size of machine required, braiding has many ad-vantages: (1) it is capable of a wide variety of braid config-urations, including triaxial (in-plane) and three dimensional; (2) it handles the yarns gently, with a minimum of damage; (3) it is rapid and economical. The development of large braiding machinery for advanced composites appears justified despite their size.

Automated Ply Location System

Locating external plies on a layup mold presents no problem at all since the lines are marked on the tool. However, internal ply buildups in a part like a wing skin can take the bulk of the layup time. A robot is not the answer at this time because of the ply size, and a tape laying machine of course, is of no use for woven fabric plies and is inefficient for short length tape plies. What is envisioned is an overhead projection system using light beams or laser beams to outline the ply for the fabrication operator. Thus, the precut ply from the Gerber cutting machine could be easily located by hand. The sequenced plies would be programmed into the projection system controller perhaps directly from the CAD system.

The projection system would consist of single or multiple projection heads depending on a part's size – empennage structure skins would probably need one system while a wing skin or fuselage skin would need multiple systems connected to a single controller. The line projected on the tool should be visible under normal clean room conditions and must be keyed to a 0,0,0 location point on the layup mold itself.

This type of system would greatly increase the efficiency of manual layup since the greatest time consumer is establishing internal ply locations after the first ply covers up all tool lines.

Improved Tooling for Composites

This is one of the more mundane areas of manufacturing technology and very little effort has been given to it. There are no major R & D programs in work to help resolve tooling problems associated with tool cost, tool cycle life, dimensional stability, and a host of other problems. Since most of the past R & D programs have fabricated relatively flat wing, empennage or

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control surface parts the tooling problems have not gotten much attention. However, with the advent of compound curvature, and with fuselage skins and internal structure requiring tight assembly tolerances, more emphasis will be placed on low expansion, high quality tools. These requirements might be best met by reliable, long life composite tools. Principal advantages of composite (graphite/resin) tools is their near zero (0) expansion coefficient, ability to match contour, low cost, and adaptability to large size structure. Their disadvantages are basically an unpredictable life, tendency to leak, need to be recoated for release after very few cure cycles, and general durability problems. The problem is that the epoxy resins currently used for tools are operating on every autoclave cycle at the maximum temperature exposure permitted. Thus, minor porosity and cracks can grow and cause ultimate failure of the tool.

Manufacturing technology research is required to determine if better high temperature resin matrix tools can be satisfactorily demonstrated. Candidate resins should include some of the newer, easier to process polyimides. Thermal durability, dimensional stability, repair techniques, fabrication cost, and leak-free operation must be determined. The method of fabrication could be wet layup or prepreg autoclave sure. Research should be performed with fuselage skin and substructure tools so that contour challenging tools are evaluated.

Improved NDI Methods

At present, quality control costs for composite structure fabrication are higher than they are for metallic structure. For composites, quality control costs are running 15% to 25% of direct labor costs which are about double the 8% - 10% typical of general aviation metallic structure. This ratio changes based on whether primary or secondary structure is being considered, but the above values are typical of an entire aircraft.

Efforts must be taken to lower this high quality control cost, part of which is generated by NDI requirements and techniques.

Current ultrasonic C-Scan NDI techniques are partially satisfactory but are very time consuming. They do well in analyzing relatively flat surfaces such as wing skins but have more difficulty with channel type shapes (i.e., wing spars and ribs) where the intersection of various surface planes presents a problem (web-cap intersection). These intersections are examined by angle beam ultrasonics or radiography to determine their quality. This of course requires additional set up time as the part is moved from one fixture to the next. This intersection area is one of the more defect prone areas of the structure yet is one of the most difficult to inspect. This area should be evaluated with the same equipment that does the relatively flat surfaces such as the cap or web area itself. The other alternative is to use one automated set up for the entire examination. Two separate set ups per part would raise NDI costs far above their expected levels.

There are several new NDI techniques which show promise of reducing the quality control cost substantially. There are thermography and acoustographics which are techniques designed to give near instantaneous answers on part quality at a possibly lower initial investment. The standard ultrasonic C-Scan system cost and operational rate are as follows:

C-Scan Type	initial cost ($)	operational speed (incl. set up time) m^2/hr. (ft.^{2}/hr.)
Standard Single Bridge Single Transducer Type	$250-350,000	0.4-0.5 (4 - 6)
5-Axis, Multiple Transducer System	$1,250,000	4.5-6.3 (50 - 70)

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The newer methods have been demonstrated on a laboratory scale and are in need of scale-up demonstration and evaluation. The laboratory demonstrations have been somewhat, although not 100%, successful. Thermography has shown an ability to evaluate a 0.2 m^2 (2.25 ft.^{2}) flat, constant thickness composite panel in <u>10 seconds</u>, but does required operator training and technique development. A thick, stepped laminate required up to 20 seconds for evaluation of 0.28 m^2 (3 ft.^{2}). These speeds translate to 47 - 74 m^2 hr. (500 - 800 ft.^{2}/hr.) exclusive of set up time. This is considerably faster than fixture locations required for ultrasonic C-Scan inspection. Thermography units, such as the Inframetrics 525, cost approximately $30 - 35,000. This would be a considerable savings in equipment costs over the ultrasonic C-Scan systems. The thermography systems respond to a transient heat source applied to the backside of the structure being inspected and then measuring the infrared radiation signature. Porous or void laminate areas show a lower infrared temperature profile on the infrared camera because of their insulative effect on hear transfer through the laminate.

The acoustographics system uses an equipment set up similar to ultrasonics C-Scan but the system can evaluate larger areas in a shorter period of time using an acoustographic film on the reverse side of the part from the ultrasonic transducer. The ultrasonic impulse is converted into a signal that provides an instantaneous readout on the specially developed film. Clarity of results is as good, if not better than, standard ultrasonics C-Scan printouts. This technique has been developed on a small laboratory scale at present and needs scale-up demonstration, particularly on contoured parts.

Attempts should be made to scale-up these, and other emerging NDI technologies, for comparisons to the standard ultrasonic C-Scan. Evaluation should be with real aircraft parts and assemblies that contain contour, buildups, cutouts, and cocured stiffeners. Degree of sensitivity to defects, size

of parts that can be evaluated, and relative cost of operation $(m^2/hr.$ translated to labor costs) should be defined.

Surface Preparation Techniques

Current surface preparation techniques require improvement whether it be for bonding or painting, although the biggest concern is bonding. Present methods consist of peel ply removal, hand sanding, or grit blasting. These methods all have shortcomings whether it be contamination, lower bond strength, excessive manhours, excessive moving of assembly details, or a human controlled operation. To top it all off, some manufacturers then apply a water break test to be sure of surface cleanliness.

Peel ply prepared surfaces generally show a 10% - 20% reduction in adhesive bond strength over a grit blasted or sanded surface. This has been demonstrated at practially every aerospace manufacturer. In addition, peel ply removal is not as easy as it is thought to be. Many times it tears, especially if it is in a resin rich part of the surface. Peel ply must be treated with fluorocarbon or silicone to assist its removal and these same products can stay on the "cleaned" surface as residual contaminates. Thus, the bond strength is reduced and further surface preparation is necessary.

The grit blasting and sanding operations are human controlled, time consuming, and generators of particulate contamination. They can be tolerated for detail parts, but when assembly bonding operations are underway the operation is much more difficult. The assembly has to be removed from the assembly area possibly disrupting part alignment and raising the cost.

A system of cleaning composite parts that can be done in the assembly area without the follow-on water break test would greatly enhance producibility. Perhaps a high energy source that would disrupt the resin oxidation layer without damaging

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the basic composite structure itself could perform the task.
This would have to be a portable unit to be usable in the assembly area. Another possibility is an improved peel ply material
that does not leave contaminants behind. A release coating
system other than the currently used fluorocarbons and silicones
should be investigated.

The same system could be used to prepare external surfaces
prior to painting. The savings would be two-fold, bonding and
painting operations. The surface appearance is often critical
for general aviation aircraft.

Lower Cost Fastener Systems

Graphite/epoxy structure by its nature is galvanically detrimental to the usual fastener metals such as aluminum, cadium
plated steel, martensitie stainless steels, and other anodic
metals. This results in fasteners manufactured from titanium,
austenitic or semi-austenitic stainless steel, high nickel content alloys, and copper alloys. These metals in turn raise
the price of the fasteners to several times more than the fasteners made from aluminum, low alloy steel, etc.

A worthwhile manufacturing technology program would be to
develop fasteners or fastener coating systems that allow the
lower cost fasteners to be used with graphite/epoxy composites.
Perhaps thin, lubricative type ceramic coatings could be used
to coat aluminum and low alloy steel fasteners for galvanic
protection from graphite/epoxy. Another possibility would be
a toughened and strengthened organic coating that would also
make the fastener impervious to galvanic corrosion. The resultant savings would come from the use of lower cost fasteners
and reduced installation costs. Further exploration of the
initial work done by LTV with composite fasteners could also
be appropriate as they ended the program with glass reinforced
"tough" resin fasteners. For some areas of the aircraft where

shear stresses are relatively low, this type of fastener could prove successful.

Improved fastener hole drilling and countersinking could also reduce fabrication costs. Currently used drilling tools have very limited lives when drilling composite laminates. Most tools are either carbide or diamond faced, both of which are expensive compared to High Speed Steel (HSS) tools used with aluminum structure. An improvement in composite drilling techniques would also reduce the quality control effort required to assure structurally satisfactory laminates around the fastener holes. A typical all composite type general aviation aircraft could use $50 - $100,000 worth of the expensive fasteners (Monogram Big Foot's, Hylok, rivets, screws, etc.) compared to a far lower value for an aluminum general aviation aircraft. The total cost of fasteners depends on the design philosophy of fastened vs. bonded structure. A reduction of fastener cost is a pleasant fall-out of designing with adhesively bonded structure.

Assembly Techniques

The assembly of composite structure is one of the more labor intensive operations. Whether bonding or mechanical fastening is utilized, the cost is still very high compared to detail part and subassembly fabrication. In a typical aircraft the direct labor hours for detail part/subassembly fabrication is about equal to the systems installation/assembly operations. In composite aircraft, there are more manhours dedicated to the latter operations because of the time involved in large assembly bonding; mechanical fastening; and form/fit work with doors, windows, and system installations. A manufacturing technology program directed at general assembly production techniques working a multitude of individual tasks that could include rapid adhesive curing, improved tooling for in-situ bonding, more

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cost-effective drilling and fastening techniques, portable rapid edge trimming, improved bond surface preparation methods, and more accurate tooling for part location. The last item is important because composite structure can not be deformed or bent to match the mating structure as aluminum can be. In addition, on assembly repair methods need to be improved wherein full structural integrity can be re-established without having to remove the structure from the assembly jig.

Table E-1. Raw Material Costs

Material	Cost
Honeycomb Core – 3.18 mm (0.125 in.) cell x 48 $\frac{kg}{m^3}$ (3 $\frac{lbs}{ft^3}$) x 19mm (0.75 in.) thickness	$3200/m^3 ($90/ft^3)
T-300/Epoxy Prepreg broadgoods, – style 3K-70-PW woven fabric	$120/kg ($55/lb)
– tape, grade 190 unidirectional	$122/kg ($56/lb)
Kevlar/Epoxy prepreg, – type 285 aramid fabric	$54.5/kg (25/lb)
E-Glass/Epoxy prepreg, – type 181 glass cloth	$11/kg ($5/lb)
Adhesive – film adhesive	$46/kg ($21/lb)

Table E-2. Manufacturing Costs - Skin Stringer Construction

Operation	Part	E-Glass	Kevlar	Carbon
			Man Hours	
Lay up				
	Spar	13.4	14.1	16.8
	Ribs	27.2	32.2	39.0
	Z-Sections	10.5	12.0	15.2
	Angles	5.0	4.8	6.8
	Skin	21.9	21.9	21.9
		78.0	84.9	99.7
Cure				
	Spar	5.7	5.7	5.7
	Ribs	14.2	14.2	14.2
	Z-Sections	4.6	4.6	4.6
	Angles	4.6	4.6	4.6
	Skin	5.7	5.7	5.7
		34.8	34.8	34.8
Trim				
	Spar	2.2	2.4	2.2
	Rib	3.4	4.3	3.4
	Z-Sections	2.9	3.3	2.9
	Angles	2.9	3.3	2.9
	Skin	3.4	4.0	3.4
		14.9	17.3	14.9
Assembly		9.3	9.3	9.3
TOTAL		137.0	146.3	158.7

Table E-3. Manufacturing Costs- Multiweb Construction

Operation	Part	Man Hours		
		E-Glass	Kevlar	Carbon
Lay up				
	Spar	26.8	28.2	33.7
	Skin	<u>13.3</u>	<u>13.3</u>	<u>13.3</u>
		40.1	41.5	47.0
Cure				
	Spar	12.3	12.3	12.3
	Skin	<u>6.1</u>	<u>6.1</u>	<u>6.1</u>
		18.4	18.4	18.4
Trim				
	Spar	4.5	4.9	4.5
	Skin	<u>3.4</u>	<u>4.1</u>	<u>3.4</u>
		7.9	9.0	7.9
Assembly		<u>8.2</u>	<u>8.2</u>	<u>8.2</u>
TOTAL		74.6	77.2	81.5

Table E-4. Manufacturing Costs - Sandwich Construction

Operation	Part	Man Hours E-Glass	Kevlar	Carbon
Core		5.4	5.4	5.4
Lay up				
	Spar	13.4	14.1	16.8
	Skin	17.8	17.8	17.8
		31.2	31.9	34.7
Cure				
	Spar	9.8	9.8	9.8
	Skin	10.4	10.4	10.4
		20.2	20.2	20.2
Trim				
	Spar	2.2	2.2	2.2
	Skin	3.4	8.5	3.4
		5.7	10.7	5.7
Assembly		4.0	4.1	4.1
TOTAL		66.5	72.3	69.9

A = FULL PLIES

PAD
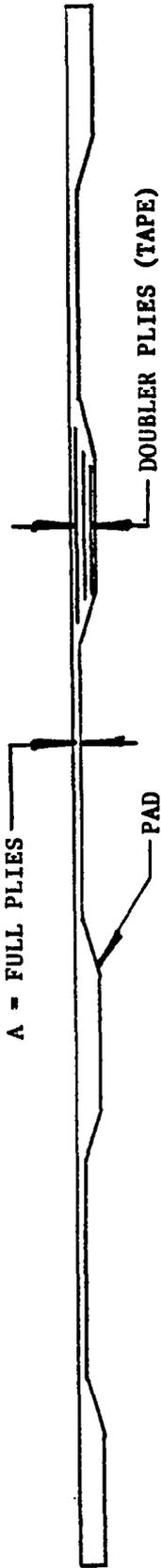
DOUBLER PLIES (TAPE)

BASIC COST/SKIN = 2.956 HOURS

NUMBER OF PLIES

	2	4	6	8	10	12
A	.616 HRS	1.232	1.848	2.464	3.080	3.696
B	.219	.351	.483	.615	.747	.879

ESTIMATED SKIN COST = BASIC + A + (NUMBER OF PADS x B)

EXAMPLE: SKIN WITH 4 FULL PLIES AND 4 PADS BUILT UP WITH 8 DOUBLER PLIES (TAPE)

$$2.956 + 1.232 + (4 \times .615) = 6.648 \text{ HOURS}$$

NOTE: COMPLEXITY OF THE PART BEING ESTIMATED HAS A SIGNIFICANT IMPACT ON COST. THIS PART IS A SIMPLE LAYUP WITH LITTLE OR NO CONTOUR AND NO INTERIOR OR EXTERIOR CORNERS. THEREFORE THERE ARE A MINIMUM OF COMPACTION CYCLES REQUIRED.

Table E-5. Composite Box Beam Assembly Lay-Up Cost Analysis, - Skin.

A = FULL PLIES (FABRIC)

BASIC COST/LAYUP = 2.269 HOURS

NUMBER OF PLIES

	2	4	6	8	10	12
A	1.304	2.607	3.911	5.214	6.518	7.821
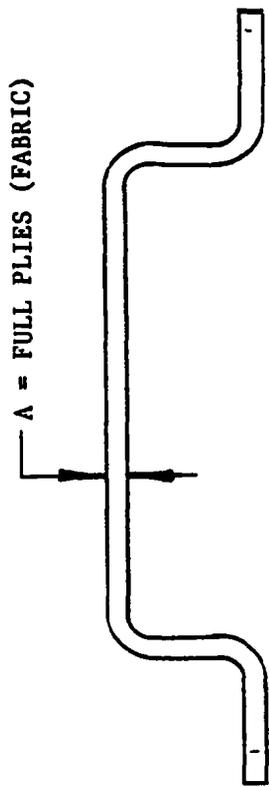
NOTE: THIS LAYUP YIELDS TWO STRINGER DETAIL PARTS

ESTIMATED STRINGER COST = BASIC COST + A

EXAMPLE: STRINGER WITH 12 FULL PLIES

2.269 + 7.821 = 10.09 HOURS

$$\frac{10.09 \text{ HOURS}}{2} = 5.045 \text{ HOURS/STRINGER}$$

COMPLEXITY: ALTHOUGH THIS PART IS A RELATIVELY SIMPLE LAYUP, IT REQUIRES EVERY PLY TO BE COMPACTED DUE TO THE EXTERIOR AND INTERIOR CORNERS.

Table E-6. Composite Box Beam Assembly Lay-Up Cost Analysis, - Stiffener.

A = FULL PLIES (FABRIC)

B = DOUBLER PLIES (TAPE)
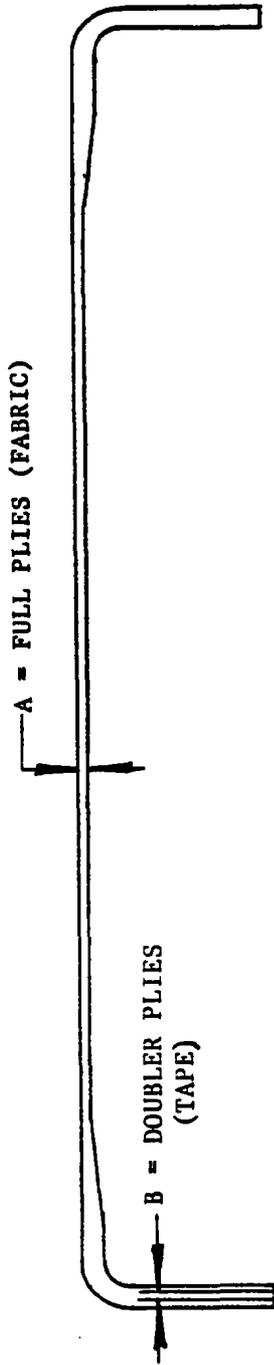
BASIC COST/LAYUP = 2.269 HOURS

NUMBER OF PLIES

	2	4	6	8	10	12
A	.996	1.920	2.879	3.839	4.799	5.759
B	.529	1.059	1.588	2.118	2.647	3.176

ESTIMATED SPAR COST = BASIC COST + A + (NUMBER OF PADS x B)

EXAMPLE: SPAR WITH 4 FULL PLIES AND 2 PADS BUILT UP WITH 8 DOUBLER PLIES (TAPE) EACH

$$2.269 + 1.920 + (2 \times 2.118) = 8.425 \text{ HOURS}$$

COMPLEXITY: SIMPLE LAYUP REQUIRING 50% (OF PLIES) COMPACTION CYCLES TO PREVENT POROSITY IN THE CORNERS.

Table E-7. Composite Box Beam Assembly Lay-Up Cost Analysis, - Shear Web.
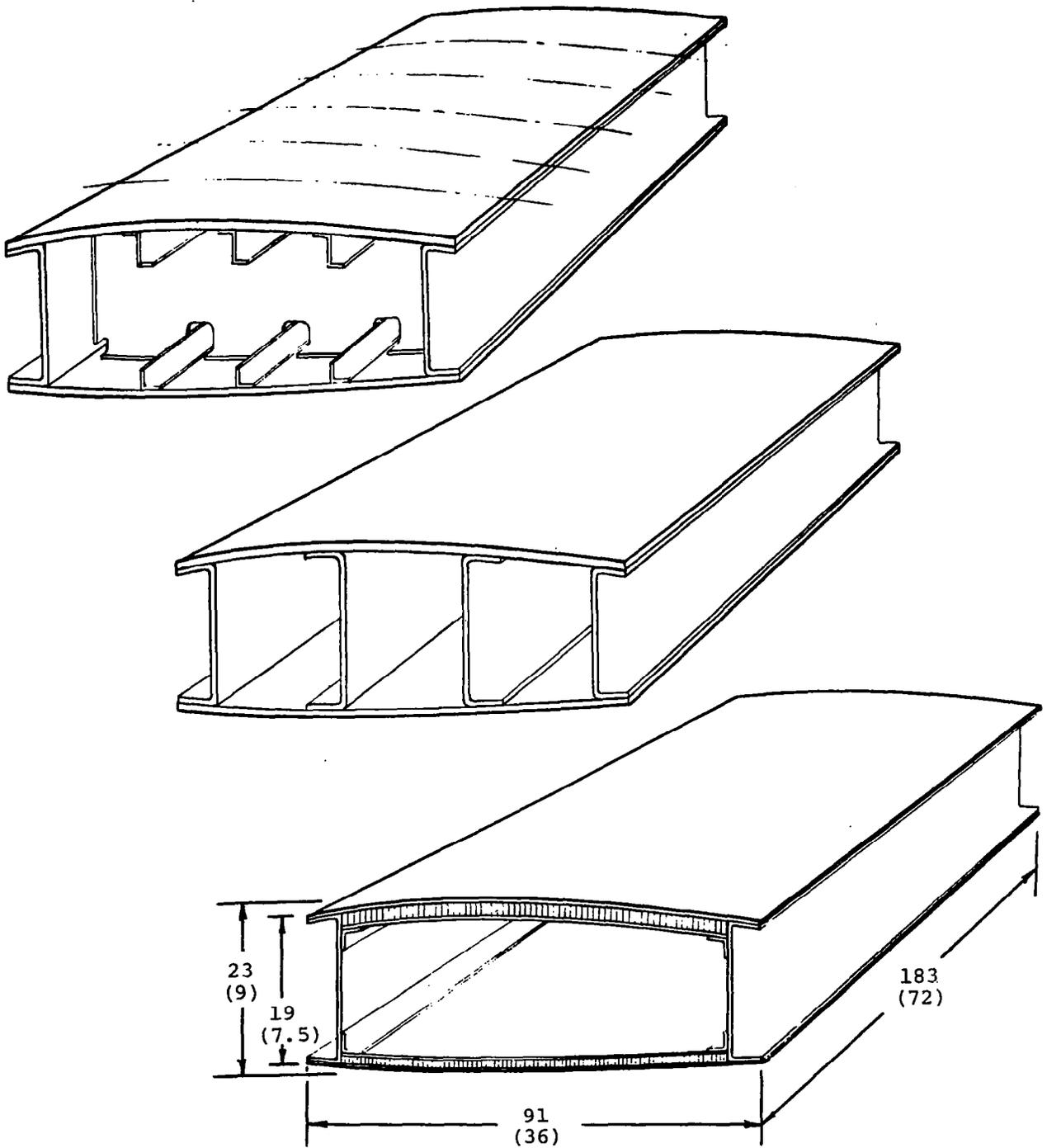
Figure E-1. Idealized Wing Box Beam Used for Manu-
facturing and Cost Studies. Dimensions
in cm (in.)

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APPENDIX F - COST ANALYSES

A number of approaches were investigated to evaluate the merit of structural performance improvement through the use of composite materials in terms of value added to an aircraft. In the process, several simple relationhips between costs and other factors were developed that will be presented as a matter of interest, and as a basis for further studies. Values for premiums for weight saving that do appear useful were also obtained, and a rationale for analysis was developed. These will also be presented.

Simple Relationships Between Costs and Other Factors

Analysis of data on current general aviation aircraft reveal that a simple, linear relationship exists relating aircraft cost in dollars to range, payload, and cruising speed, thus

$$C = 0.00077(RPS) \tag{F-1}$$

where

C   aircraft cost, $

R   range, miles

P   payload, lbs.

S   crusing speed, m.p.h.

Data establishing this relation are plotted in figures F-1 to F-4. The agreement among manufacturers on pricing shown is rather remarkable.

Further simple relations are shown in figures F-5 to F-10. In figures F-5 to F-8 the equation

$$C = 0.0078 \, W^2 \tag{F-2}$$

is shown to provide a close correlation between cost C and gross weight W for piston engine (single engined) aircraft. For twin engines, the equation becomes (fig. F-9)

$$C = 0.011 \, W^2 \tag{F-3}$$

and for turbofan/turbojet aircraft the price increases to (fig. F-10)

$$C = 175 \, W \tag{F-4}$$

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Correlating Relationship

Review of the results of the correlations shown in figures F-1 to F-10 led to the development of a unifying correlation

$$X = \text{(Useful Load)}^2 \text{ (Cruising Speed)} \frac{\text{Range}}{\text{Weight of Fuel}} \quad \text{(F-5)}$$

where

Useful load = sum of fuel load plus payload, lbs.

Cruising speed, nautical miles/hr.

Range, nautical miles

Weight of fuel, lbs.

Correlations achieved with this equation are illustrated in figures F-11 to F-13. Data for these plots were obtained from ref. 19.

Utilizing these expressions for cost, a premium for weight savings can be obtained as the derivative of cost with respect to useful load. Average values for this derivative are listed in table F-1. Because of the scatter in the data, the premiums to be used in the analyses have been modified somewhat and are also listed in table F-1. The premium data, a measure of the value of weight savings, are useful in conjunction with weight savings, cost of manufacture and material cost for the evaluations of structural economic efficiency. Not surprisingly, these realistic premiums are rather higher than are generally accepted in the industry. For generalized studies, the use of somewhat lower values should accordingly be considered.

Study of Relation of Wing Structure Cost to Wing Geometry

With the parametric studies of Appendix D for the determination of wing weights in terms of aircraft design parameters, and the ranges of manufacturing costs and premiums for weight savings established from the manufacturer visitations, the way is clear to

study overall cost-effectivenesses of various materials for
wing construction for general aviation aircraft.

Procedure for preliminary surveys was as follows:

(1) Find a raw material cost from the first section of
     Appendix E for the materials to be evaluated -
     using the cost of prepreg as raw material cost for
     composites.

(2) Assume as a zeroth approximation that manufacturing
     costs per pound of all materials are the same for the
     same type of structure. (In the following examples
     we used $100/kg.) ($45/lb.).

(3) Calculate wing weight variations with changes in con-
     figuration (aspect ratio, taper ratio) for the vari-
     ous materials.

(4) Calculate the corresponding wing costs as the product
     of (3) x [(1) + (2)].

Results of such preliminary calculations must be interpreted
with caution because of the oversimplifications (such as the
assumption of constant manufacturing cost per pound regardless
of material) employed. Nevertheless, the implication is evi-
dent that wing geometry is an important parameter affecting
relative cost-effectivenesses of various materials as the fol-
lowing examples illustrate.

The examples, shown in figures F-14 to F-19, consider the
effective costs of wings of the same area for the same aircraft,
of varying aspect ratio and two taper ratios, 1 and 1/4. Three
materials were considered: 7075-T6 aluminum alloy, MED-C/Epoxy
and E-Glass/Epoxy.

First, total wing weights were calculated, using the BOZO
code described in Appendix D, and a 3-web box beam with a thick-
ness ratio $\frac{D}{C}$ = 0.24. Results are shown in figures F-14 and
F-15. Weights are characteristically: aluminum heaviest, MED-C
lightest, glass intermediate, for the wide range of wing con-
figurations for which a 3-web structure is appropriate. For

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the very low aspect ratio, high taper ratio cases more webs are needed to utilize the composites most effectively. Such additional webs were not explored for this example because to do so would have complicated the cost analysis and anyway it would probably not change the overall inferences to be drawn.

Next, wing costs were calculated as in (4) above with the results shown in figures F-16 and F-17. Here the results are primarily a function of the assumption that manufacturing costs are the same per pound for all three materials. Thus, the MED-C is competitive where it saves the most weight -- at high aspect ratios and low taper ratios -- as is glass, though glass does well economically across the board.

If premiums are added for weight savings, the composites become more cost-effective, but substantial premiums are required to make substantial differences. Even $165/kg. ($75/lb.) does not make these composite constructions superior to aluminum at low aspect ratios and high taper ratios (figs. F-18 and F-19).

Perhaps of greatest significance for final analyses is this preliminary finding that wing configuration has such a profound influence on cost-effectiveness. General conclusions must not be drawn from specific designs as has been done in some past studies. Also, it must be emphasized that for some of the cases considered in this example, deflections may be critical. Deflections for the MED-C and the aluminum wings are essentially equal for all cases; for the worst case (aspect ratio 10, taper ratio 1) the tip deflection is approximately 24% of the semispan. For the E-Glass, the corresponding number is 56%! Stiffness criteria must be established if rational comparisons among designs are to be achieved.

DEVELOPMENT OF DIRECTIONS FOR RESEARCH AND DEVELOPMENT ON MANUFACTURING TECHNOLOGY

The studies of manufacturing technology for advanced composites for general aviation aircraft applications reported in

this Appendix have generated the following directions for research and development -

(1) Development of high energy supply systems for rapid cure cycles for composites.

(2) Development of robotics to reduce hand operations, not only for tape layup but especially for ply placement.

(3) Development of filament winding technology for fuselage sections.

(4) Development of the pultrusion process for production of advanced (curved) shapes, particularly of minimum gages, and capable of utilizing woven-fabric reinforcements and medium-low density cores.

(5) Development of large braiding machinery for the braiding of triaxial, carbon fiber, fabrics and 3-D reinforcements.

(6) Development of high speed, fully automated non-destructive inspection equipment.

Table F-1. Premium Calculations

Aircraft Class	$\dfrac{d \ (\text{Cost})}{d \ (\text{Useful Load})}$	Premium
Single Engine Piston	$190/kg. ($87/lb.)	$165/kg. ($75/lb.)
Multi-Engine Piston	$740/kg. ($335/lb.)	$660/kg. ($300/lb.)
Turbo-Prop	$1170/kg. ($530/lb.)	$1100/kg. ($500/lb.)
Turbojet	$1060/kg. ($480/lb.)	$1100/kg. ($500/lb.)

Figure F-1. Cost-Performance Relationships for General Aviation Piston Engine
Aircraft, Manufacturer A.
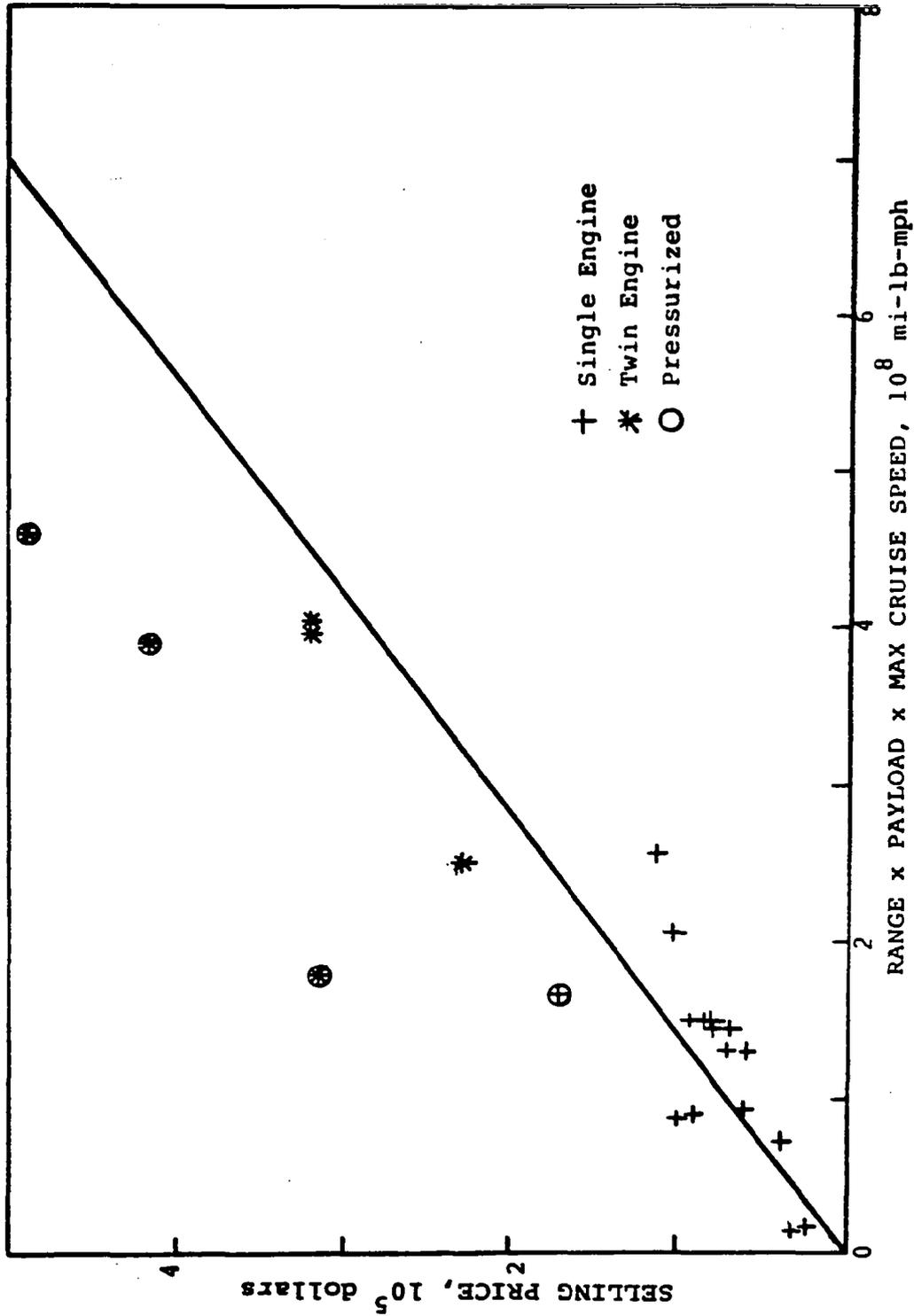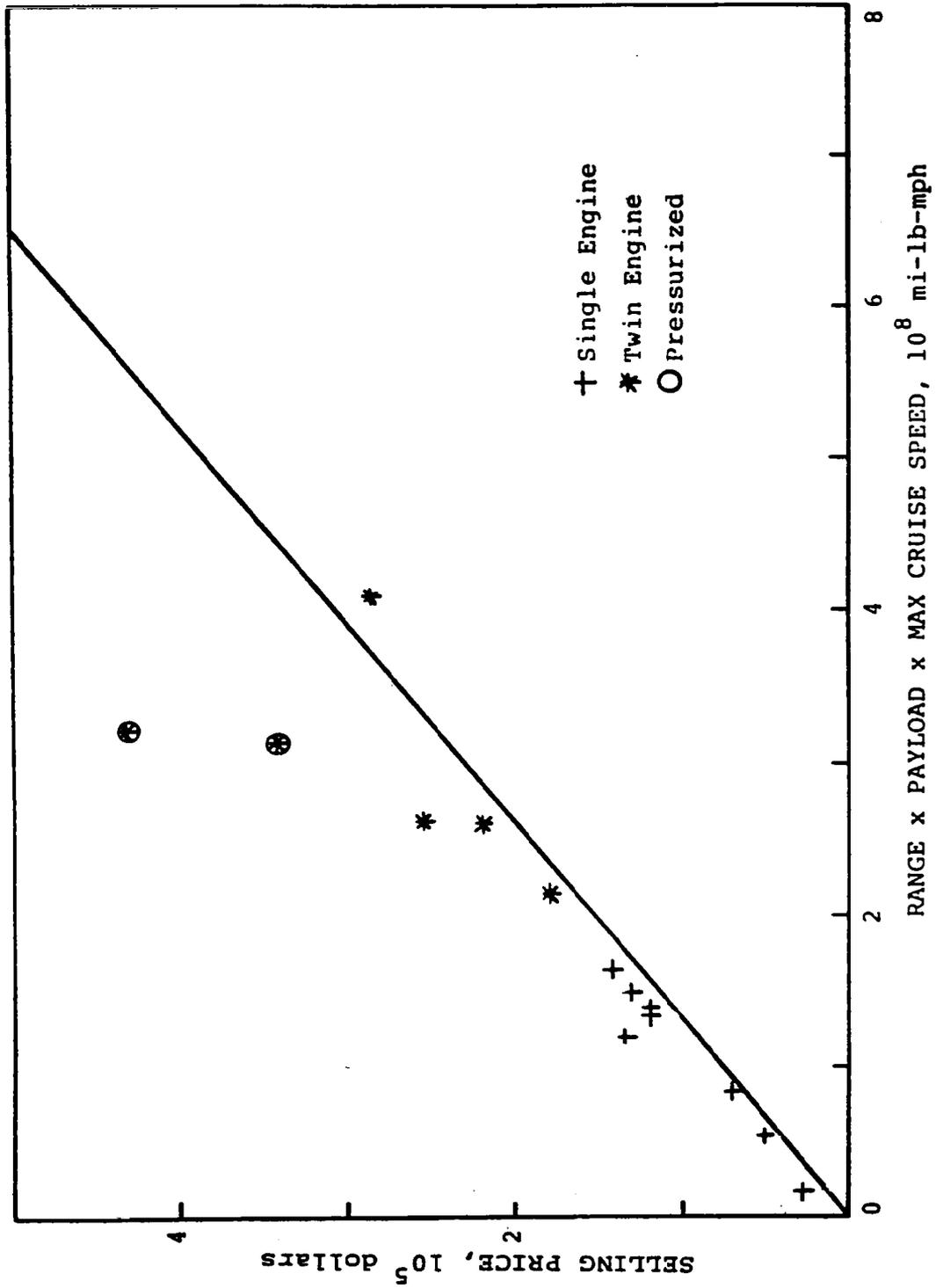
Figure F-2. Cost-Performance Relationships for General Aviation Piston Engine Aircraft, Manufacturer B.

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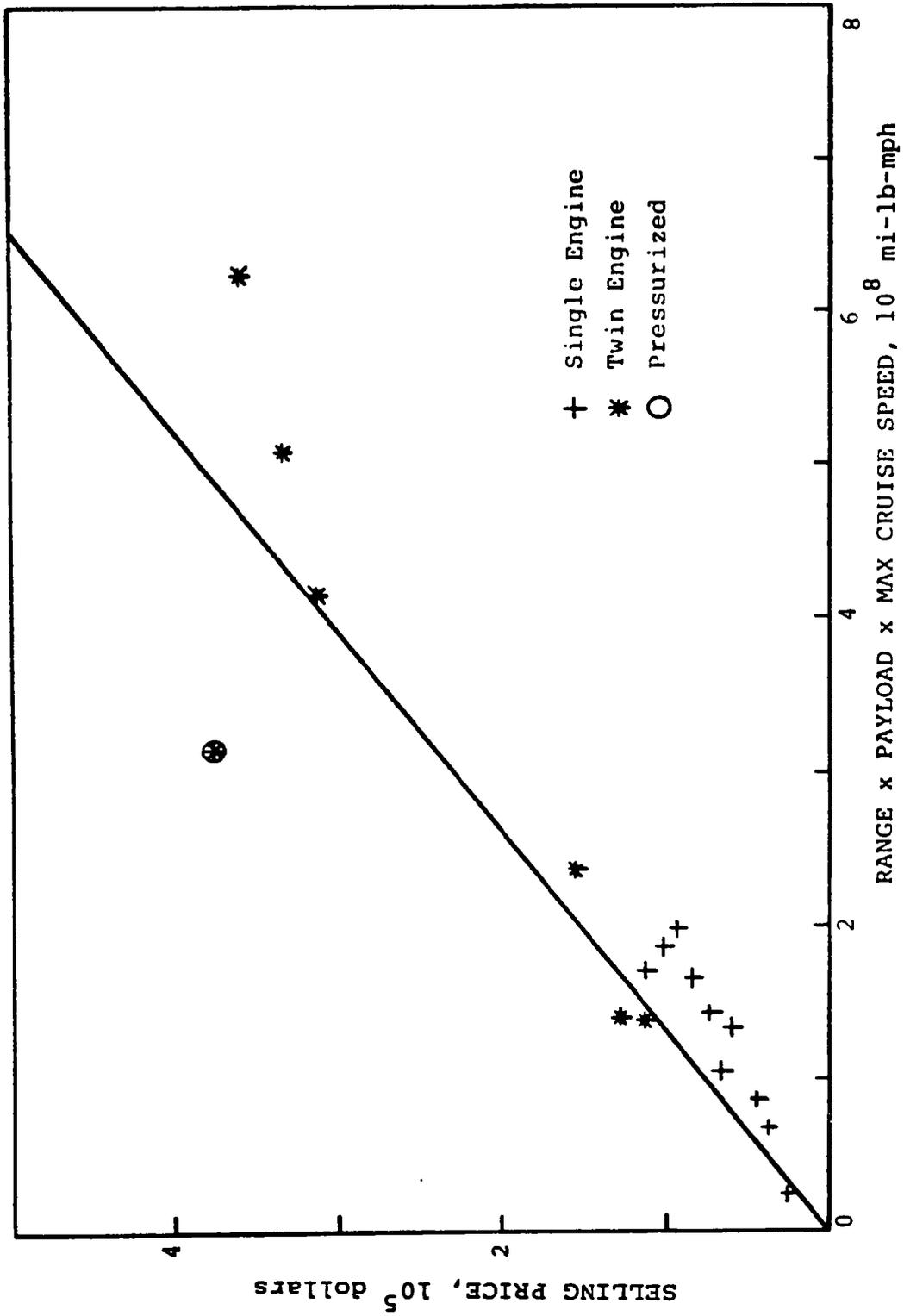
Figure F-3. Cost-Performance Relationships for General Aviation Piston Engine Aircraft, Manufacturer C.
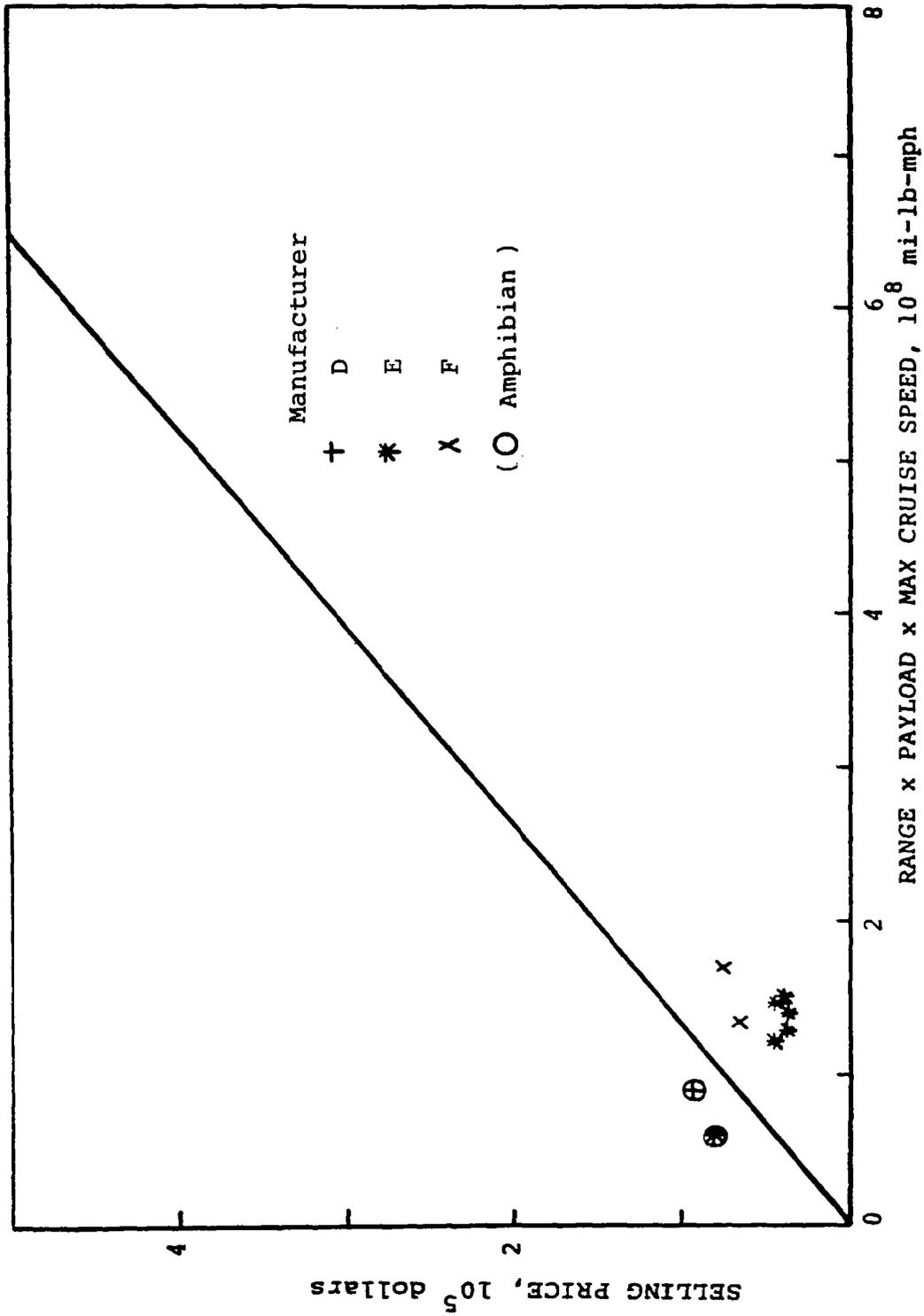
Figure F-4. Cost-Performance Relationships for General Aviation Piston Engine Aircraft, Various Manufacturers.
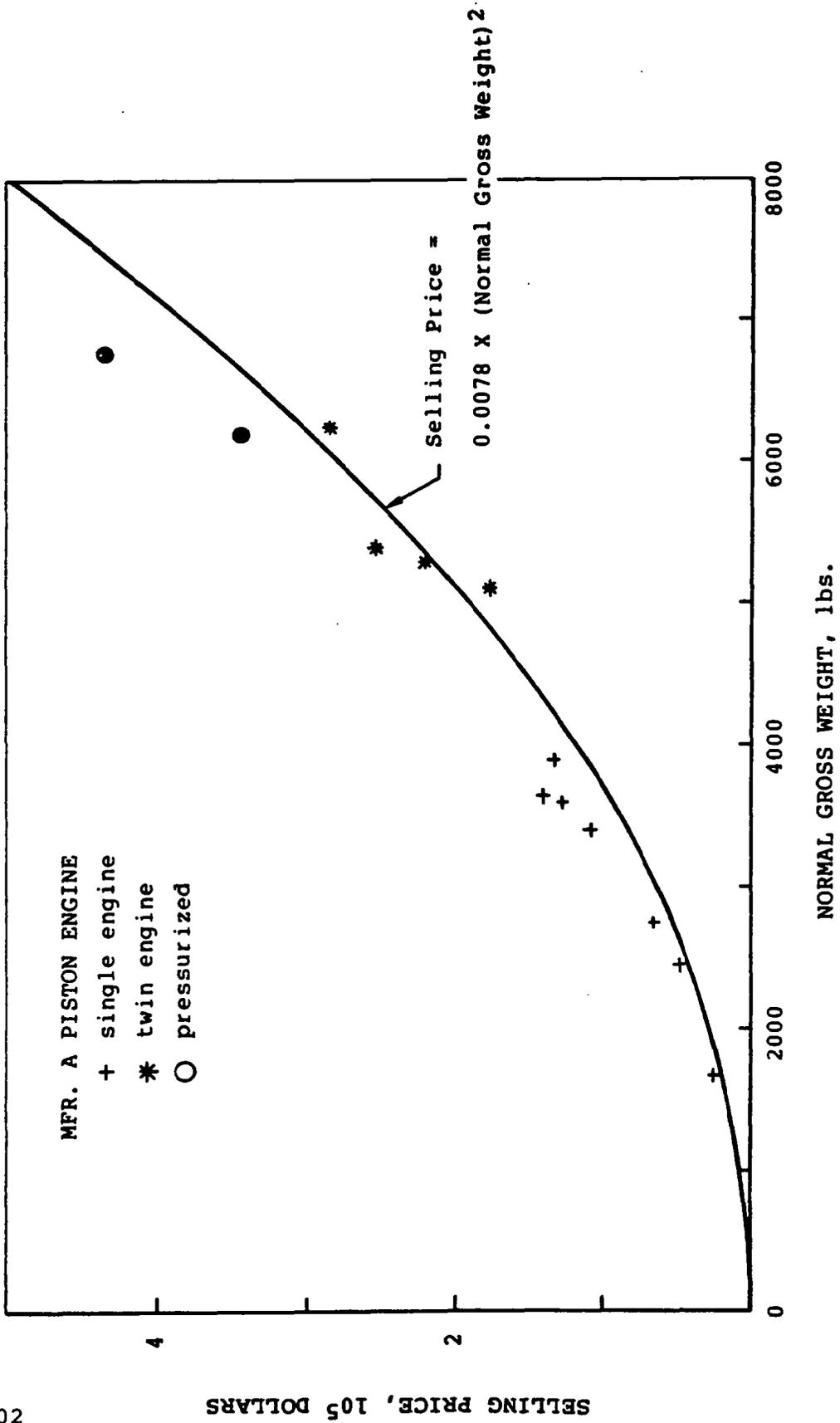
Figure F-5. Correlation of Costs of General Aviation Aircraft - Manu-
facturer A, Piston Engine.
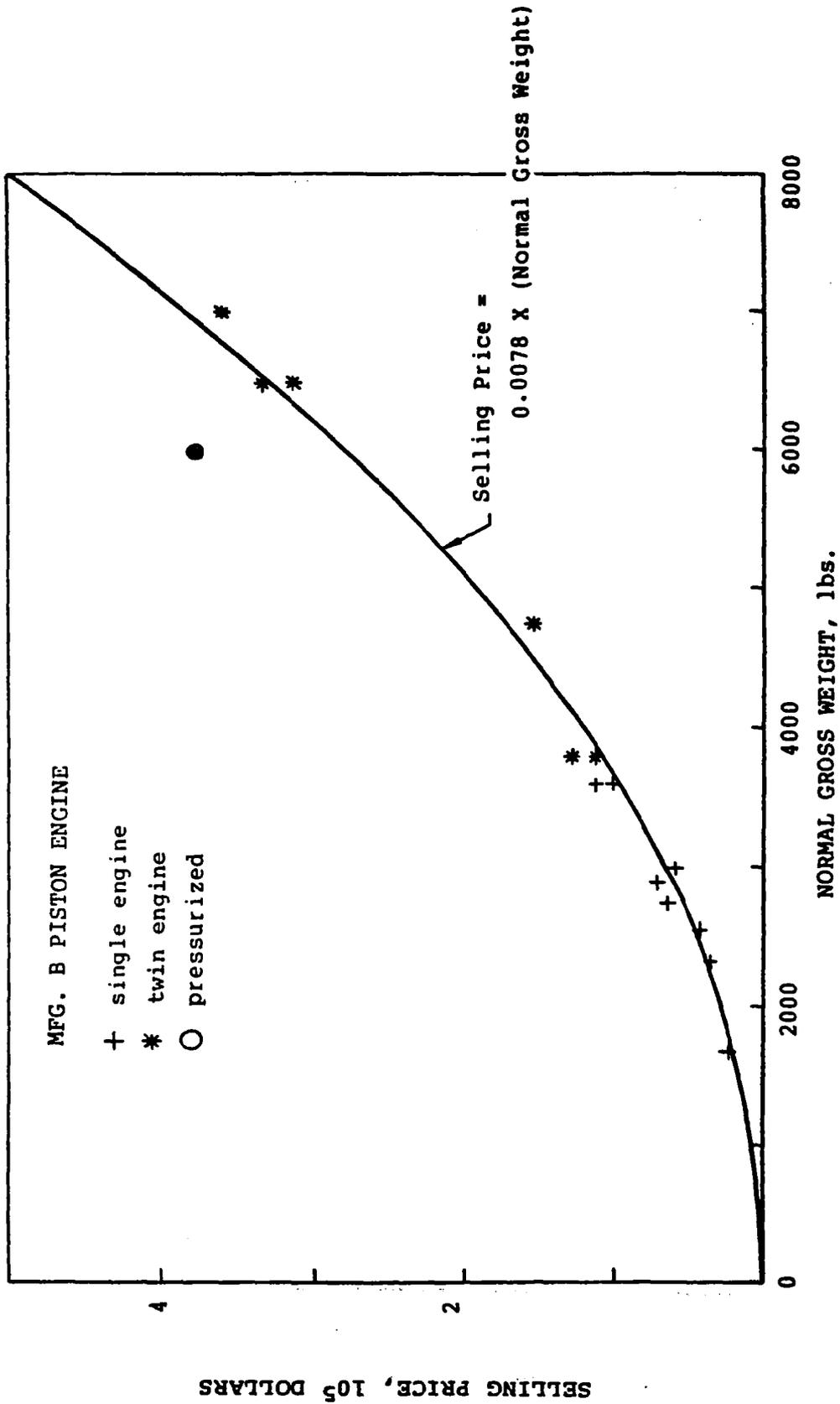
Figure F-6. Correlation of Costs of General Aviation Aircraft - Manufacturer B, Piston Engine.

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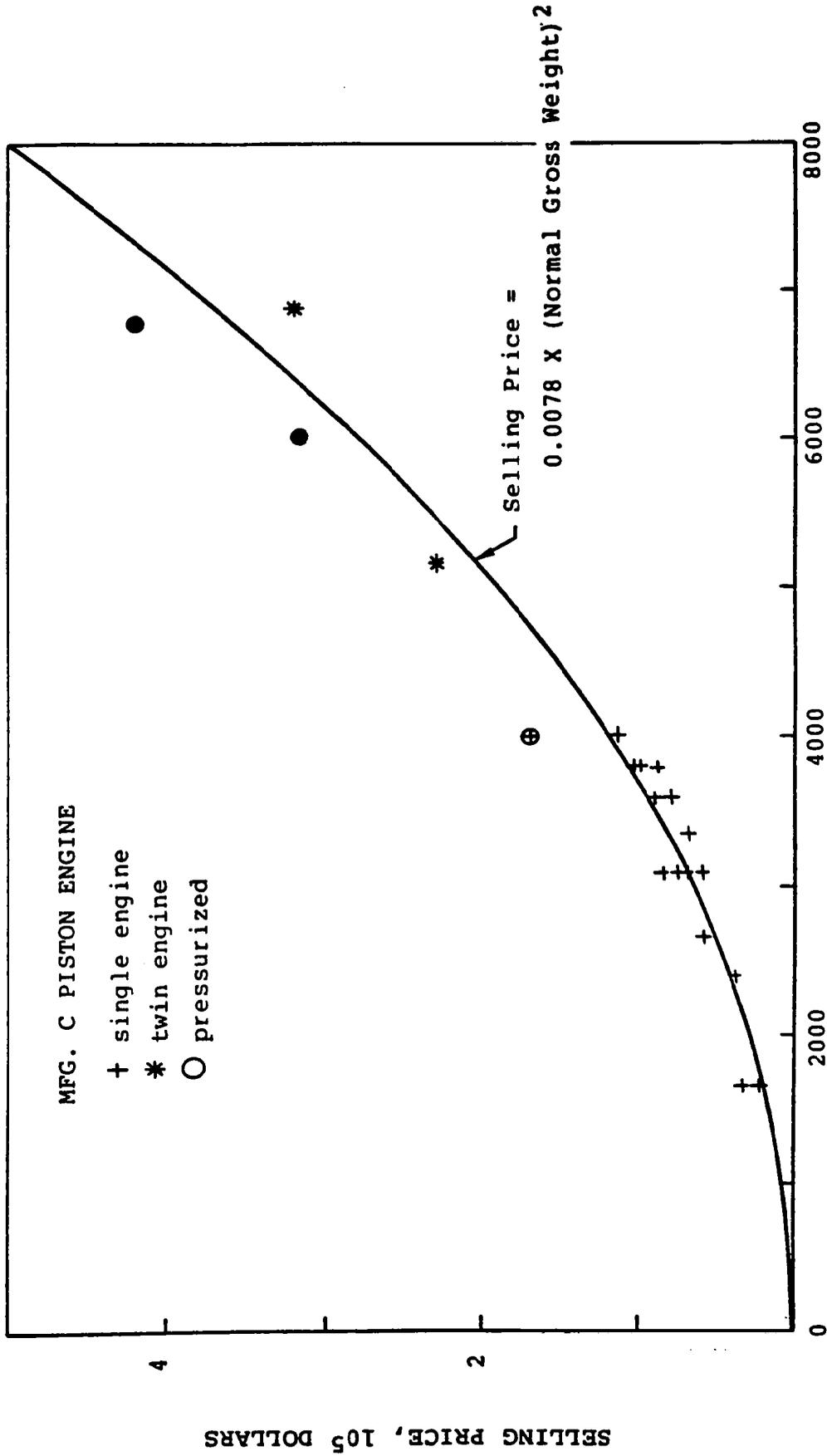
Figure F-7. Correlation of Costs of General Aviation Aircraft – Manufacturer C, Piston Engine.
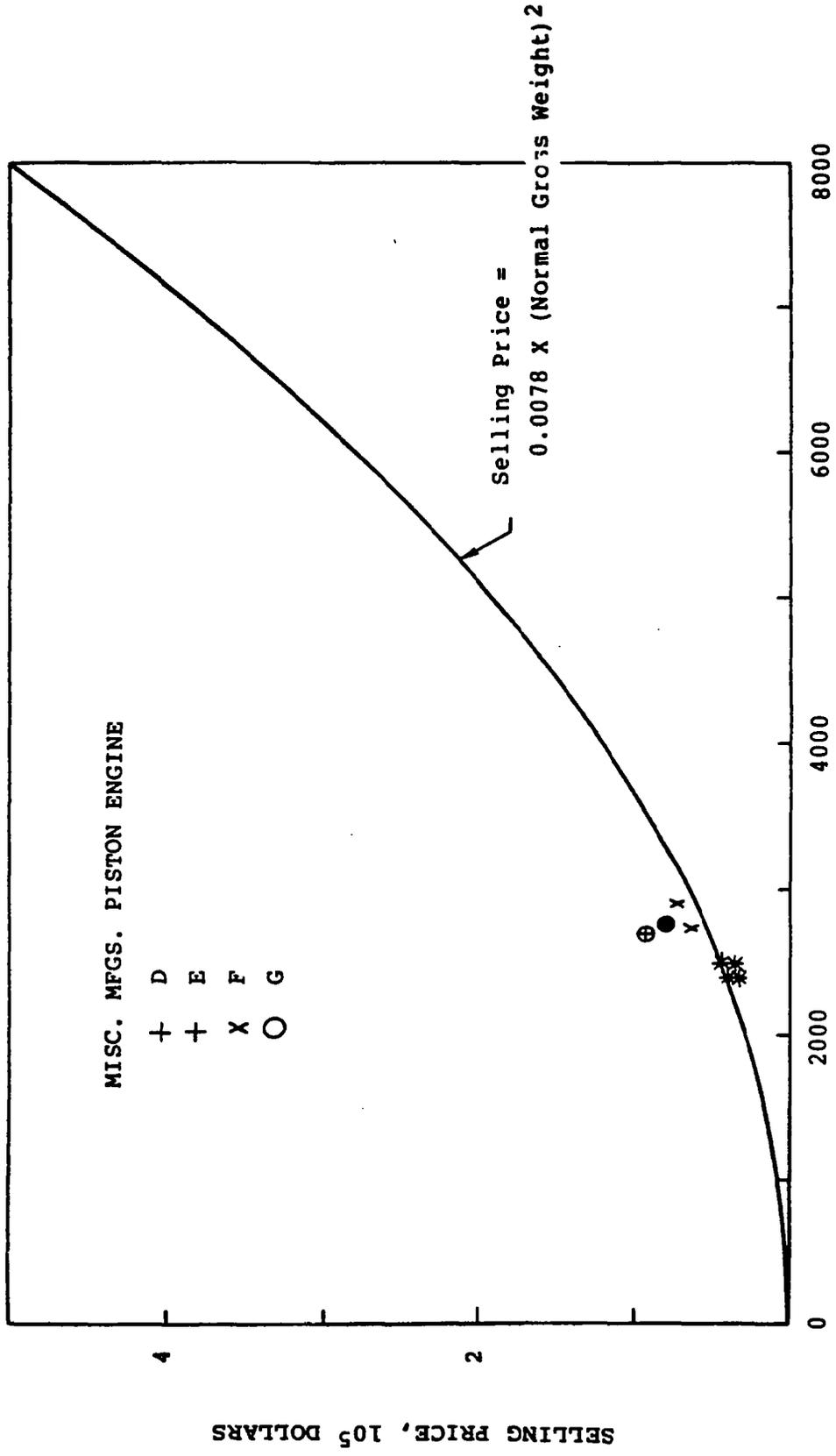
Figure F-8. Correlation of Costs of General Aviation Aircraft – Miscellaneous Manufacturers, Single Piston Engine.
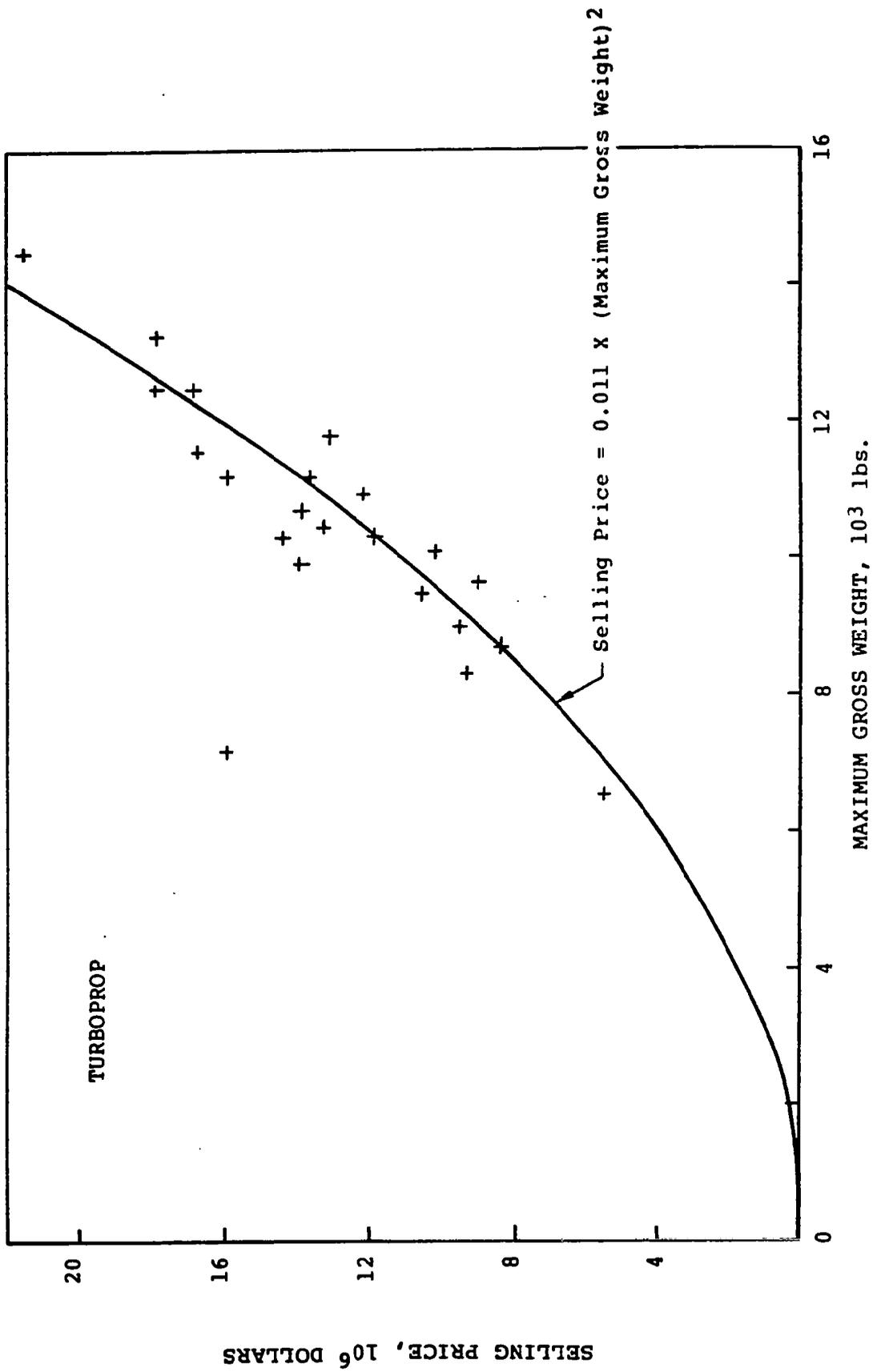
Figure F-9. Correlation of Costs of General Aviation Aircraft – Miscellaneous Manufacturers, Turboprops.
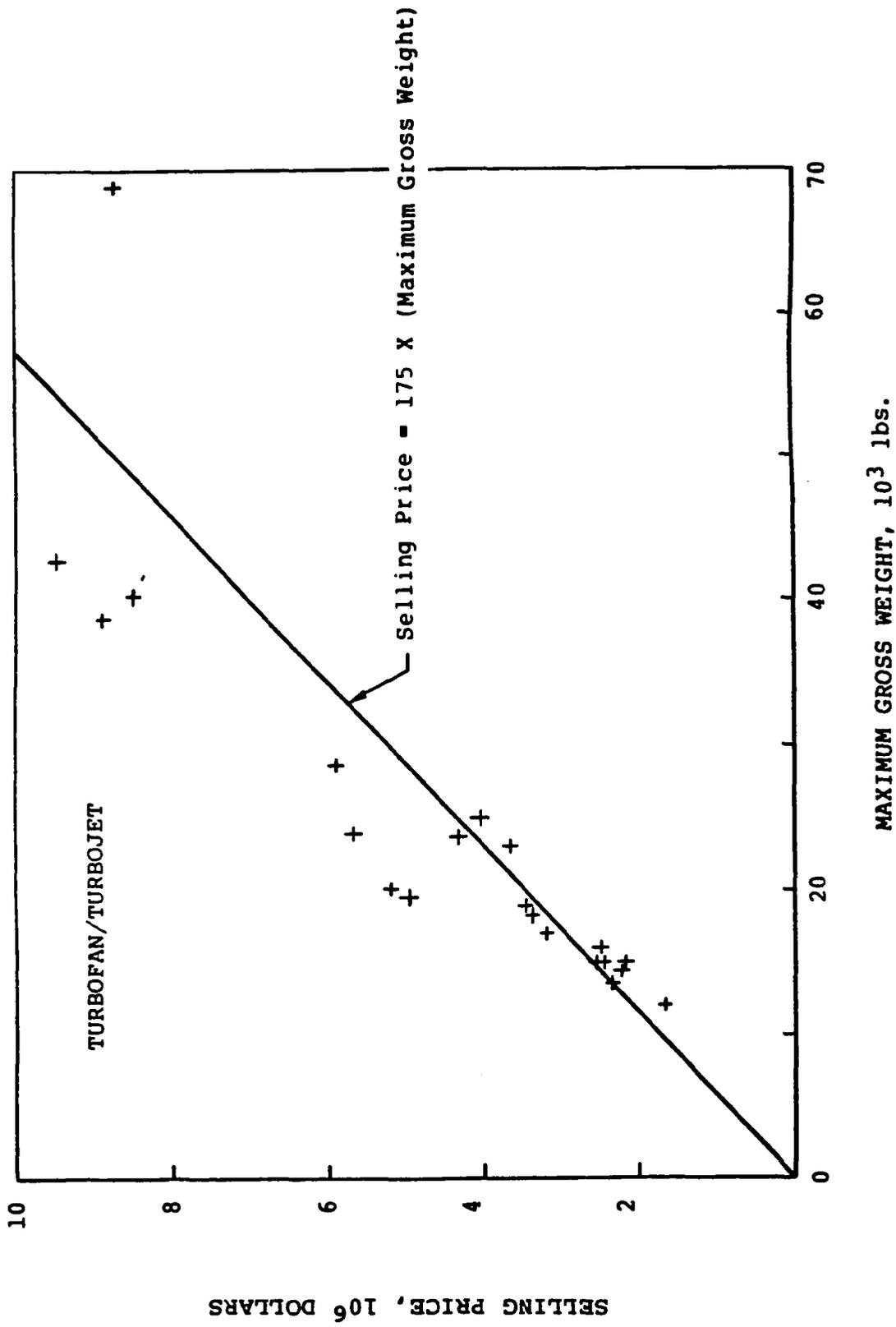
Figure F-10. Correlation of Costs of General Aviation Aircraft - Miscellaneous Manufacturers, Turbofans/Turbojets.
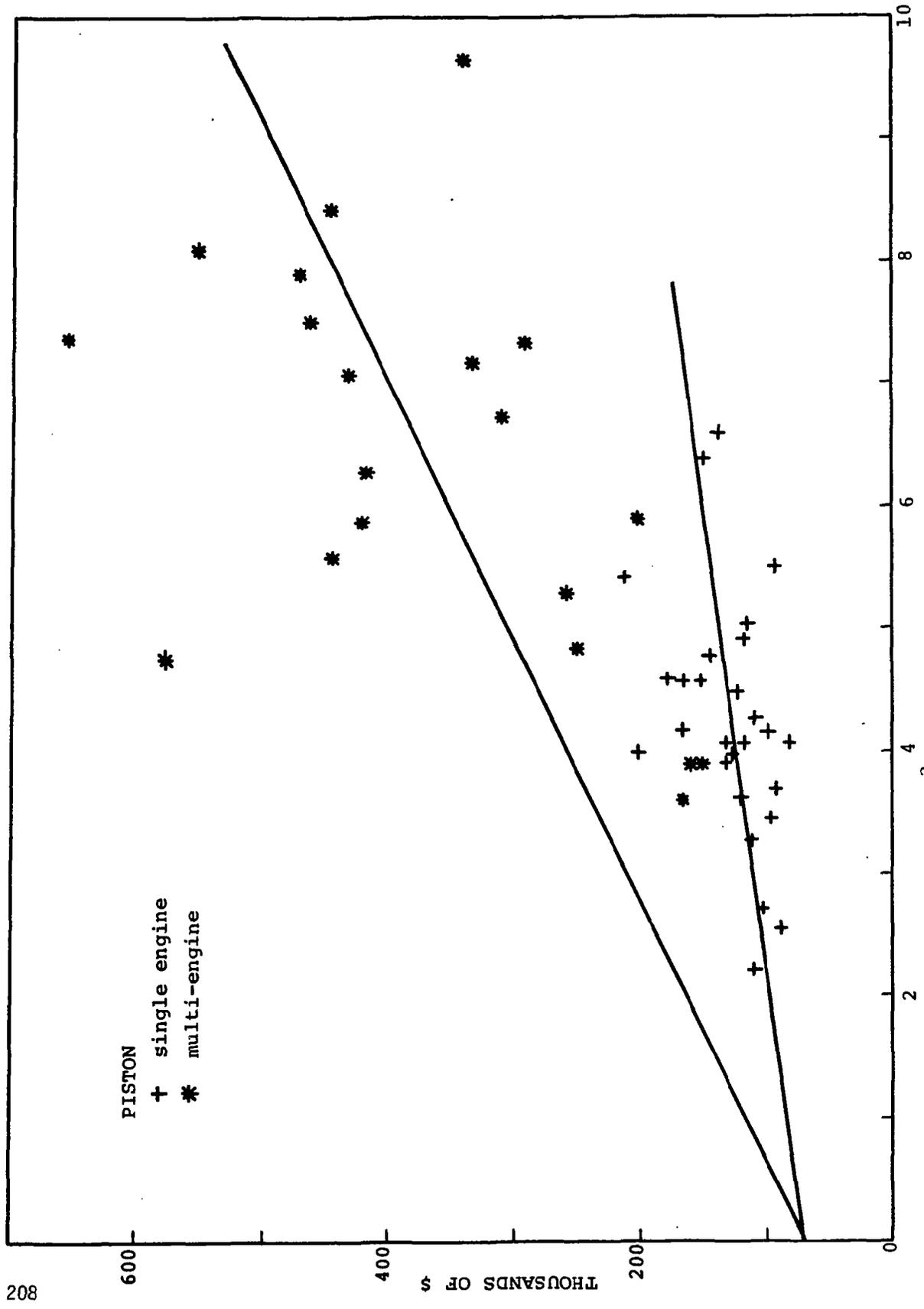
Figure F-11. Cost/Performance Relation, Single and Multi-Engine Piston Aircraft.

USEFUL LOAD^2 x CRUISE SPEED X nmi/1b, 10^8lb nmi^2/hr.

THOUSANDS OF $

PISTON

+ single engine

* multi-engine

208
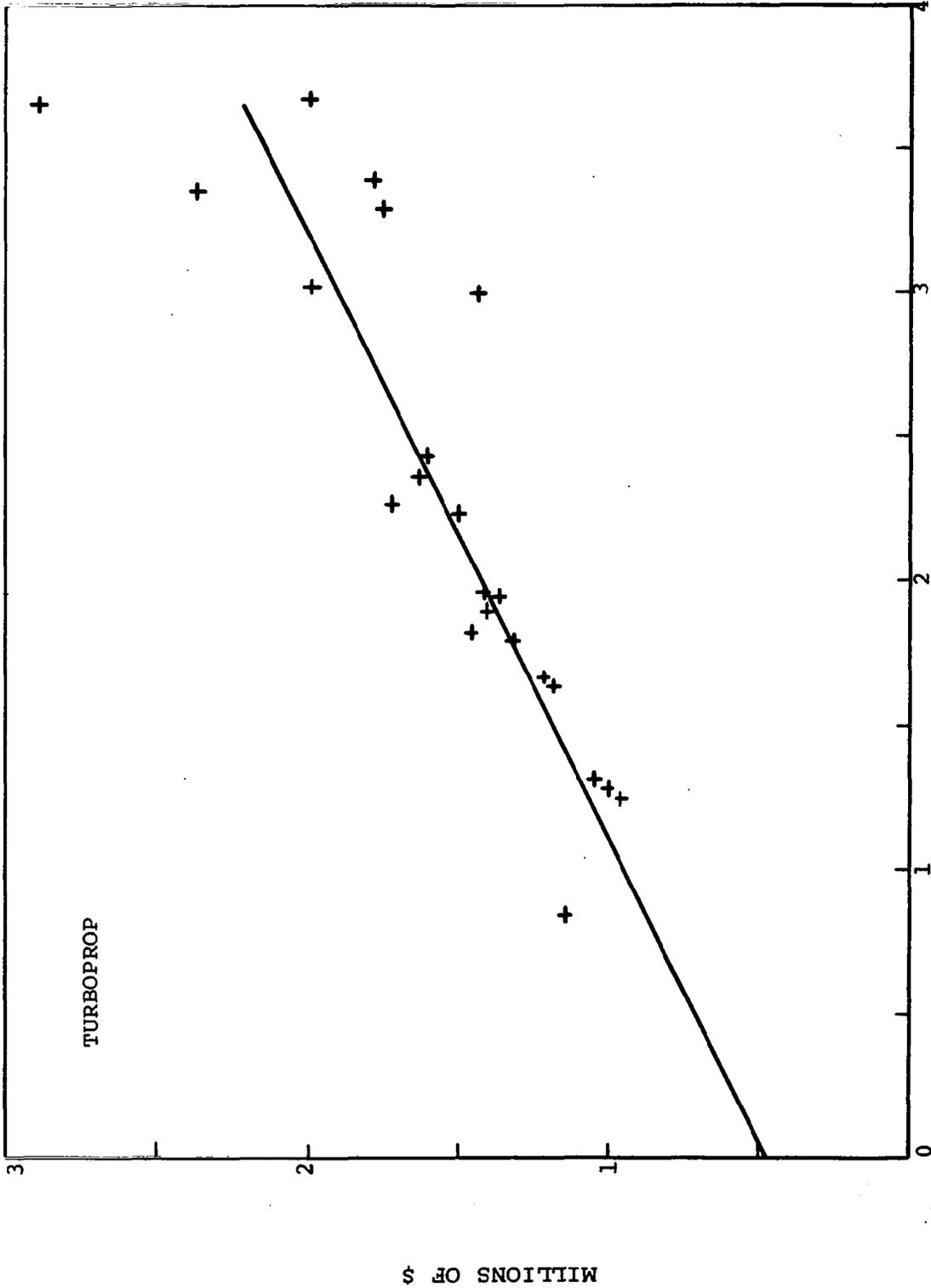
Figure F-12. Cost/Performance Relation, Turbo-Prop Aircraft.

TURBOPROP

MILLIONS OF $

USEFUL LOAD^2 x CRUISE SPEED x nmi/lb, 10^9lb nmi2/hr.
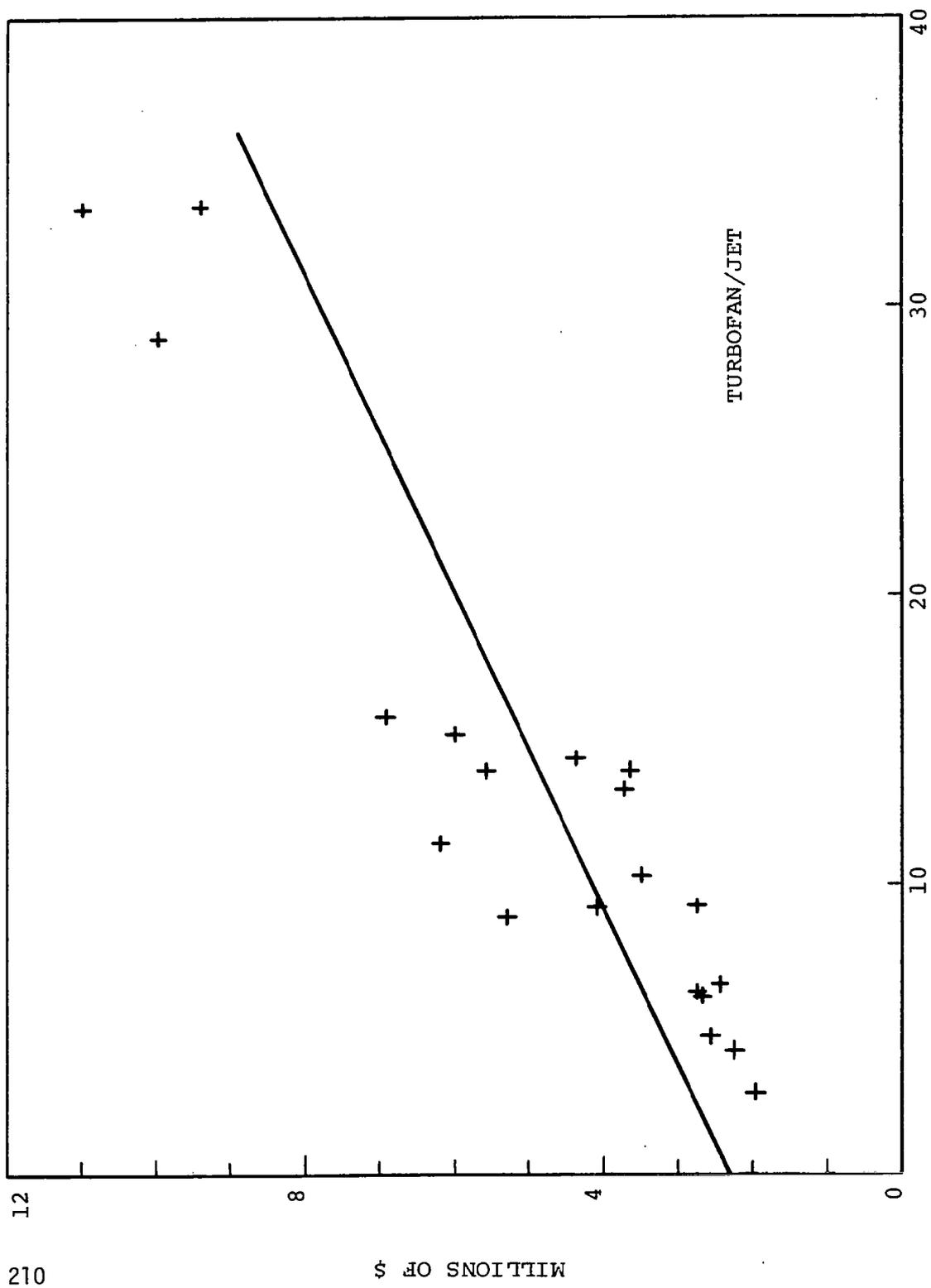
Figure F-13. Cost/Performance Relation, Turbofan/Jet Aircraft
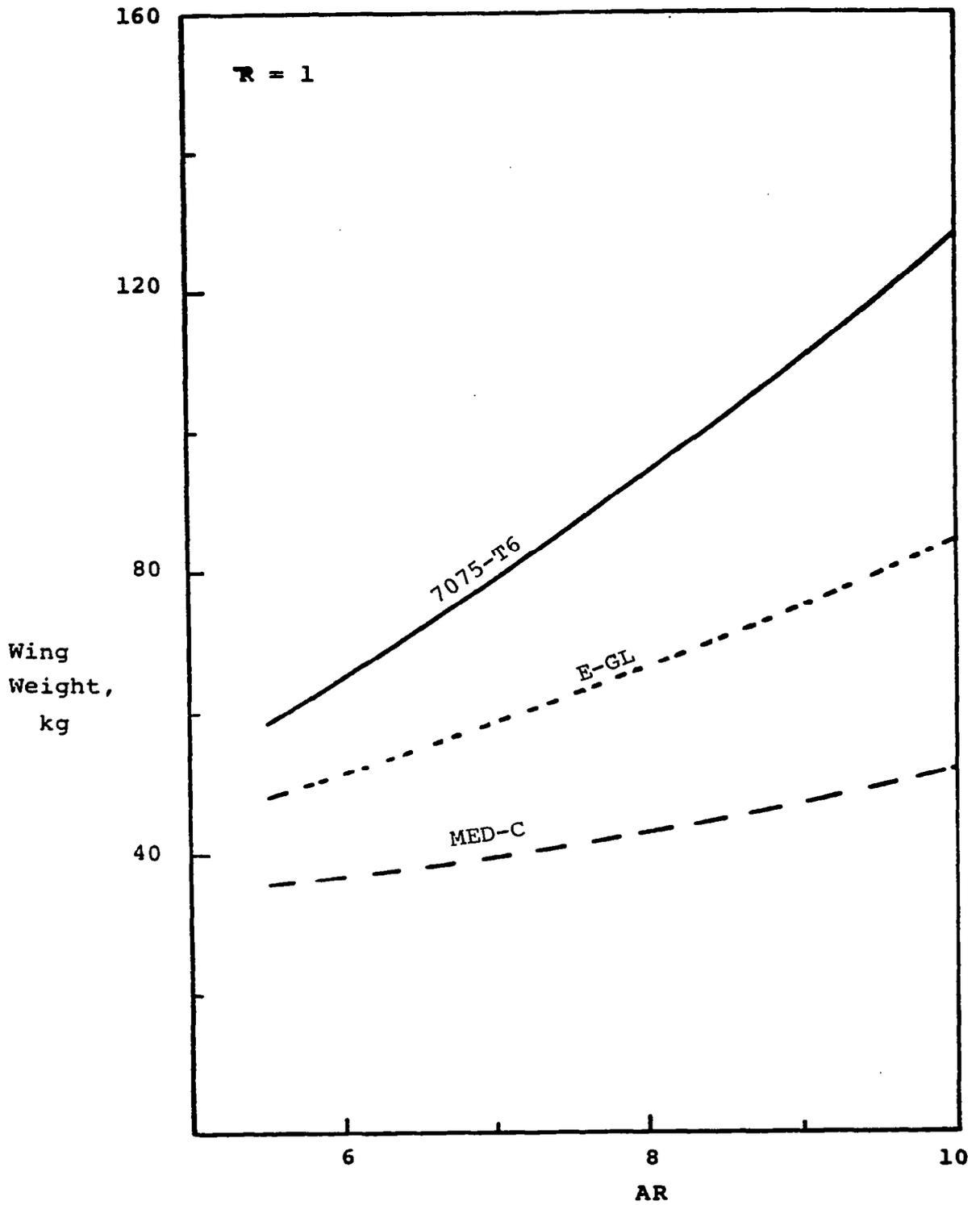
Figure F-14. Variation of Wing Weight with Aspect Ratio for a Straight Wing.
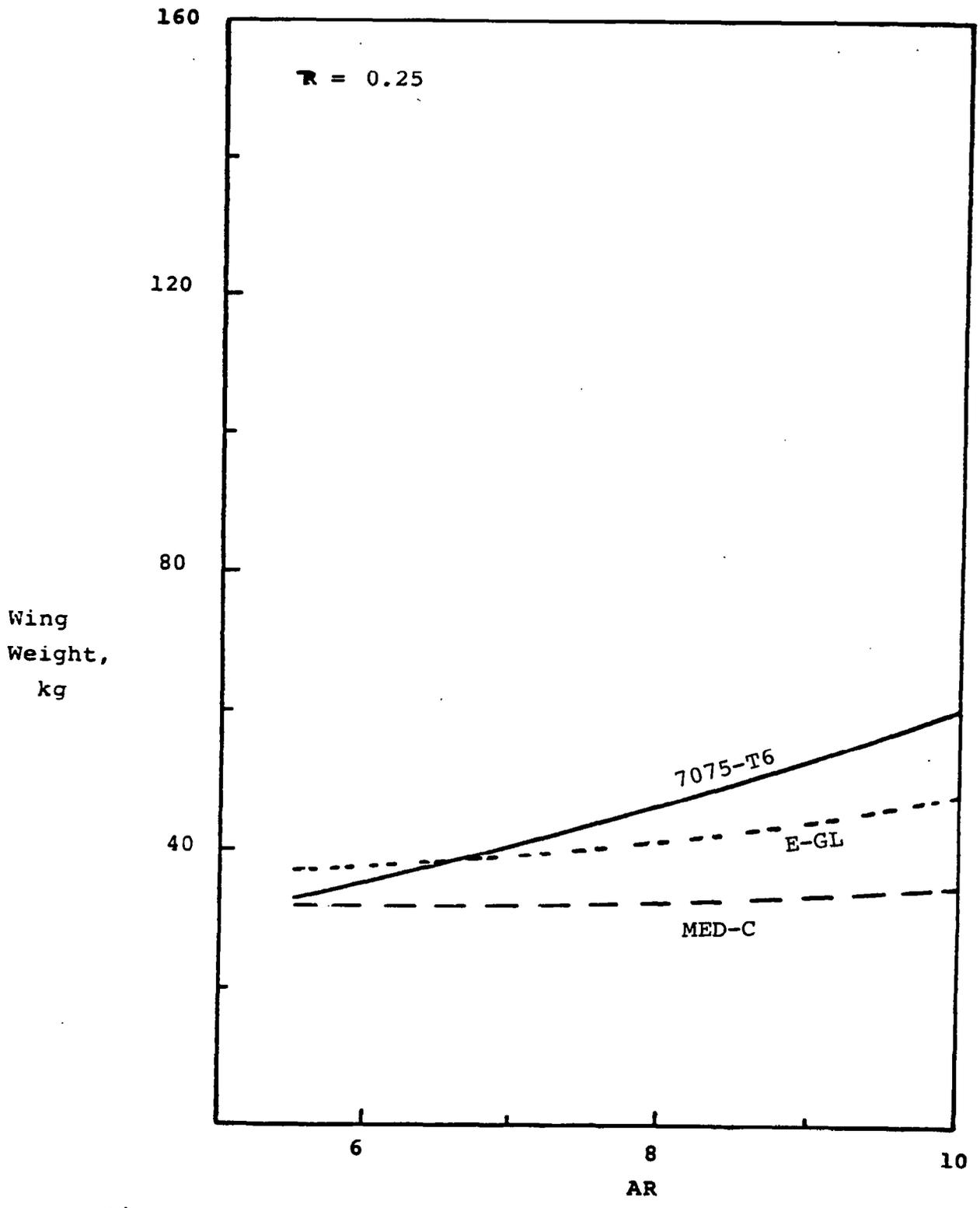
Figure F-15. Variation of Wing Weight with Aspect Ratio for a Wing with Straight Taper.
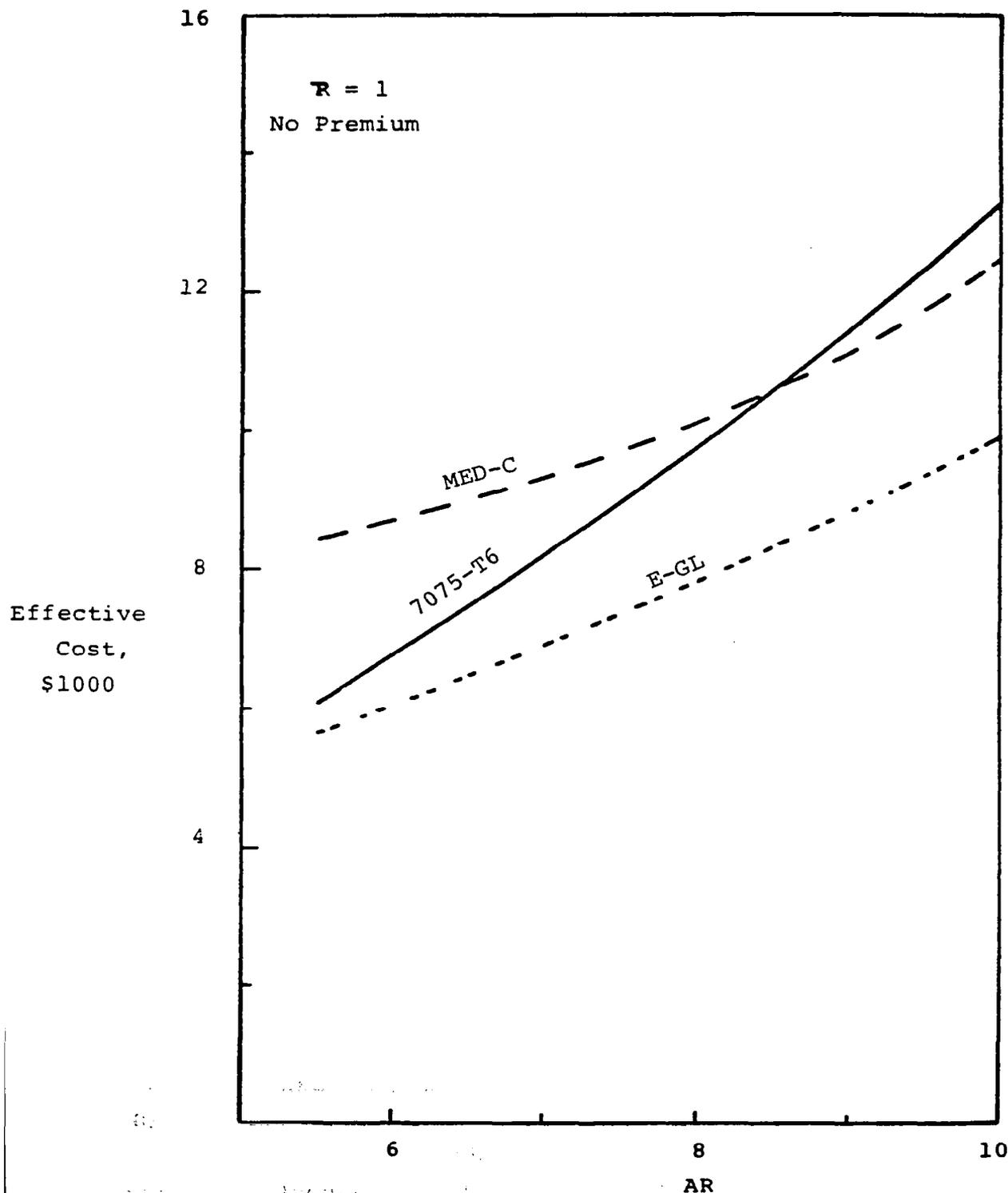
Figure F-16. Manufactured Costs of Straight Wings of Various Aspect Ratios.
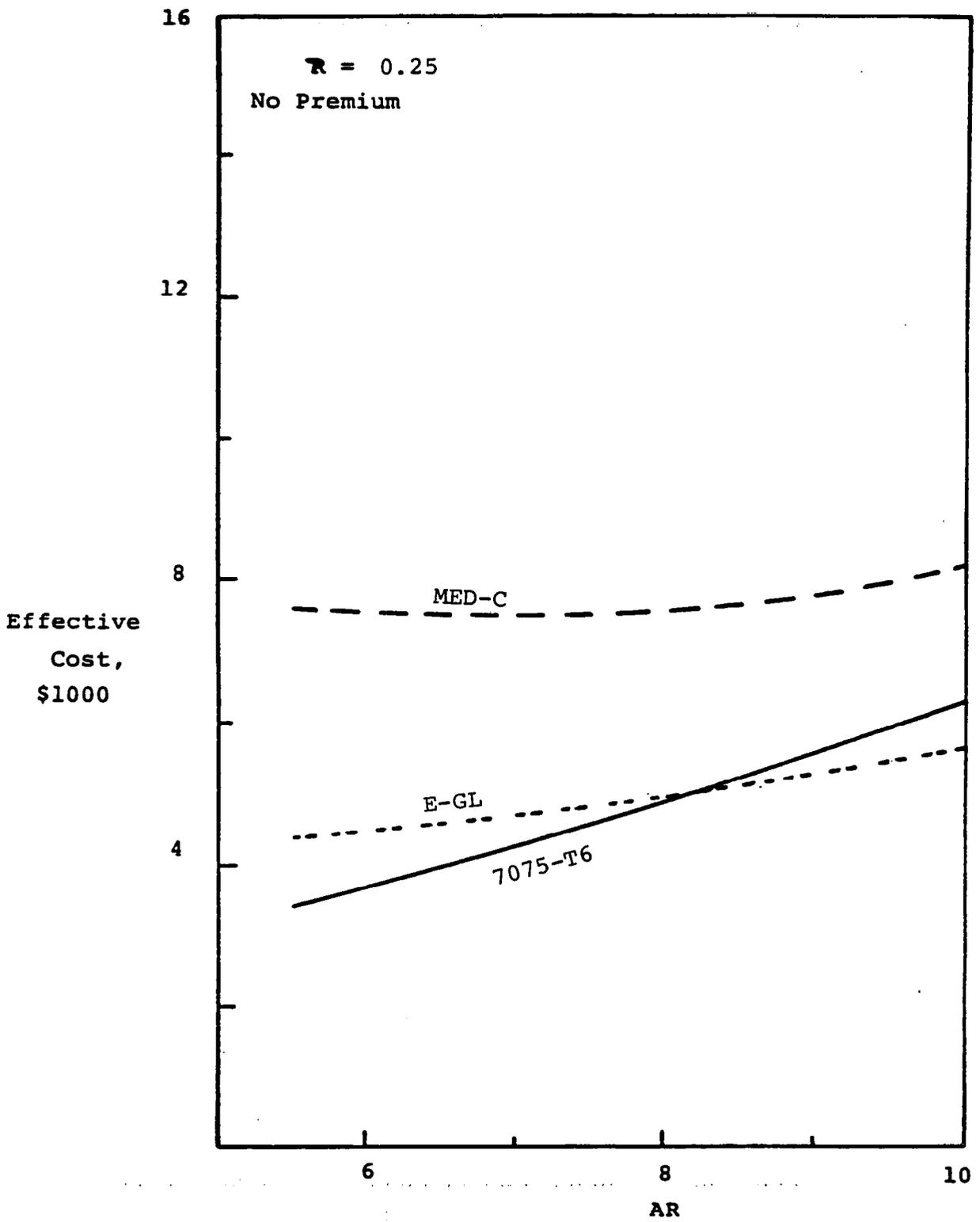
Figure F-17. Manufactured Costs of Wings with Straight Taper and Various Aspect Ratios.

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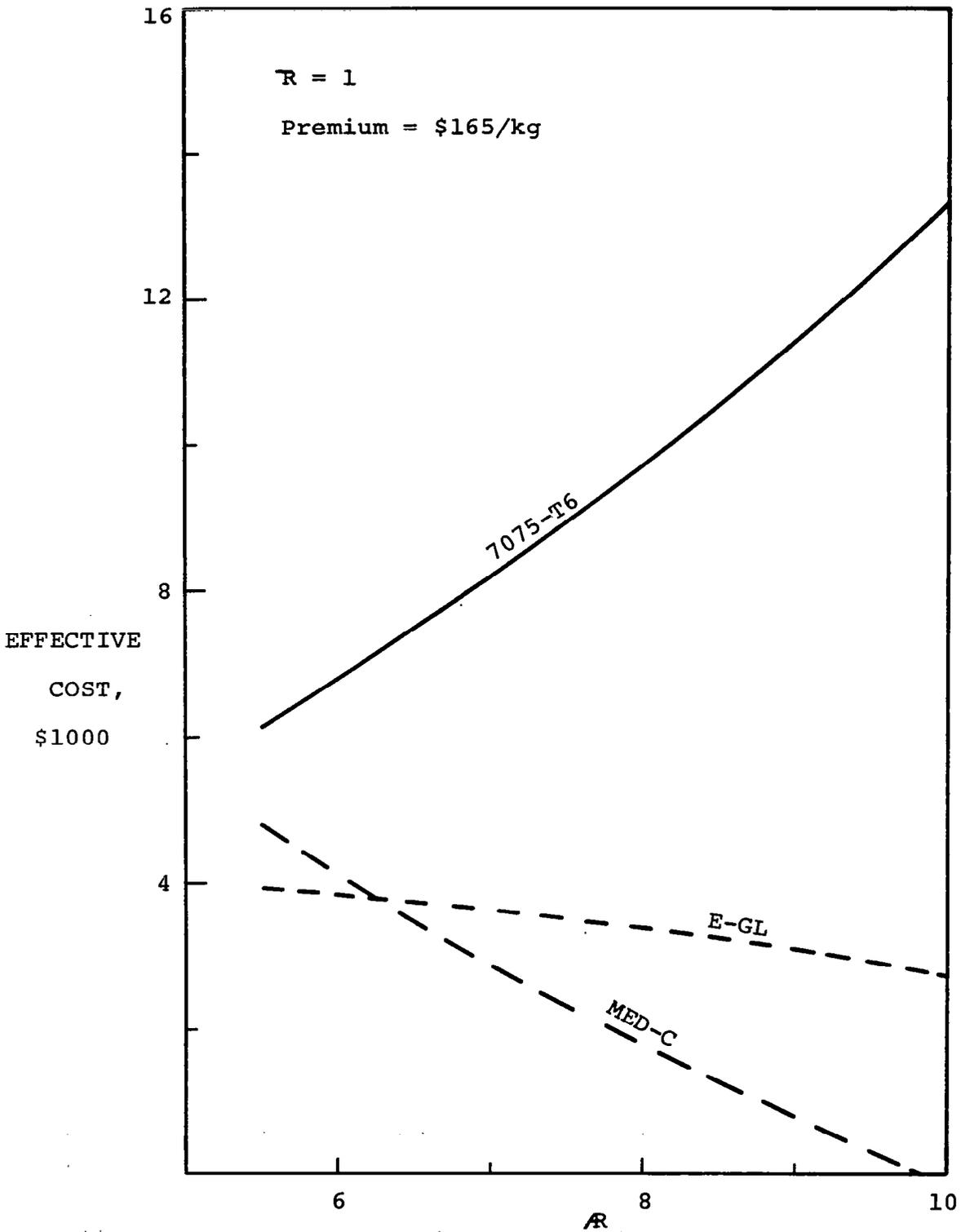
Figure F-18. Effect of $165/kg Premium for Weight Saving on Effective Costs of Various Straight Wings.
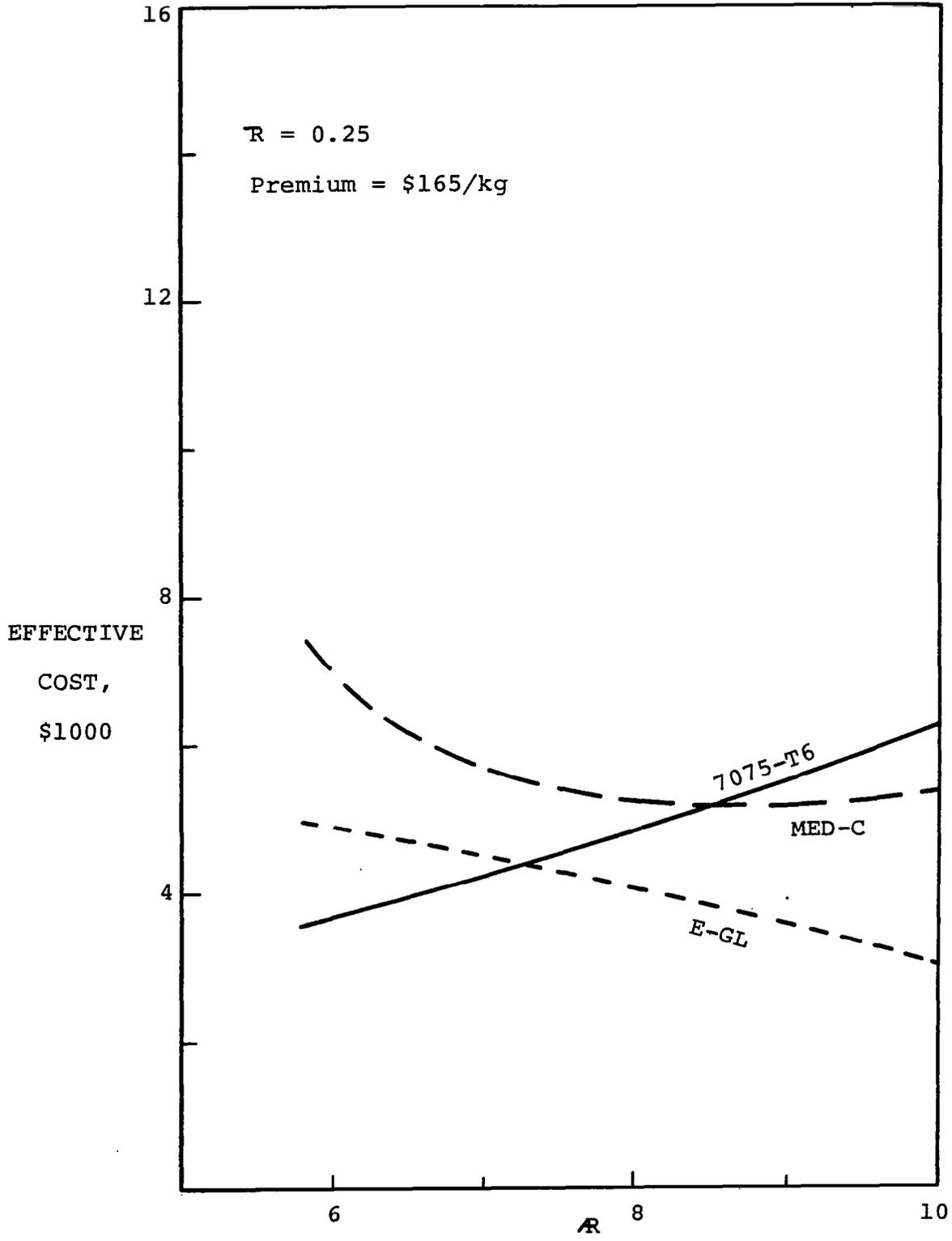
Figure F-19.   Effect of $165/kg Premium for Weight Saving on Effective Costs of Various Wings Having Straight Taper.

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15. Supplementary Notes
Langley technical monitor: Marvin B. Dow
Final Report

16. Abstract
    Guidelines for research on composite materials directed toward the improvement of all aspects of their applicability for general aviation aircraft were developed from extensive studies of their performance, manufacturability, and cost-effectiveness. Specific areas for research and for manufacturing development were identified and evaluated. Inputs developed from visits to manufacturers were used in part to guide these evaluations, particularly in the area of cost-effectiveness. Throughout the emphasis was to direct the research toward the requirements of general aviation aircraft, for which relatively low load intensities are encountered, economy of production is a prime requirement, and yet performance still commands a premium. A number of implications regarding further directions for developments in composites to meet these requirements also emerged from the studies. Chief among these is the need for an integrated (computer program) aerodynamic/structures approach to aircraft design. Such a program can lead to maximum realization of the potentials of advanced composites for general aviation applications.

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