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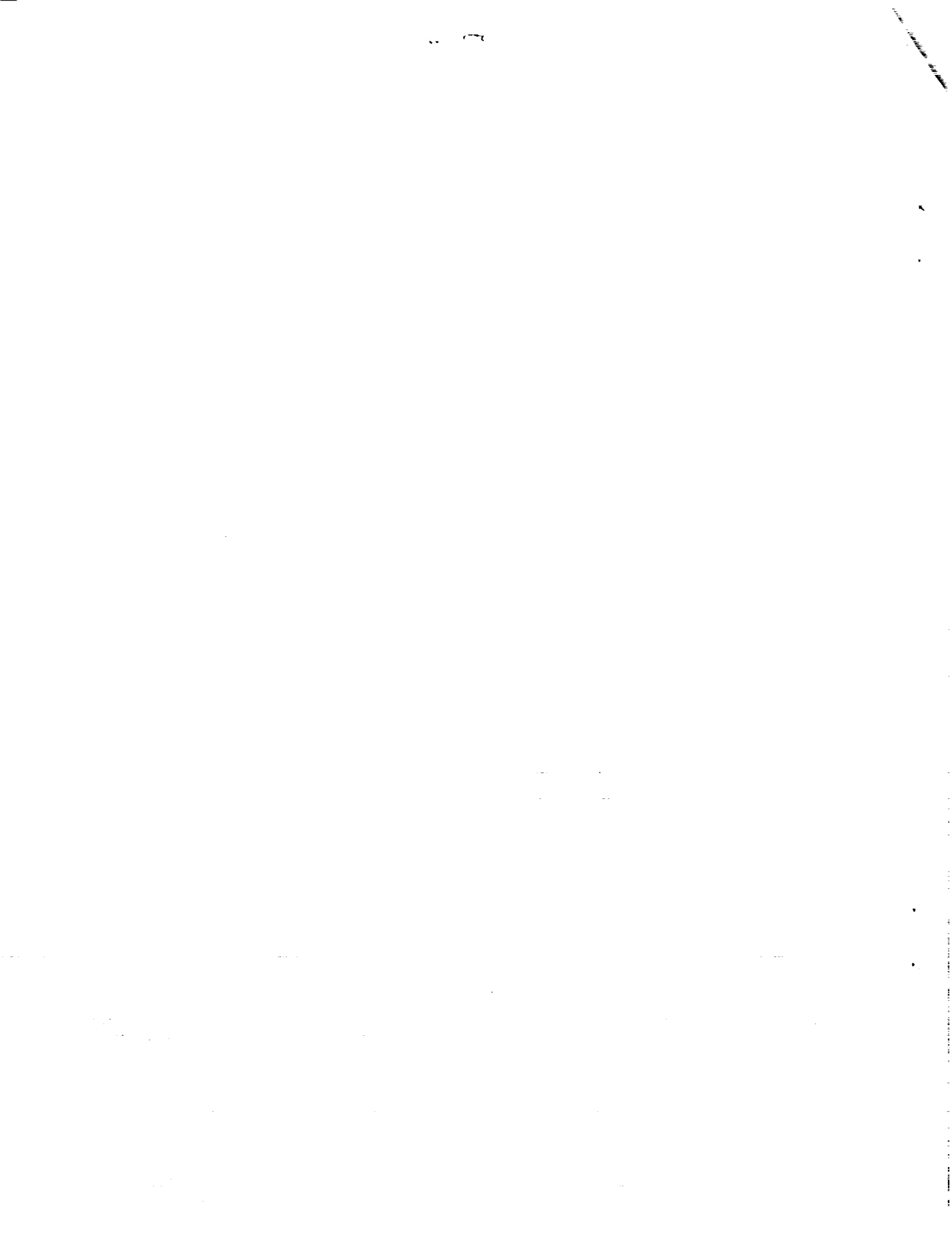
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A (0/90)SUB 2 SYMMETRIC TITANIUM MATRIX
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**STRESS-STRAIN ANALYSIS OF A [0/90]_{2s} TITANIUM MATRIX LAMINATE
SUBJECTED TO A
GENERIC HYPERSONIC FLIGHT PROFILE**

M. Mirdamadi¹, W.S. Johnson²

ABSTRACT

Cross-ply laminate behavior of Ti-15V-3Cr-3Al-3Sn (Ti-15-3) matrix reinforced with continuous silicon-carbide fibers (SCS-6) subjected to a generic hypersonic flight profile was evaluated experimentally and analytically. Thermomechanical fatigue test techniques were developed to conduct a simulation of a generic hypersonic flight profile. A micromechanical analysis was used. The analysis predicts the stress-strain response of the laminate and of the constituents in each ply during thermal and mechanical cycling by using only constituent properties as input. The fiber was modeled as elastic with transverse orthotropic and temperature dependent properties. The matrix was modeled using a thermo-viscoplastic constitutive relation. The fiber transverse modulus was reduced in the analysis to simulate the fiber-matrix interface failure. Excellent correlation was found between measured and predicted laminate stress-strain response due to generic hypersonic flight profile when fiber debonding was modeled.

Keywords: Silicon-carbide fibers, interface, residual stresses, thermal strains, viscoplasticity theory.

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INTRODUCTION

Titanium metal matrix composites, such as Ti-15V-3Cr-3Al-3Sn (Ti-15-3) reinforced with continuous silicon-carbide fibers (SCS-6), are being evaluated for use in hypersonic vehicles where high strength-to-weight and high stiffness-to-weight ratios are critical. This material system has the potential for applications up to 650°C. However, at temperatures above 400°C, titanium exhibits significant viscoplastic behavior. Since the operating temperatures of hypersonic vehicles are well above 400°C, the viscoplastic behavior of the titanium must be accounted for in an analytical evaluation of titanium metal matrix composites (TiMMC).

The objectives of this research are to (1) experimentally determine the stress-strain response of a $[0/90]_{2s}$ SCS-6/Ti-15-3 laminate due to the thermomechanical fatigue (TMF) that will occur during hypersonic flight profile testing and (2) use an analytical method to predict the measured laminate stress-strain response. The analysis will also include the fiber-matrix interface failure.

Recently, Mirdamadi, et al. [1], used the analysis to predict the stress-strain response of unidirectional SCS-6/Ti-15-3 laminates subjected to simple in-phase and out-of-phase TMF loadings. Good agreement between experiment and prediction was found. This paper will extend the previous work to a more complex laminate, $[0/90]_{2s}$, with a more complicated TMF loading history.

MATERIAL AND TEST SPECIMENS

A $[0/90]_{2s}$ SCS-6/Ti-15-3 laminate with a fiber volume fraction of 0.385 and a thickness of 1.68-mm was used in the present study. The SCS-6 fibers are continuous silicon-carbide fibers having a 0.140-mm diameter. The composite laminates were made by hot-pressing Ti-15-3 foil between tapes of unidirectional SCS-6 silicon-carbide fibers held in place with molybdenum wires. The temperature-dependent elastic fiber properties used in this study are shown in Table I. The Ti-15-3 matrix material is a metastable beta titanium alloy. Long exposures at elevated temperatures can lead to the precipitation of an α -phase which may alter the macroscopic mechanical behavior of the Ti-15-3 [2]. Therefore, the composite in the present study was heat treated at 650°C for one hour in air followed by an air quench to stabilize the matrix material. This heat treatment was the same heat treatment used by Pollock and Johnson [3]. The temperature-dependent Ti-15-3 matrix properties in the aged condition are given in Table II for room temperature, 316°C, 482°C, 566°C, and 650°C [1].

Additional properties were determined at 427°C in the aged condition in the current work. It was assumed that the matrix properties remained the same from room temperature to 150°C.

Two TMF spectrum tests were conducted on straight rectangular specimens, 152-mm x 12.7 x 1.68-mm, cut using a diamond wheel saw. Brass tabs (10-mm x 30-mm x 1-mm) were placed between the end of the specimen and the grips to avoid specimen failure in the serrated grips. The brass tabs were not bonded to the specimens but were held in place by the grips.

EXPERIMENTAL PROCEDURE

A TMF test capability was developed to conduct hypersonic flight profiles. The temperature and the load spectrum of a generic hypersonic mission flight profile are shown in Figure 1. The letters shown in Figure 1 will be used later for comparison with stress-strain results. As shown in the figure, the flight profile consists of both isothermal and non isothermal load cycling at 1 Hz with hold-times at different temperatures. The thermal loading rates during heating and cooling are 2.8 and 1.4°C/sec, respectively.

A schematic diagram of the TMF test setup is shown in Figure 2. The TMF test setup contains a 100-kN servo-hydraulic test frame with water-cooled grips, a load profiler, a 5-kW induction generator controlled by a temperature profiler, and a nitrogen supply tank. The load and temperature spectrum are independently controlled by a load and a temperature profiler. The temperature profiler was modified to accept a command signal from the load profiler to initiate temperature spectrum at any desired point in the load profile. The nitrogen supply tank was designed to supply gaseous and liquid nitrogen simultaneously. The flow of gaseous and liquid nitrogen was controlled by the temperature profiler. Gaseous nitrogen was used in the temperature range of 427°C to 316°C and liquid and gaseous nitrogen were used in the temperature of 316°C to room temperature. The use of gaseous and liquid nitrogen with induction heating was necessary to maintain the desired cooling rate of 1.4°C/sec.

A 50-mm length in the center of the specimen was subjected to the thermal loading using three independent adjustable induction coils such as developed by Ellis and co-workers [4] at NASA Lewis Research Center. A temperature distribution at 650°C of $\pm 10^\circ\text{C}$ was achieved by adjusting the induction coils. The induction coils were adjusted by monitoring five K-type thermocouples spot welded in the gage section of the specimen. An infrared

camera was then used to monitor the temperature distribution in subsequent tests. Axial strains were measured on the edge of the specimen using a high temperature quartz rod water-cooled extensometer with a 25-mm gage length. An eight channel analog/digital PC based data acquisition system was used to record and store the test data. The analog/digital data acquisition system was developed by George Hartman, University of Dayton, Ohio, and is capable of recording load, strain, time, and temperature data at predetermined levels. Prior to the flight profile test, the specimen was subjected to the temperature profile alone to ensure thermal stability and synchronization with the load profiler command.

ANALYTICAL METHOD

The stress-strain response of the $[0/90]_{2s}$ laminate was predicted using a micromechanics analysis. The VISCOPLY code, developed by Bahei-El-Din, is based on constituent properties. The program uses the vanishing fiber diameter (VFD) model [5] to calculate the orthotropic properties of a ply. The ply properties are then used in a laminated plate analysis [6] to predict the overall laminate stress-strain response. Both the fiber and the matrix can be described as a thermo-viscoplastic material. Combinations of thermal and mechanical loads can be modeled. Sequential jobs can be run for varying order and rate of load and temperature. Fiber and matrix average stresses and strains and the overall composite response under thermomechanical loading conditions are calculated. The VISCOPLY program, due to the nature of the VFD model, does not account for the local stress concentrations around the fiber nor for the lateral constraint in a ply due to the presence of the fiber. However, based upon the second author's experience using the VFD model for over 12 years on numerous metal matrix composites, the model predicts overall laminate response very well, usually within 10 percent of the experimental results.

The viscoplastic theory used in the VISCOPLY program was developed by Bahei-El-Din [7] for high temperature, nonisothermal applications and was based on the viscoplasticity theory of Eisenberg and Yen [8]. The theory used in the VISCOPLY program assumes the existence of an equilibrium stress-strain curve which corresponds to the theoretical lower bound of the dynamic response. The theory further assumes that the elastic response is rate independent and that inelastic rate-dependent deformation takes place if the current stress state is greater than the equilibrium stress. The inelastic strain rate is described using a power law function of the overstress, H , which is defined as the difference between the current rate-dependent stress and the equilibrium stress.

The total strain rate, $\dot{\epsilon}^t$, is the sum of the elastic strain rate, $\dot{\epsilon}^e$, the thermal strain rate, $\dot{\epsilon}^T$, and the inelastic strain rate, $\dot{\epsilon}^{in}$. Under uniaxial loading, the total strain rate is defined as

$$\dot{\epsilon}^t = \dot{\epsilon}^e + \dot{\epsilon}^T + \dot{\epsilon}^{in} = \frac{\dot{\sigma}}{E(T)} + m(T)\dot{T} + k(T)H^p(T) \quad (1)$$

where $m(T) = \alpha(T) - \left(\frac{\sigma}{E^2(T)}\right) \frac{dE(T)}{dT}$, and

$$H = \sigma - \sigma^*$$

E is Young's elastic modulus, T is the temperature, $\alpha(T)$ is the coefficient of thermal expansion, and σ^* is the equilibrium stress defined by the quasi-static stress-strain curve. The viscoplastic parameters k and p in equation (1) must be determined experimentally and are temperature dependent. These parameters have been determined for the Ti-15-3 matrix [1]. The fiber was assumed to behave elastically with the temperature dependent material properties given in Table I. Although not used in the current work, the program has the capability to model the fiber as a viscoplastic material with transverse orthotropic properties.

A simple procedure was used to analytically simulate the fiber-matrix interface failure known to occur in the SCS-6/Ti-15-3 material. In room temperature fatigue tests [9], a distinct knee was observed in the stress-strain response at stress levels well below the yield stress of the matrix material. In the first cycle, this knee was found to correspond to the stress required to overcome the thermal residual stresses and fail the fiber-matrix interface in the off-axis plies. In subsequent fatigue cycles, the knee was observed at a lower stress level, the stress required to overcome the thermal residual stresses in the matrix. To simulate the fiber-matrix interfacial failure, the transverse modulus of the fibers in the 90° plies was reduced for stress levels above the stress level corresponding to the observed knee in the stress-strain response of the [0/90]_{2s} laminate at room temperature. In elevated temperature fatigue tests, however, no knee was apparent in the stress-strain response [3] and it was assumed that fiber-matrix interfacial failure occurred upon loading. Thus, the fiber transverse modulus in the 90° plies was reduced at the start of loading for temperatures above 400°C.

RESULTS AND DISCUSSION

In this section, the experimental and analytical results are presented. The isothermal stress-strain response of the $[0/90]_{2s}$ laminates is analyzed to assess the effects of fiber-matrix separation and loading rates. The experimental results and the theoretical predictions for the flight profile are presented.

Isothermal Laminate Behavior

It was first necessary to determine the appropriate reduction of the transverse modulus of the fibers in the 90° plies to simulate the fiber-matrix interface failure. The experimental and predicted stress-strain response of the SCS-6/Ti-15-3 $[0/90]_{2s}$ laminate at room temperature (stress rate of $\dot{S}=840$ MPa/s) and 427°C (stress rate of $\dot{S}=1250$ MPa/s) are shown in Figures 3 and 4, respectively. The VISCOPLY predictions are shown for various ratios of the fiber transverse modulus to the fiber axial modulus (E_t^f / E_a^f) in the 90° ply ranging from 1.0 to 0.001. At room temperature (Figure 3), the fiber-matrix interface failure occurred at a stress level of 70 MPa. This was determined from the knee in the experimental stress-strain curve. However, at 427°C , it was assumed that the fiber-matrix interface failed instantly upon loading. Reducing the 90° fiber transverse modulus by a factor of 0.1 produced very good predictions at 427°C . The room temperature predictions for $E_t^f / E_a^f=0.1$ were also fairly good. Therefore, a reduction factor of 0.1 was used to model fiber-matrix interface failure at all temperatures. This reduction factor may be dependent on temperature, fiber volume fraction, processing parameters, fiber-matrix interface strength, and, of course, fiber and matrix properties.

Next, the effect of loading rate on the predictions was examined. Figure 5 shows the composite experimental stress-strain response during the second cycle (i.e., subsequent to the fiber-matrix interface failure of the 90° plies) at a stress rate of 10 MPa/sec at 650°C [3]. Included in the figure are the VISCOPLY predictions with fiber-matrix interface failure of the 90° plies ($E_t^f / E_a^f=0.1$). The VISCOPLY prediction at a rate of 900 MPa/sec is also shown. The 900 MPa/sec rate corresponds to the loading rate used in the hypersonic flight profile. As seen in the figure VISCOPLY accurately predicted the initial elastic modulus but was somewhat less accurate at higher stress levels. The predicted maximum strain was 7% smaller than observed experimentally. The VISCOPLY prediction at the rate of 900

MPa/sec resulted in a nearly linear stress-strain response which demonstrates the effect of the matrix rate-dependent behavior on composite stress-strain response.

Flight Profile Behavior

In this section, the stress-strain response of the laminate subjected to the flight profile shown in Figure 1 will be analyzed and compared to experimental results. Predictions will be made assuming perfect bonding of fiber-matrix in the 90° plies and assuming failure of the 90° fiber-matrix interfaces. For clarity, during the rapid cycling segments of the flight profile (e.g., segments B, F, I and J in Figure 1), only the first loading and last unloading cycle will be shown in the figures.

A test was conducted by applying only the thermal history of the flight profile shown in Figure 1. The measured thermal strains and the VISCOPLY predictions are shown in Figure 6. The measured thermal strains match the applied temperature profile previously shown in Figure 1 indicating excellent control of the heating and cooling rates. The thermal strain of the laminate was accurately predicted by VISCOPLY.

Specimens were then subjected to the full thermal and mechanical flight profile shown in Figure 1 at two stress levels. In one test, 100% load equaled 420 MPa; in the second test 100% load equaled 620 MPa. The stress-strain responses of the fifth repetition of the flight profile are shown in Figures 7 and 8 for the two stress levels. The letters placed at various locations on the stress-strain response can be referenced back to Figure 1 to find the associated point in the flight profile. The horizontal portions of the predictions and the experimental data indicate an increase in strain due solely to temperature changes while the mechanical loads were held constant. The VISCOPLY predictions assume perfect fiber-matrix interface bonding. As seen in the figures, VISCOPLY predicted a stiffer response than was observed experimentally. The predictions of the cyclic loads shown in locations F and I appear broad because the temperature was changing. The experimental and predicted creep strain during hold period at H was very small. Predictions of the composite response under the flight profile made with simulated interface failure of the 90° plies are shown in Figures 9 and 10. The predictions agree much better with the experimental behavior when the interface failure is modeled as previously discussed.

If the fatigue behavior of the laminate is to be well understood and prediction methodology developed, the behavior of the composite constituents must be understood. Previous work by Johnson, et al. [9] showed good correlation between the stress range in the 0° fiber and the number of cycles to failure of the laminate at room temperature. More recently,

Mirdamadi, et al. [1] used the 0° fiber stress range calculated from a micromechanics analysis to compare the TMF data of Castelli, et al. [10], Gabb, et al. [11], and the isothermal fatigue data of Pollock and Johnson [3]. They determined that for a given condition, the fatigue strength of the 0° fiber was controlled by a combination of temperature, loading frequency, and time at temperature. Furthermore, for a given temperature, loading frequency, and time at temperature, the stress range in the 0° fiber controlled the fatigue life. Bigelow and Johnson [12] and Bakuckas, Johnson, and Bigelow [13] accurately predicted the static strength of virgin specimens and fatigued specimens by monitoring the 0° fiber stress. Therefore, the 0° fiber stress (or strain) plays a major role in the static and fatigue strength of TiMMC. Under isothermal loading conditions, the 0° fiber strain is equivalent to the overall composite axial strain. However, under non-isothermal loading conditions, where the load and the temperature are cycled, determination of the 0° fiber stress is not straight forward and micromechanics based models are required to predict the 0° fiber stress. Figures 11 and 12 show the VISCOPLY predictions of the 0° fiber stress as a function of time during the flight profile for the two stress levels. These predictions were made assuming fiber-matrix interface failure in the 90° plies. Therefore, such predictions are important when analyzing the fatigue behavior of the composite and could be used in a failure criteria.

CONCLUSIONS

A TMF test capability was developed to conduct a simulation of a generic hypersonic flight profile. The VISCOPLY analysis was used to analyze the stress-strain response of the $[0/90]_{2s}$ SCS-6/Ti-15-3 laminate subjected to the flight profile. The following conclusions were made:

- o Combining nitrogen cooling with the induction heating gave excellent cooling control.
- o In this material system, fiber-matrix interface failure must be modeled for accurate predictions. Fiber-matrix interface failure was modeled in VISCOPLY program by dividing the transverse modulus of the fibers in the 90° plies by a factor of 10.
- o The mechanical response of these composites are rate-dependent at elevated temperatures. The VISCOPLY analysis can predict such dependence.

- VISCOPLY accurately predicted the composite stress-strain response for a generic hypersonic flight profile.

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Table I. Temperature-Dependent Properties of the SCS-6 Fiber.

Temperature (°C)	Elastic modulus E (GPa)	Poisson's ratio ν	CTE x 10 ⁶ α (mm/mm/°C)
21.10	393.0	0.25	3.560
93.30	390.0	0.25	3.564
204.44	386.0	0.25	3.618
315.56	382.0	0.25	3.726
426.27	378.0	0.25	3.906
537.78	374.0	0.25	4.068
648.89	370.0	0.25	4.266
760.0	365.0	0.25	4.410
871.0	361.0	0.25	4.572

Table II (a). Temperature-Dependent Properties of the Ti-15-3 Matrix [1].

Temperature (°C)	Elastic modulus E(GPa)	Poisson's ratio ν	CTE x 10 ⁶ α (mm/mm/°C)
25	91.80	0.36	8.48
150	91.80	0.36	8.48
316	80.44	0.36	9.16
427	74.52	0.36	9.40
482	72.24	0.36	9.71
566	64.40	0.36	9.98
650	53.00	0.36	10.26

Table II (b). Temperature-Dependent Properties of the Ti-15-3 Matrix [1].

Yield stress Y (MPa)	Equilibrium(+) curve constants		Inelastic strain (++) rate parameters	
	Λ^* (MPa)	n^*	k (MPa ^{-p})/s	p
772.00	1180.87	0.05	3.37E-13	4.37
772.00	1180.87	0.05	3.37E-13	4.37
492.00	769.00	0.05	3.37E-13	4.37
387.00	528.00	0.04	5.4E-18	6.25
142.00	273.00	0.08	4.53E-09	5.68
20.48	116.81	0.18	5.53E-17	5.35
10.20	36.33	0.14	1.17E-14	4.72

(+) Equilibrium stress-inelastic strain curve, $\sigma^* = \Lambda^*(\epsilon^{in})^{n^*}$

(++) Constitutive equation, $\dot{\epsilon}^{in} = kH^p$

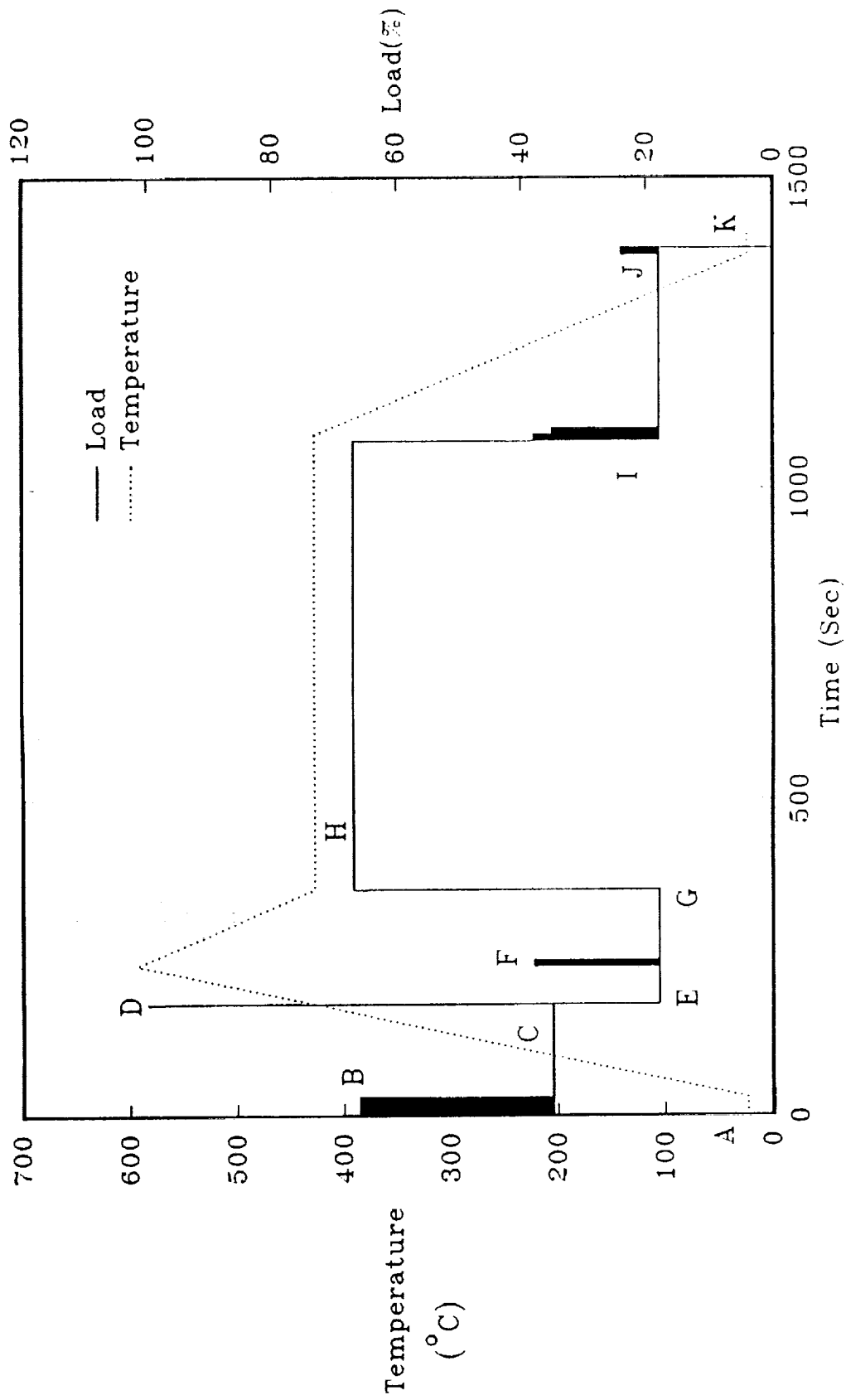


Figure 1. Generic hypersonic flight profile.

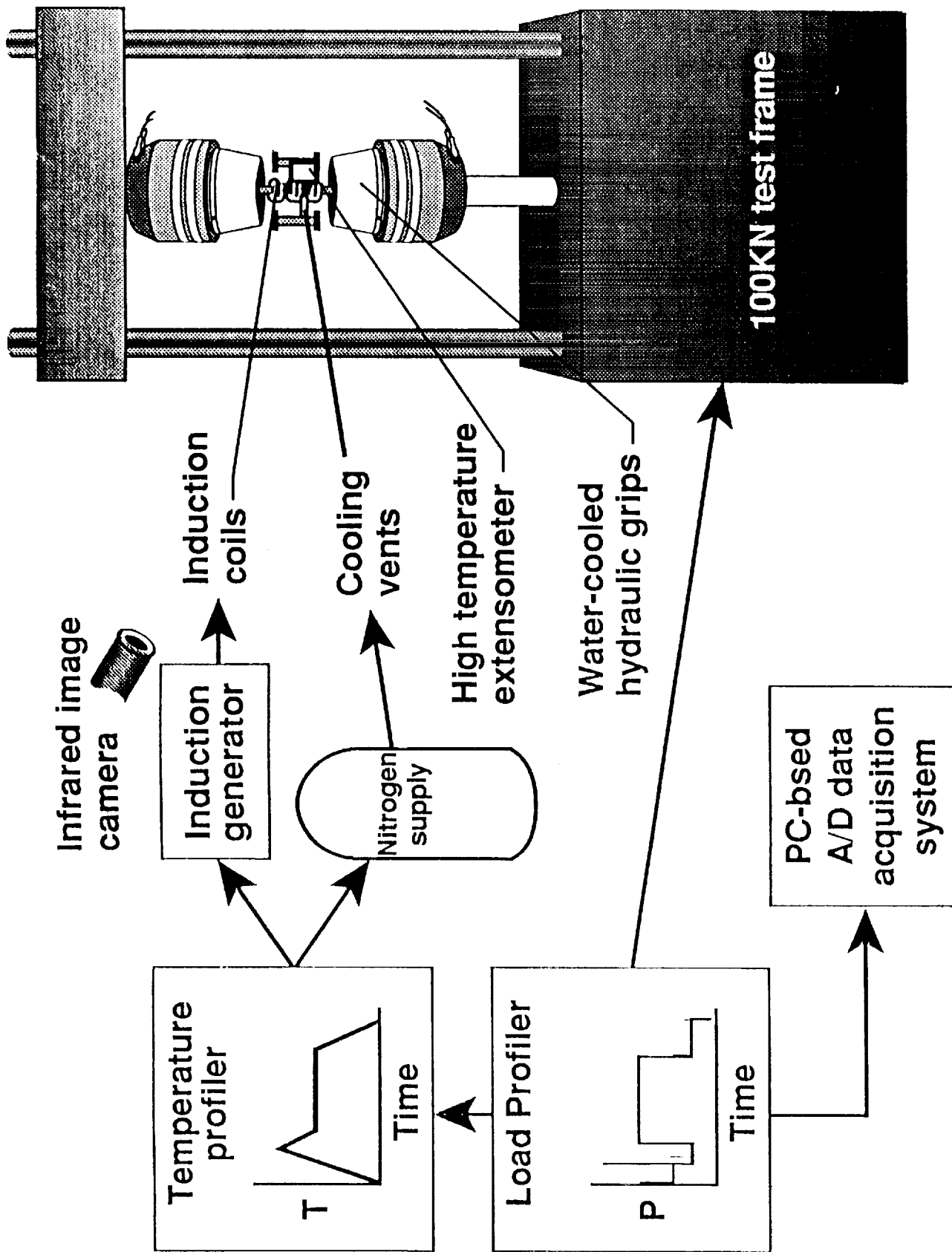


Figure 2. Thermomechanical test setup.

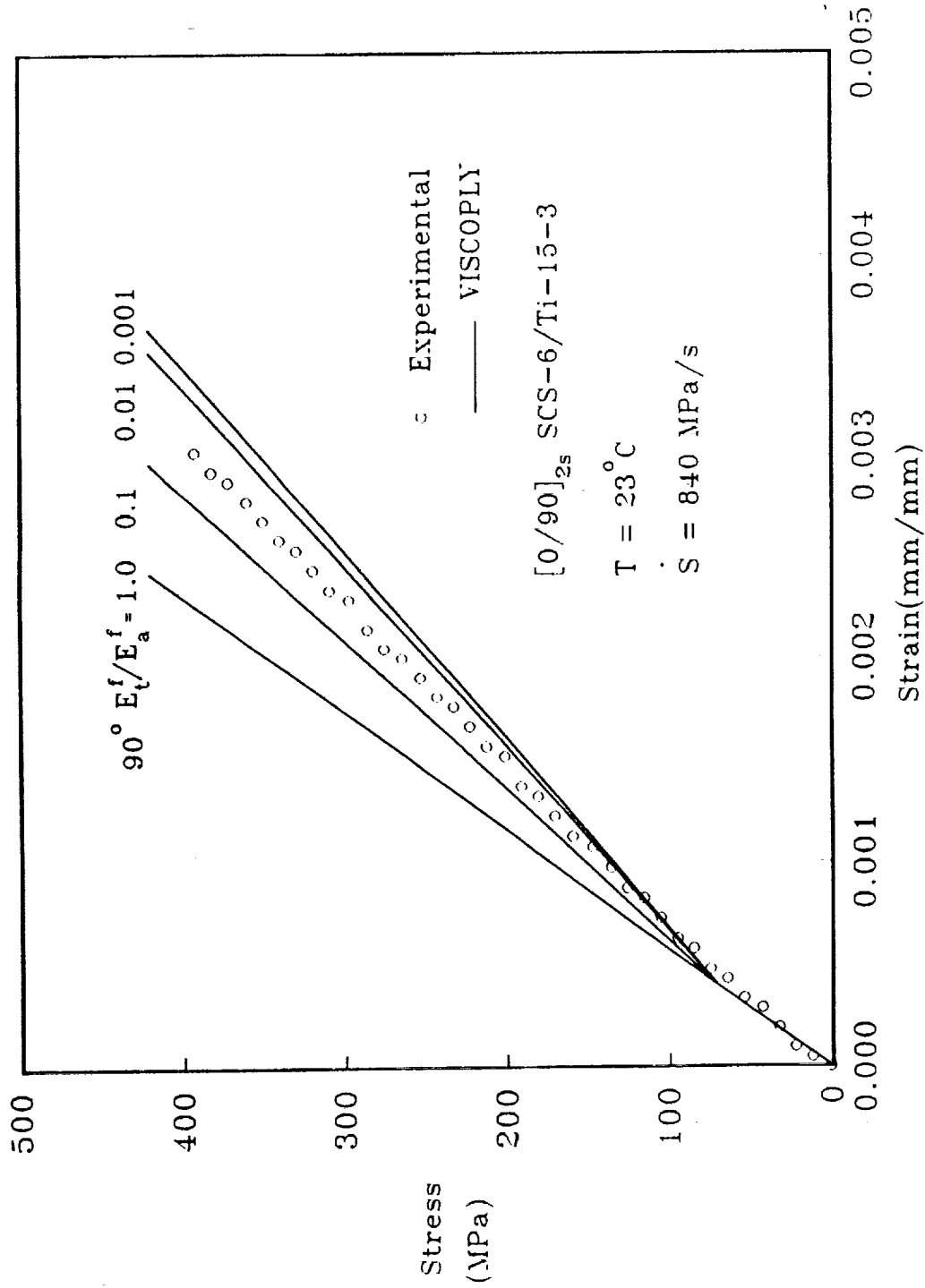


Figure 3. Effect of reducing 90° fiber transverse modulus on VISCOPLY predictions at room temperature.

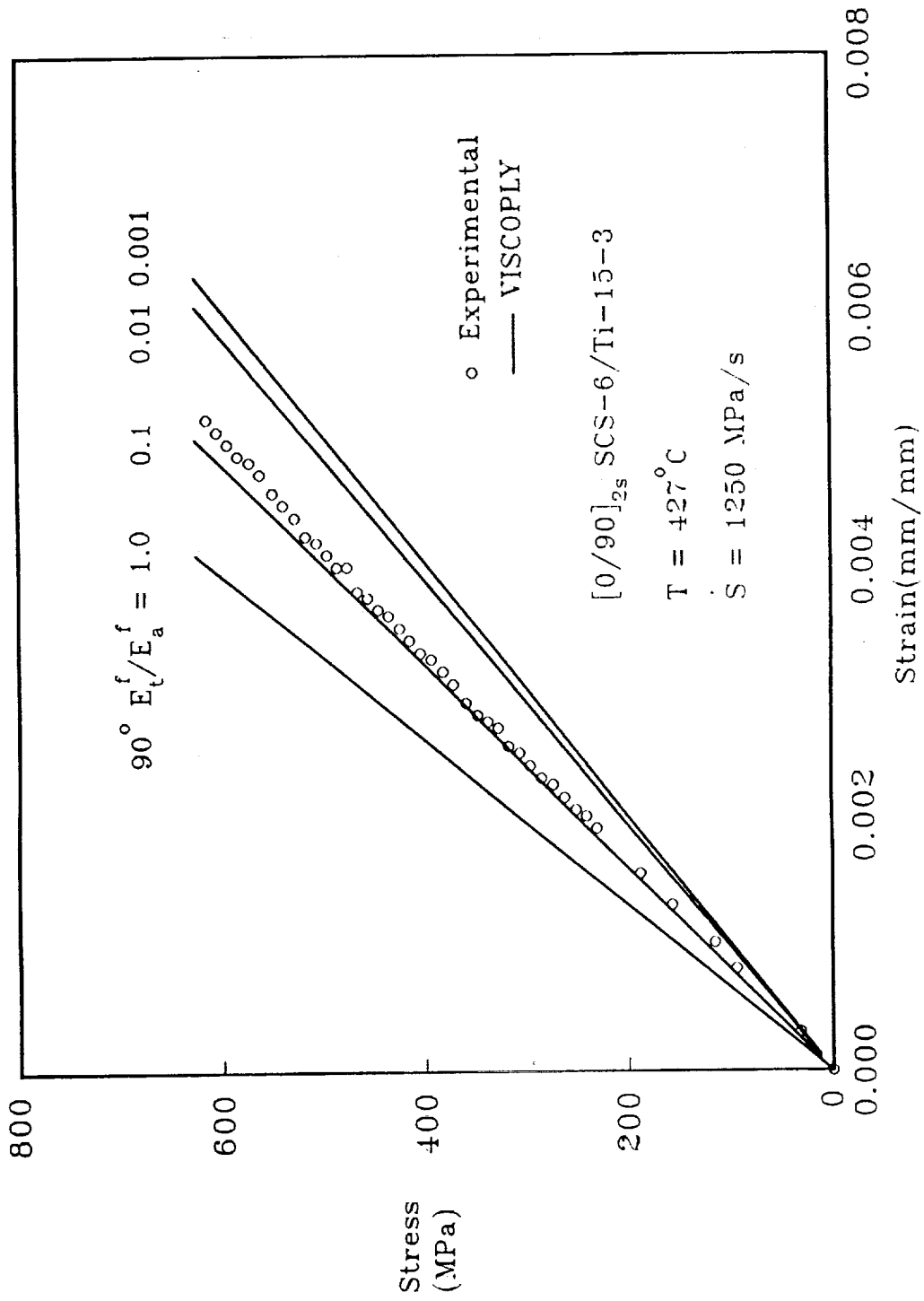


Figure 4. Effect of reducing 90° fiber transverse modulus on VISCOPLY predictions at 427°C .

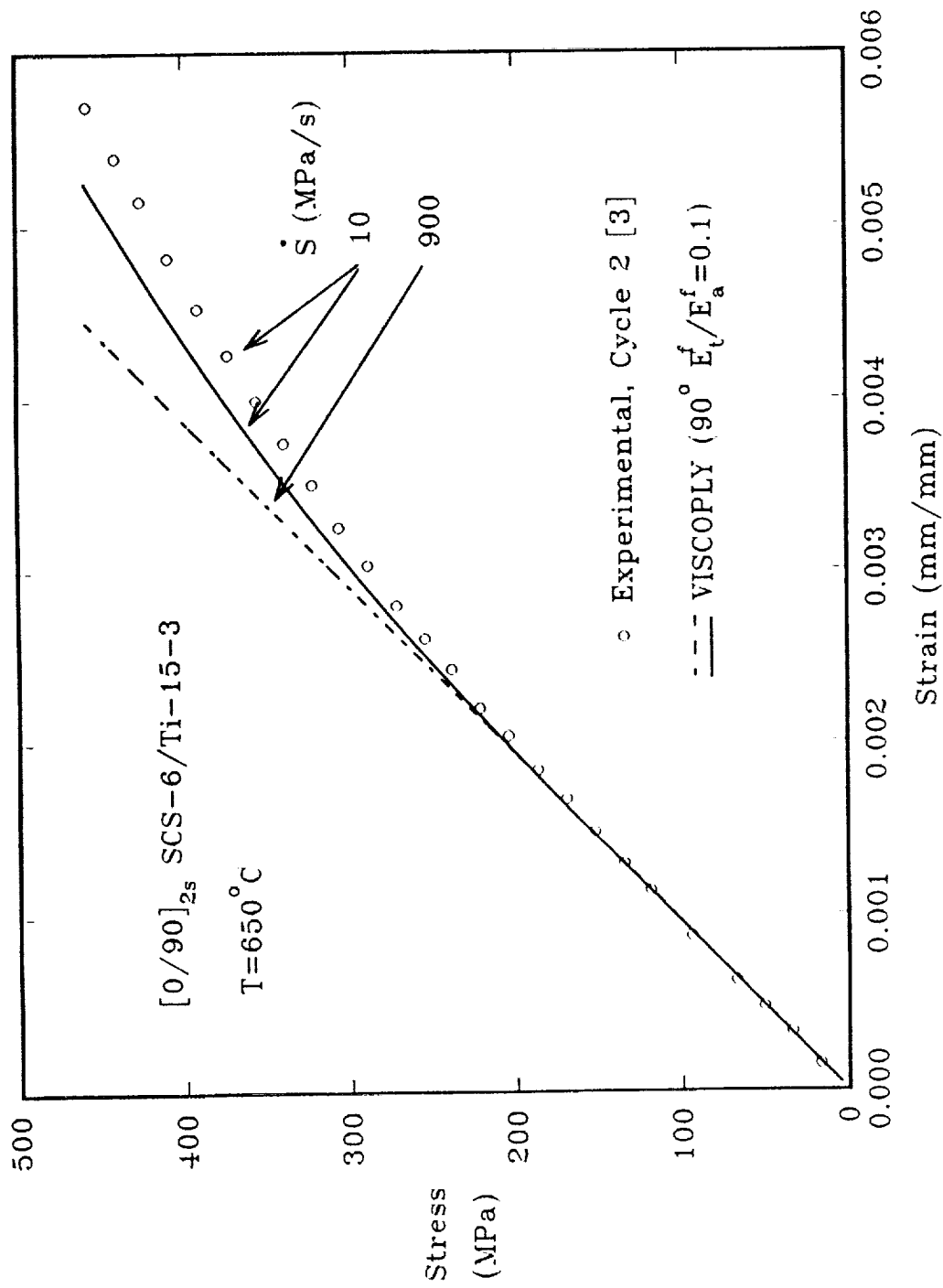


Figure 5. Prediction of stress-strain response of [0/90]_{2s} SCS-6/Ti-15-3 laminate at 650°C.

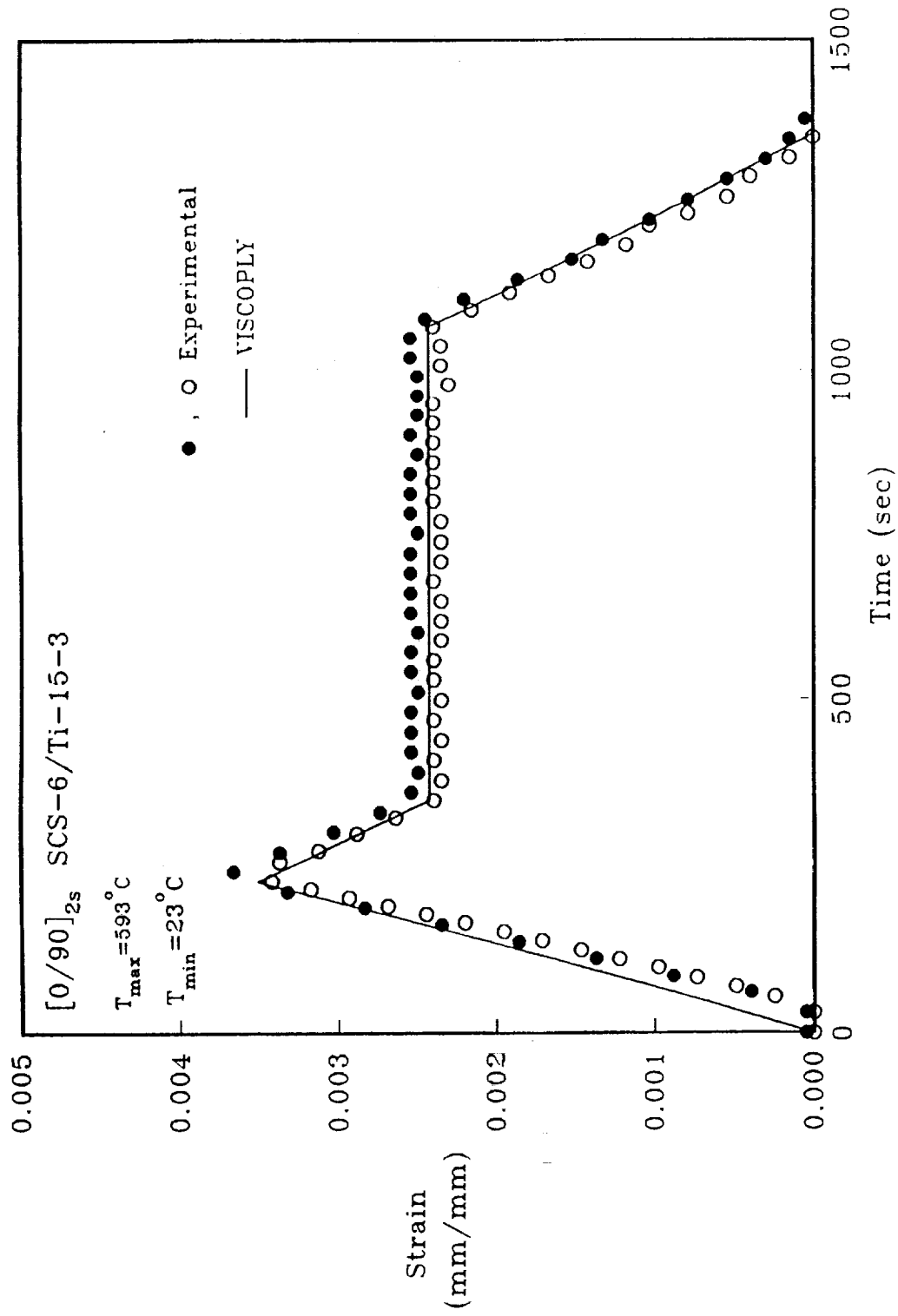


Figure 6. Predicted and experimental thermal strains as a function of time.

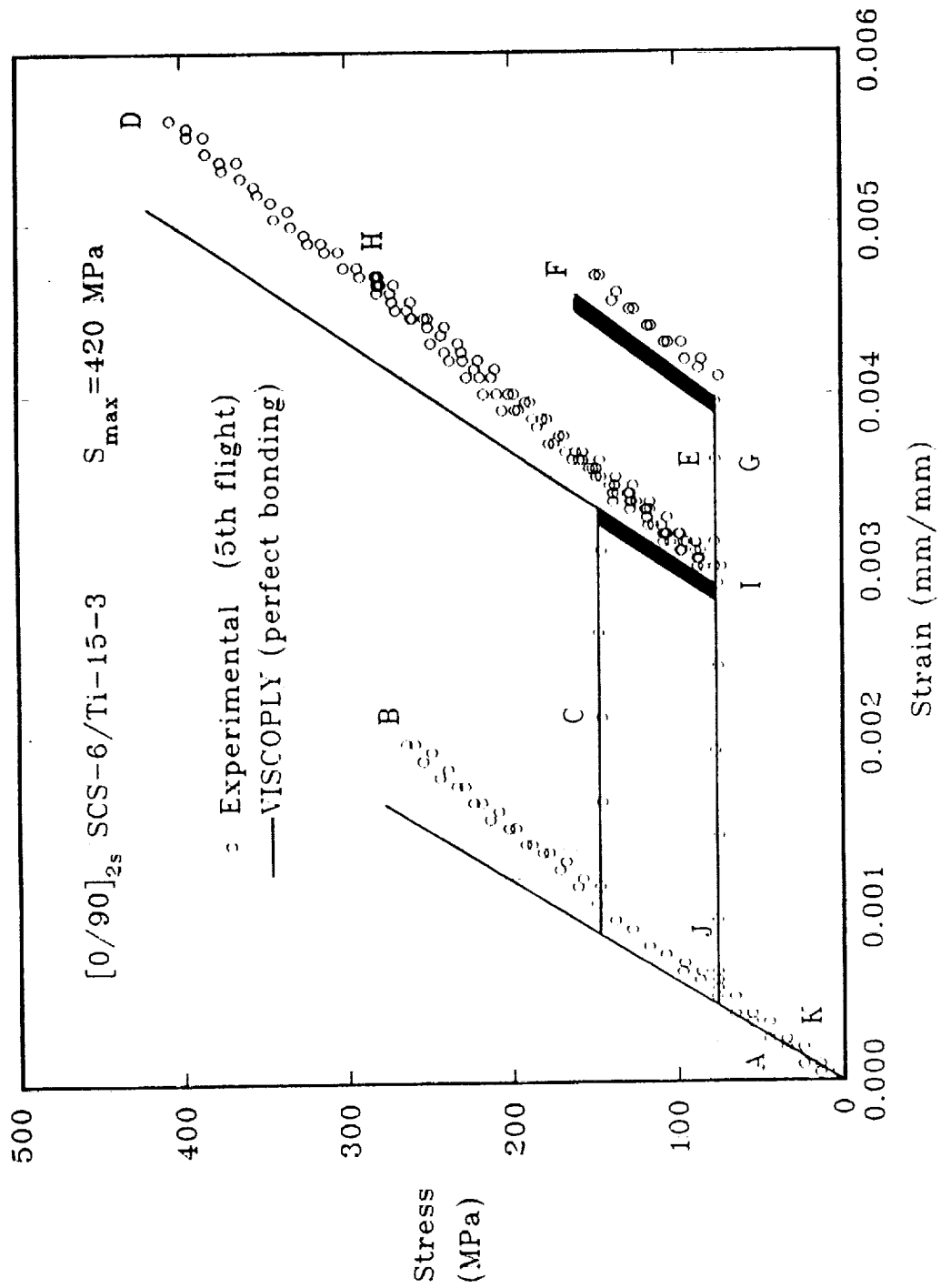


Figure 7. VISOPLY prediction of composite response to the flight profile at the maximum applied stress of 420 MPa assuming perfect bonding.

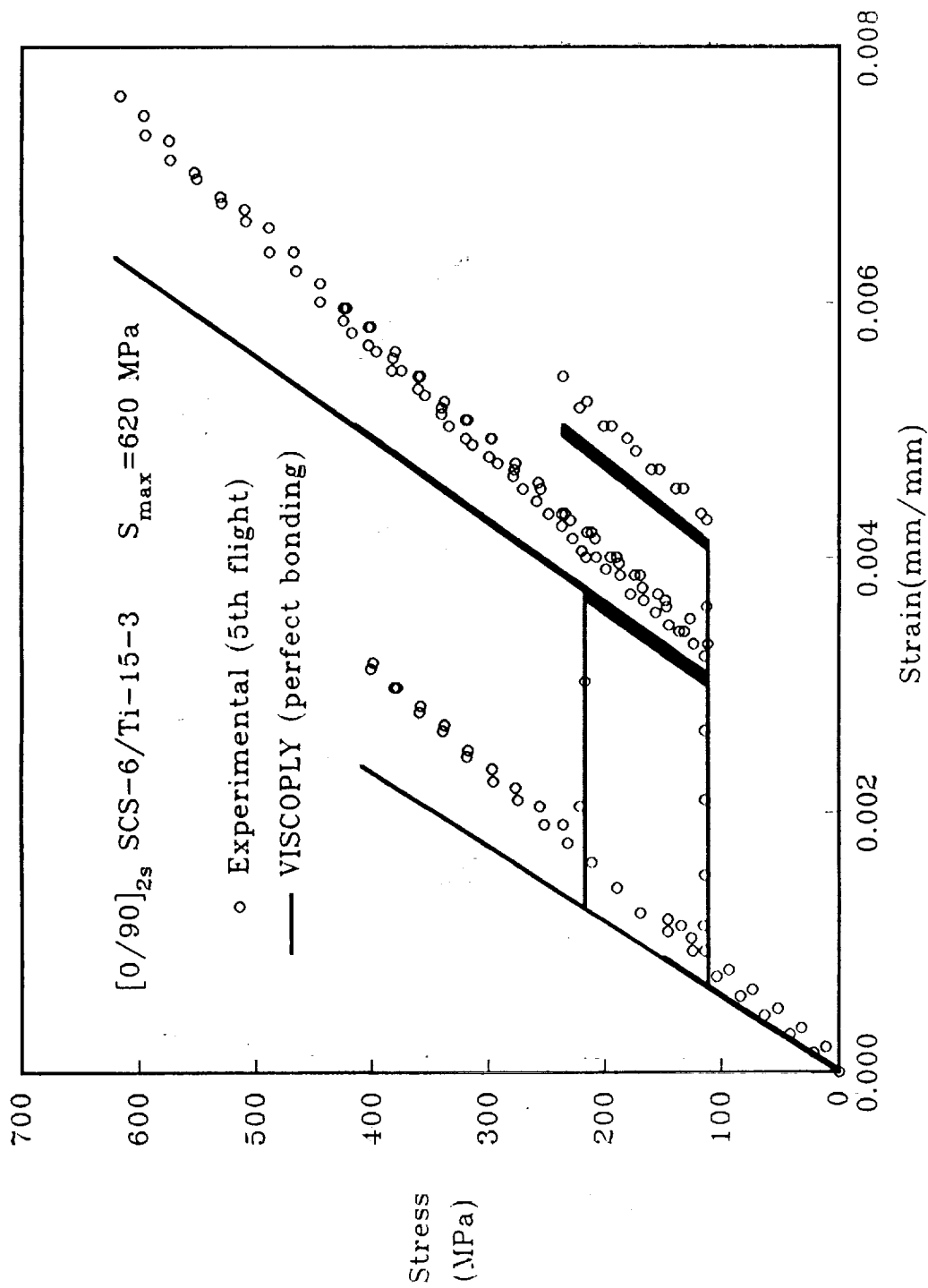


Figure 8. VISCOPLY prediction of composite response to the flight profile at the maximum applied stress of 620 MPa assuming perfect bonding.

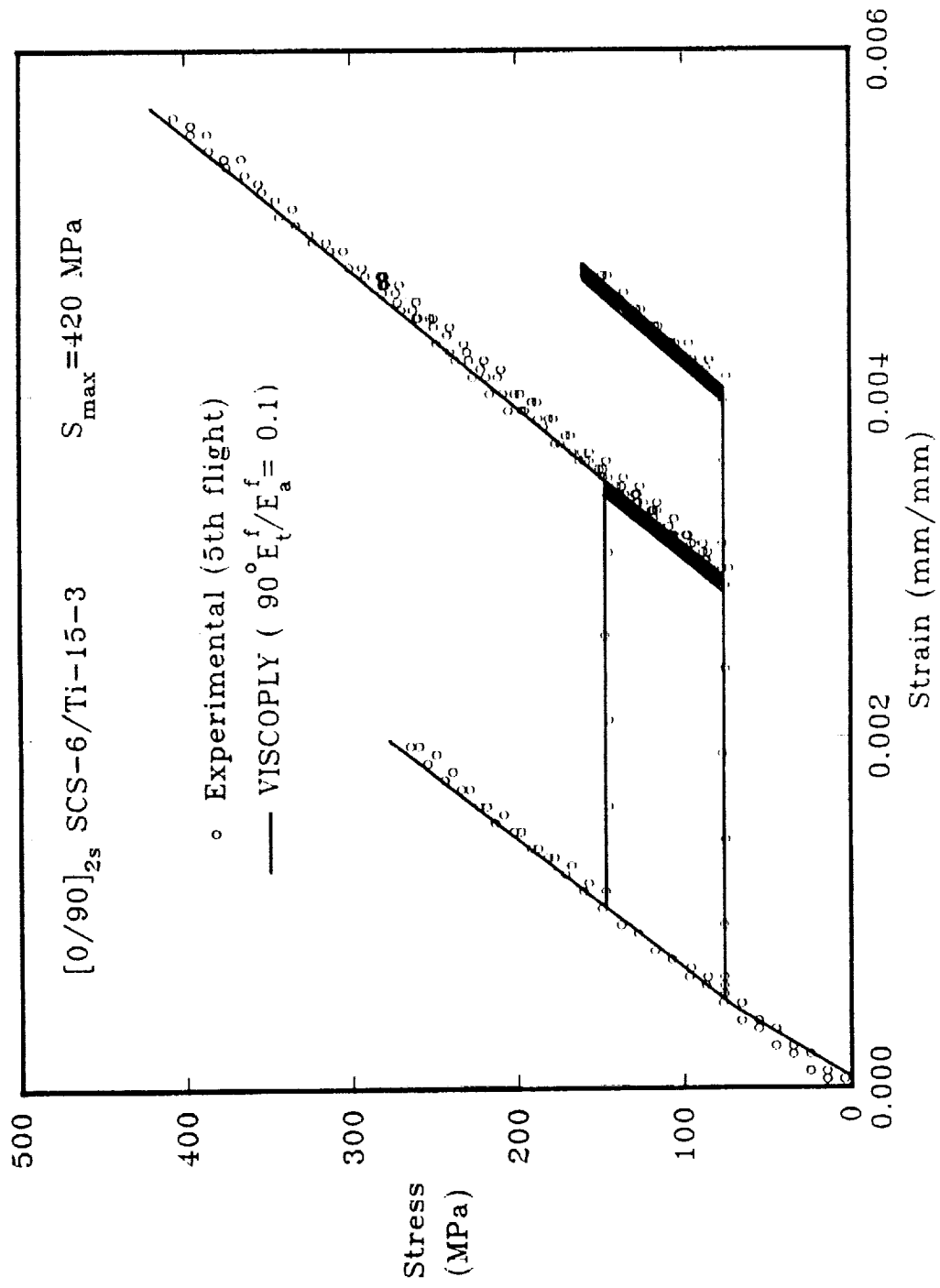


Figure 9. VISOPLY prediction of composite response to the flight profile at the maximum applied stress of 420 MPa simulating fiber-matrix interface failure of 90° plies.

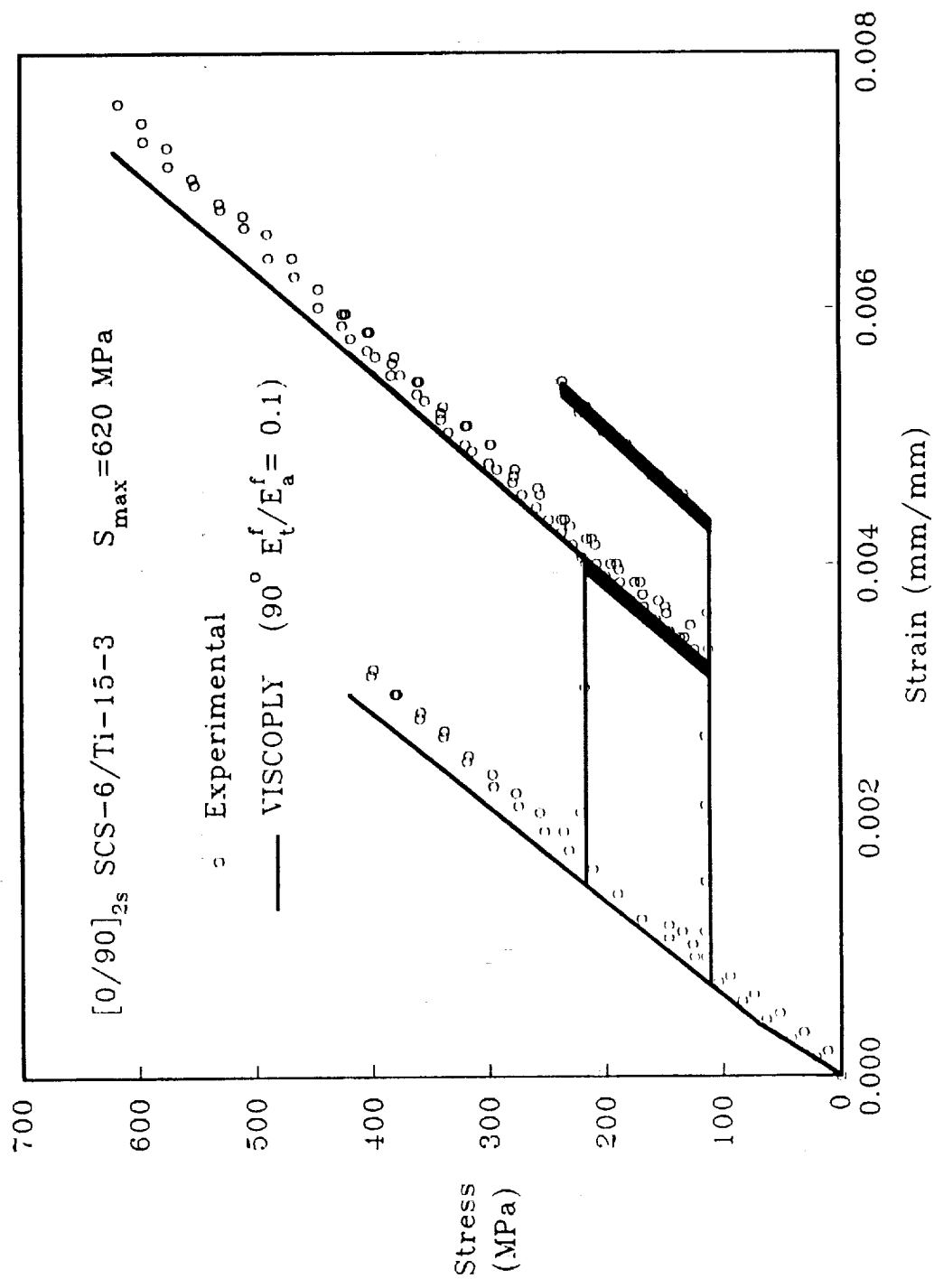


Figure 10. VISCOPLY prediction of composite response to the flight profile at the maximum applied stress of 620 MPa simulating fiber-matrix interface failure of 90° plies.

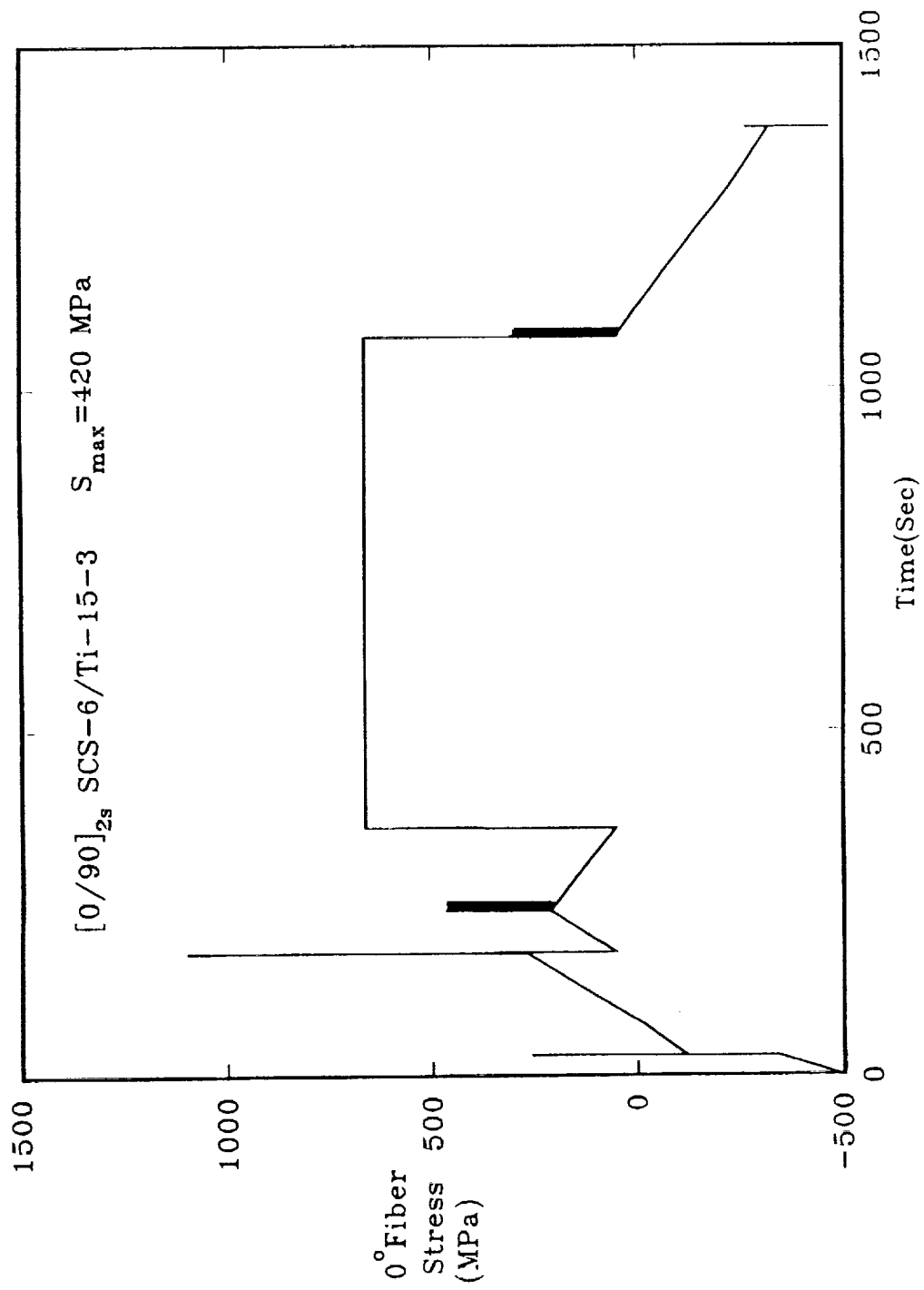


Figure 11. VISCOPLY prediction of 0° fiber stress under flight profile at maximum applied stress of 420 MPa simulating fiber-matrix interface failure of 90° plies.

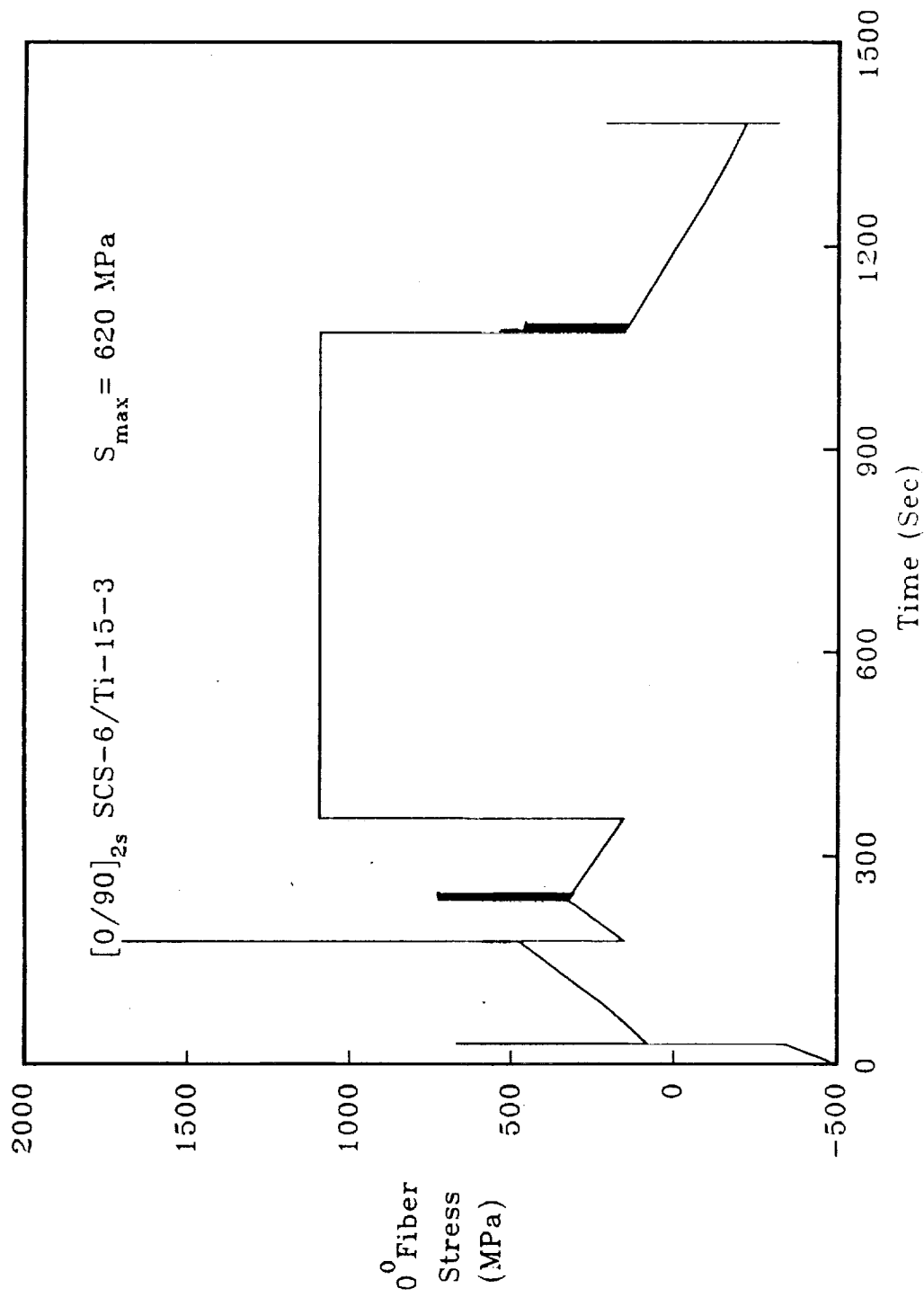


Figure 12. VISCOPLY prediction of 0° fiber stress under flight profile at maximum applied stress of 620 MPa simulating fiber-matrix interface failure of 90° plies.

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13. ABSTRACT (Maximum 200 words) Cross-ply laminate behavior of Ti-15V-3Cr-3Al-3Sn (Ti-15-3) matrix reinforced with continuous silicon-carbide fibers (SCS-6) subjected to a generic hypersonic flight profile was evaluated experimentally and analytically. Thermomechanical fatigue test techniques were developed to conduct a simulation of a generic hypersonic flight profile. A micromechanical analysis was used. The analysis predicts the stress-strain response of the laminate and of the constituents in each ply during thermal and mechanical cycling by using only constituent properties as input. The fiber was modeled as elastic with transverse orthotropic and temperature dependent properties. The matrix was modeled using a thermo-viscoplastic constitutive relation. The fiber transverse modulus was reduced in the analysis to simulate the fiber-matrix interface failure. Excellent correlation was found between measured and predicted laminate stress-strain response due to generic hypersonic flight profile when fiber debonding was modeled.				
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